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Study and development of a small lunar habitat inside Moon Lava Tubes based on ARTEMIS model

Supervisors:
Laura Mainini
Paolo Maggiore

Candidate:
Roberto Landolina

Abstract

In the next years humans will come back to the Moon after decades of absence, marking the beginning of the new space exploration era which has the aim to explore Moon, Mars and beyond. In this context, ARTEMIS program, developed by NASA, plans to land the first woman and next man in 2024; starting from that, ARTEMIS plans to build the first Lunar base at the Moon South Pole to support human surface missions.
Lunar base design presents several challenges to be investigated due to complexity of missions and the harsh lunar environment. Structural problem concerns basically the need to design new light and little bulging solutions to protect astronauts from the biggest hazards of lunar environment (radiation, temperature and meteorites impact). Power challenge concerns the need to develop solutions to overcome the huge power consumption of habitat. These solutions shall consider longer duration of the mission and other features of Moon scenario which bring to exclude some of existing power generation option. Due to long distance between Earth and Moon and the orbital mechanics of the natural satellite around Earth, communication cannot be performed with conventional solutions utilized for satellites; in addition to that, also communication with other part of the Moon is currently impossible due to the lack of communication infrastructure on lunar surface. Habitability and human life and support represents another big challenge to be faced up as there is not experience of life in partial gravity for long missions and so it is necessary to design new systems to ensure safety and wellbeing of astronauts. These are only the most important challenges, but there are still much more gaps to be overcome.
Several studies have been proposing their solution for one or more of these challenges, focusing on the South pole scenario.

The aim of this work is to provide a review of state-of-arts technologies and solutions to overcome the most critical challenges of lunar base design. In addition, this work implements described solutions to propose a preliminary design of habitat for maximum 4 astronauts inside the Moon Lava Tube.
First it is given a general overview about the mission from the habitat set-up up to disposal. After that, some of the most critical gaps will be faced up proposing a feasible solution.
Analysis and data estimation have been carried out building parametric Excel spreadsheets and Matlab scripts which resume the state-of-arts assumptions. The parameters utilized are number of astronauts and mission duration.

This work is composed by nine chapters. Chapter one will be a short introduction to exploration mission and habitability problem on the Moon. Chapter two will be a general overview on reference mission where it will be defined: Mission Architecture, Concept of operations and High Level Requirements. Chapter three will focus on Functional Analysis to identify all systems needed inside the habitat. Chapter four will be focused on structure; in this case it has been proposed an inflatable module to reduce weight and volume of habitat during transportation. Chapter five concerns the habitability and wellbeing problem for which it is proposed an overview of major habitability features. Chapter six concerns power generation; in this chapter it is carried out a comparative analysis between two option to select and size the best power generation solution, nuclear reactors for this study. Chapter seven concerns the communication problem for which it has been proposed a communication architecture to communicate with Earth and on Moon surface. Chapter eight concerns life support capabilities for which it is proposed a general system architecture and it is estimate the main consumables requirements; thermal analysis and oxygen production estimation are also shown. The last chapter gathers all conclusion consideration, resuming the whole proposed design and highlight the technologies need, the still open gaps and the future work to be performed.

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Abbreviations

AIT&V	Assembly, Integration, Testing & Verification
BEAM	Bigelow Expandable Activity Module
C&DH	Command & Data Handling
CCAA	Common Cabin Air Assembly
CLEP	Chinese Lunar Exploration Program
CLTV	Cis-Lunar Transfer Vehicle
CM	Crew Member
CO2RA	CO2 Reduction Assembly
ConOps	Concept of Operations
DSA	Deep Space Antennas
DSN	Deep Space Network
ECLSS	Environmental Control and Life Support System
EMU	Extravehicular Mobility Unit
EPS	Electrical Power System
ESA	European Space Agency
ESPRIT	European System Providing Refuelling, Infrastructure and Telecommunications
EVA	Extra-vehicular activity
F/E	Function/Equipment
FC	Fuel Cells
GaAs	Gallium-Arsenic
GRAIL	Gravity Recovery and Interior Laboratory
HALO	Habitational and Logistic Outpost
HLS	Human Landing System
I-HAB	International Habitation Module
ISRU	In-Situ Resources Utilization
ISS	International Space Station
JAXA	Japan Aerospace Exploration Agency
KRUSTY	Kilopower Reactor Using Stirling Technology
LCNS	Lunar Communication and navigation Services
LCROSS	Lunar CRater Observation and Sensing Satellite
LEO	Low Earth Orbit
LLO	Low Lunar Orbit
LRO	Lunar Reconnaissance Orbiter
LTV	Lunar Transfer Vehicle
MIT	Massachusetts Institute of Technology
MLI	Multi-Layers Insulation
MMDO	Micro Meteorites & Orbital Debris
MRE	Molten Regolith Electrolysis
NASA	National Aeronautics and Space Administration
PCA	Pressure Control Assembly
PEMFC	Proton-Exchange Membrane Fuel Cells
PPE	Power and Propulsion Element
PRETZEL	PREssure PEM water elecTrolyZer tEchnoLogy
RAAN	Right Ascension of the Ascending Node
RFC	Regenerative Fuel Cells

SLS	Space Launch System
SoM	Skidmore, Owings & Merrill LLP
STK	System Tool Kit
UPA	Urinal processing Assembly
US-HAB	U.S. Habitation module
WPA	Water Processing Assembly

1. Introduction

After decades of absence of men on the Moon, in recent years the major space agencies have been working to bring again humanity on Moon opening a new era in space exploration. New space exploration era will have the purpose to prepare humanity for life beyond the Earth; in fact, according to major space program roadmaps, this phase of exploration will include the building and set-up of lunar base camp to support long duration activities on Moon surface.
There are several reasons why space agencies put efforts towards space exploration of Moon and settlement of lunar surface. First, Moon settlement will allow to astronauts and scientists to perform long duration missions without the need to back on Earth every two weeks; performing long duration missions is very important to gather more and more information on the characteristics of the moon and its habitat. Secondary, according to past missions (as for example LRO and LCROSS from NASA), on Moon there are deposit of ice water which can be extracted and used for several applications, as for example fuel for rockets. In addition to water, also Helium-3 can be extracted by lunar Regolith [1]; Helium-3 is rare resources on Earth due the fact that, thanks to atmosphere, particles coming from Sun are blocked. Helium-3 is fundamental to perform nuclear fusion and so the extraction of this resources will allow production of huge amount of clean power in the future. Finally, also concerns resource extraction, it has been discovered the presence of important minerals on Moon surface [1]. In particular rare earth minerals, Lithium, Cobalt and other minerals which are abundant on Moon and will be, due to the growth of electronics market, more and more requested in future on Earth. Harvesting of Moon resources will be one of the reasons one of the reasons that will drive more and more companies to invest in space exploration. In addition to these reasons, Moon settlement represent a fundamental step towards Mars exploration and colonization; in fact, to design new solutions for Mars colonization it is necessary first to make experience of "extra-terrestrial" life on the Moon to qualify new system and identify new gaps on life outside Earth.

Exploration of Moon poses in front several gaps and challenges which shall be face up.
The first challenge concerns the Power generation; electrical power is fundamental for space mission as it feed almost all habitat systems. Producing power on Moon it is not easy due to the large volume of infrastructure needed to produce power as it happens on Earth, and the lack of resources needed to do it. Nowadays to generate power in space there are several different technologies as solar panels, batteries, fuel cells, etc… Depending on scenario features, some of these technologies cannot be used while others required large volume and masses; to choose and design of the correct power generation method is critical to ensure the success of missions.
Another relevant gap in space exploration concerns Communication and Navigation capabilities. During space missions, especially the manned mission, it is fundamental to provide communication capabilities between crew and mission control centres to ensure mission goes on correctly and to provide support in case of emergencies as well as to transmit data about research. At the moment, a solid infrastructure which allow communication between Moon and Earth does not exists; only recently space agencies started feasibility study too build up a network which will allow in future to communicate easily and fast from all parts of the Moon.
Another relevant gap concern habitability feature and habitat module design. As the fact that we have no experience about life on Moon surface in partial gravity condition, environmental and life support systems used in orbit (for example ISS) cannot be used on Moon habitat due to the different conditions. For this reason, a new design of all these systems is needed. What is more, due the fact that cargo missions are more complex compared to those for ISS, it is necessary to design life support systems so that it is possible to recycle as much as possible resources reducing the frequency and costs of missions.
Structure gaps concerns the need to develop innovative structures which can provide protection for astronauts while keeping love mass and volume to get it transportable and to reduce costs of launches.

Other gaps concerns in-situ extraction and resources utilization. According to state of art, it does not exist qualified systems which can produce useful resources on Moon, there are only studies.
Mobility on Moon represents another gap that shall be solved in future studies.
In these years several agencies have built their roadmap to bring again humans on the Moon. China developed Chinese Lunar Exploration Program (CLEP) [2] which has the purpose first to explore moon surface using rovers and landers, then, after 2030, they plan to build the first Lunar base. Program started in 2007 with the first orbiting missions (Chang'e 1) and currently it is in the rovers exploration phase. Concerning human settlement plan, there is not available information in bibliography. Similarly, Russia have been developing the LUNA program which has the same purpose as the CLEM one; an initial robotic exploration of Moon surface and, after the 30', the building of first moon base.
One of the most important program for exploration of the Moon and human settlement is the American NASA Artemis program [3], which has the aim to build the needed infrastructure to allow future permanent or semipermanent settlement of human on the Moon; the program consists different phases.
The first is a preparation phase during which all supporting elements for future mission will be designed and built, as for example the new launch system or the new capsule.
In the second phase has the objective to build a new space station orbiting around the Moon in a low lunar orbit, the Lunar Gateway; this station will be used both to carry out experiment in deep space condition and to support logistically the future missions on Moon surface. According to NASA plan the building should start in 2024 and it should be completed around in 2028. It is important to point out that different space agencies support NASA on building of this station, demonstrating a great international cooperation on space exploration objectives.
The third phase of ARTEMIS program is the land, after decades, of first women and next men on Moon surface. This is planed on ARTEMIS III mission in 2024 (according to current plan). After this event, ARTEMIS program will officially enter in the next phase which has the aim to built and consolidate the needed infrastructure to allow a safe settlement on surface. In particular it will be necessary to bring on the Moon alle needed equipment to set-up the first Moon base, which is supposed to be built on south pole starting in 2028; what is more, it will be necessary to consolidate the orbiting systems to provide navigation and communication capabilities.
Also for ARTEMIS program there are not available detailed information about the moon base planned.
A relevant preliminary study concerning the design of habitat on south pole as been made by the European Space Agency (ESA) in collaboration with MIT and Skidmore, Owings & Merrill LLP (SoM) [4]. This study proposed his own vision to overcome some of the existing gaps in settlement problem.

The aim of this study is to provide an overview of state-of-arts technology and solutions and to implement them into a real habitability problem. In particular, in this study it will be conducted analysis on design of lunar habitat inside the Lava Tube.
Lava Tubes represent a very interesting scenario as the provide some important advantages for design of habitat but some additional challenges. The most important advantages of building habitat inside the Lava Tube concerns the free protection from radiation and impacts of meteorites; what is more, as it will be explained in following chapters, as the temperature remain roughly constant, there will not be the problem of huge temperature gradient which could affect structure and systems. Even if it has not been proved yet, Lava Tube could be optimum candidate to store up ice water, essential to increase the sustainability of mission. The biggest challenge of Lava Tube is accessibility; according to the few information we have about these, most of skylights represents point of possible Lava Tubes where the roof is collapse [5]. It is expected these points could not be used as access point due to rock which block the access. In addition to that, due to hight of holes and the inclination, it is not possible to reach this area with simple rovers, it is necessary to design new systems to allow safe access inside the cave. Another challenge concerns the power production; unlike what happens for South Pole bases, which can experience up to 86% of time in sunlight condition, in equatorial region of Moon, where possible Lava Tube have been found, sunlight periods last

half Moon day (roughly 14 Earth days) with the other half day in eclipse; this represent a limitation for power generation with solar panels and force to considers other technologies.
Based on these consideration and taking as reference the preliminary studies conducted for South pole habitat, in this work it has been proposed an implementation for this little discussed scenario.

In the first chapter it will be given an overview of whole the mission from the habitat building up to operation phase; the mission overview will define the ConOps and Mission Architecture as well as the High Level Requirements.
In the second chapter focus will be directed to only the habitat system in a specific phase of mission, the operational phase. Here it will be conduced a general functional analysis which has the aim to identify the high level functions and decompose them building the functional tree. In addition, it will be suggested a general system architecture for each high level function.
In the third chapter the focus will be on structure; It will be analysed the state-of-arts concerning structure and it will be suggested the own configuration.
The fourth chapter will focus on accommodation function; here it will be defined all needed equipment to enable habitability of habitat and it will be given a first estimation of mass and volume.
The fifth chapter will focus on power generation problems; here it will be define the power requirements and it will be conduced a comparative analysis to choice the best solution for power generation.
The sixth chapter will focus on communication; here it will be defined the communication architecture and given a first estimation of data volume required. After that communication elements will be preliminary designed.
The seventh chapter will be focused on environmental control and life support system; here it will be defined all requirements and it will be conduced a preliminary budget analysis to estimate volume and masses of atmosphere gases, water and food. Preliminary thermal analysis has been also carried out.
In the final chapter it will be given a general overview of habitat design and it will be highlighted all remain gaps, technology needs and open point.

2. Mission Overview

Design of habitat requires a deep understanding of the scenario within which the moon base will be developed, and all infrastructures needed to support first set-up mission and then operational missions.
As mentioned in the introduction, Moon scenario and In particular Lava Tubes one, they pose various challenges to be faced and various constraints to the mission.
In this chapter it will be given a general overview on mission design starting from mission objectives arriving to define the high level requirements which will represents the starting point for further habitat design.
First it will be given a short introduction about mission objectives for which habitat could be used.
Secondary it will be proposed a concept of operations, which represents how the mission will work from the set-up of habitat up to operational phase. Concept of operations will highlight constraints and challenges of each scenario where the mission will be.
After that it will be proposed the Mission Architecture which has the aim to explain which elements compose the mission. Each element will be explained, and the state-of-arts will be presented.
Finally, it will be proposed a set of initial high level requirements coming from missions consideration and assumptions made from other similar projects.

2.1 Mission objectives

The main objective of the mission is to prove the possibility to live inside Lava Tubes on other planets outside Earth; It represents a key test to plan future inhabitability strategy for future missions on Mars and beyond.
In addition to this, mission has other side purposes. The first side objective is to study in loco Lunar Lava Tubes improving knowledge gathered by robotic missions; in particular, it will be studied: morphology of caves, internal mapping of all surfaces (walls, floor and roof), temperature variation and soil composition.
The second objective is to provide an further laboratory on Moon surface to carry out experiments on Moon surface in a different scenario compared to South Pole one; during the stay inside the cave, crew will also carry out different experiment and will test new technologies which will be used in future missions.
Finally, it is useful to enlighten that in future this habitat, and his supporting systems, could be used to host an expanded population and support colonization strategy of the Moon.

2.2 ConOps

The Concept of Operations describes how the mission will work from the Assembly, Integration and Testing phase to disposal phase. For each phase it is described: which are the main scenarios, which are the main constraints and challenges, which are the main requirements and how elements described in Mission Architecture will work together to achieve the mission objectives.
In Table 2.2-1 is presented the main phases of this mission and their main scenarios.

Phase	Description	Duration	Scenarios	Scenarios duration
Earth operations	This phase involves all the operations that must be carried out on Earth before the launch of systems to ensure the correct functionality the system and to prepare for launch	about 2 years	Assembly, Testing and Integration	Up to 2 years
			Transport to Launch site	Up to one week
			Packing on Launch vehicle	Up to one week
Launch/Transfer and Landing	This phase involves all the operations needed to bring the system from Earth to Moon by using one of reference Launche system as vehicle	up to 1 week	Launch from Earth	6 hours
			Trasfer to Lunar Orbit	3-5 days
			Landing on lunar surface	6 hours
			Trasfer to Deployment site	about 2 days
Set Up of Habitat	This phase involves all the operations needed to build up the base both deploying structure and setting up vital systems	TBD	Power Plant set up	TBD
			ISRU and storage unit set up for ECLS system	TBD
			Communication infrastructure set up	TBD
			Habitat deployment and internal systems set-up	TBD
New crew settlement	This phase involves all the operations needed to bring a new crew from Earth to Lava Tubes	up to 1 week	Launch from Earth	6 hours
			Trasfer to Lunar Orbit	3-5 days
			Landing on lunar surface	6 hours
			Trasfer to Lava Tubes habitat	about 2 days
Operations	This phase involves all the activities performed by crew during their permanence in the habitat	5 years	EVAs activities inside Lava Tubes	from 4 to 8 hours for EVA
			EVAs activities on Moon suface (outside Lava Tubes)	from 4 to 8 hours for EVA
			Living time inside habitat	up to 180 days per crew
			Sleeping time inside habitat	about 8 hours per day
			Work out inside habitat	from 1 to 2 hours per session
Crew back to Earth	This phase involves all the operations needed to allow safe return of crew on Earth	up to 5 days	Trasfert to landing site	up to 1 day
			Ascent to Lunar orbit	6 hours
			Trasfert to Earth orbit	3 days
			Re-entry on Earth surface	6 hours
Cargo	This phase involves all the operations needed to bring cargos to Lava Tubes habitat	up to 1 week	Launch from Earth	6 hours
			Trasfer to Lunar Orbit	3-5 days
			Landing on lunar surface	6 hours
			Trasfer to Deployment site	about 2 days
Disposal	This phase involves all the operations needed to dispose or grow habitat	TBD		

Table 2.2-1 Phases and scenarios of ConOps

This work will focus on habitability phases during which crew will carry out their experiments inside habitat and perform exploration or maintenance operation during EVAs activities, for this reason it will be given only a short description each phase and it will be given a in deeper overview about Operations phase.

2.2.1 Earth Operation

This scenario includes all activities which have to be performed on Earth before to launch all systems. The starting point of this phase is the Assembly, Integration and Testing activity which has the objective to validate and verify all requirements following the AIT&V Plan prepared for the mission. Test and assembly are usually carried out in clear room and under Moon-like condition (when possibly) like vacuum condition, radiation condition and low temperature condition.

Once the testing campaign is completed, the systems are ready to be launched and shall be transported from testing facility to launch site. This phase is quite critical, because systems shall be protected from any damages and contamination on Earth and it requires specific protocols to transport. In addition to that, it is important to enlighten that systems shall be compacted enough to be transported by common terrestrial vehicles which have maximum size to transport; typically planes or ships are used for this purpose depending on location of launch site and size of systems.

The last step of Earth operation is the integration with launcher. During this phase, the systems are packed inside the launch vehicle fairing and get ready for launch. It is important to notice that fairing has maximum weight and volume that could be (see Mission Architecture section) and it represents constraints for packaged systems.

Earth Operations	
Short description	This phase includes all activities carried out on Earth to prepare systems to launch
Starting point	All components ready to be integrated
Final point	Systems packed inside launcher fairing
Interfaces	- Ground support equipment (testing) - Transportation vehicle (on Earth) - Launcher Adapter and Fairing
High Level Requirements & Constraints	- System shall be able to be tested in Lunar-like environment - Systems shall be able to be transported inside terrestrial vehicle - Systems shall be withstand to transportation loads on Earth - System shall be able to be packed inside fairing of launcher
Challenges	-

Table 2.2.1-1 Earth Operations phase resume

2.2.2 Launch/Transfer and Landing

This phase includes all activities needed to bring the payload from Earth surface to Moon surface; It has assumed that lander will bring payload near the supposed existing Lunar base on South Pole.
The first scenario is the launch from Earth, the main objective is to bring payload from Earth surface to a parking orbit where the finals stage of launcher, the transfer vehicle and payload will remain up to they will able to perform the transfer maneuvers to get into a lunar transfer orbit; it is assumed that the parking orbit is a LEO orbit. The launch represents hazard for the payload because during this phase both vibrations and thermal flows will be high and can damage the payload, for this reason it is necessary to design the system to withstand to this launch environment. What is more, once the final stage reaches the LEO parking orbit, payload will be subjected to microgravity condition and it is so necessary to design payload so that minimal function can be performed also in this condition.
Once launcher reach point in orbit to perform the lunar transfer maneuver, payload will be subjected to a different environment that can be considered Deep-space environment. This environment is much more dangerous than LEO one due that higher radiation which can represents a problem both for avionics systems and for communication; in this phase, in fact, it is important to gather housekeeping data to monitor the health status of payload, this represents an important requirements that should be take into account.
The final stage is landing on Moon surface, it is a critical step because it is needed to design all system to allow a smooth arrival to lunar soil. The environment is similar to deep space one and location is considered the South Pole, where it will present a first lunar base. Critical aspects of this phase concern design the need to protect payload during landing from vibration and landing load.
Once systems are landed at Lunar base in South Pole, it is necessary to transfer it to Lava Tube site. The transportation will be carried out thanks to pressurized rover which will have to be large enough to contain payloads. One of the most critical challenge to front up to is to carry inside all systems. Due to their shape, skylights discovered will not allow rovers to get inside cave, this means it would be probably needed to

design new rovers capable to reach the site inside cave or to design systems to bring payloads from lunar surface to deployment site inside cave. This represents an open point which shall be studied in future.

Launch/Transfer and Landing	
Short description	This phase includes all activities related to transportation of systems from Earth to landing site on Moon
Starting point	Systems packaged inside launcher fairing on Earth surface
Final point	Systems packaged inside lander on Moon surface
Interfaces	- Launcher Adapter and Fairing - Service module of transportation vehicle - Lander - Ground station system
High Level Requirements & Constraints	- Systems shall withstand to launch loads - Systems shall withstand radiation environment during transfer to moon - Systems shall withstand to thermal environment during transfer to moon - Lander shall land not far from expected landing site - Systems shall be able to communicate to ground station system all housekeeping data - Systems shall withstand to landing loads on Moon
Challenges	- To transport payload inside cave

Table 2.2.2-1 Launch/Transfer and Landing phase resume

2.2.3 Set Up of Habitat

This scenario involves all activities performed to set up habitat inside the cave. It includes deployment and systems set up.

Set up takes place in different moment as it is possible that the different parts of habitat are delivered in different moment. Typically, studies suggest to first set up additional systems as Power plant and communication plant and after that, habitat could be deployed; airlock is also added when habitat have been deployed. [6].

During set up activities, astronauts shall have a base point where live, this because activities can last many days and it is non convenient come back to South pole base every day. This means pressurized rovers shall be designed to support human life for some days. Also, as the equipment used during EVAs has a maximum of oxygen storable, duration of extravehicular activities is limited; this means that it is necessary to conclude the set up within the limits imposed by EMU (Extravehicular Mobility Unit). Another requirement concerns the need to have a source of power to support set up activity; this source of power could be represented by batteries with can provide a wide range of power for short period of time. Also communication is required during the whole activity period to receive support from mission control and to send feedback. Unluckily communication inside the cave is not easy, especially if habitat is not yet deployed, for this reason, it represents still challenge to be face up.

Set Up of habitat	
Short description	This phase includes all activities performed to set up habitat
Starting point	Stocked module and system are on the deployment site
Final point	Module is deployed and system installed
Interfaces	- Humans
High Level Requirements & Constraints	- Deployment duration shall be lower than TBD hr - Pressurized rovers shall support humans during all duration of set up activities - Deployment technologies shall be included in design of habitat - During whole set up process it is necessary to guarantee real time communication at least with Earth - Electrical power source shall provide the needed power to set up habitat
Challenges	- To deploy habitat and set up in the shortest possible time - To provide communication capabilities during set up mission

Table 2.2.3-1 Set up of habitat phase resume

2.2.4 New Crew Settlement

This scenario involves all activities needed to bring new crew from Earth to Moon base. In this work we will consider only a stay inside the Lava Tube base with maximum length of 180 days from the departure to the re-entry on Earth. It is necessary to marks that it is possible to include also stay period on others bases during the stay outside Earth as for example stay on Lunar gateway in orbit or stay in South pole base, this would require a different ConOps which is not presented in this work.
Operations performed during this phase are similar to those presented to bring systems on Lava Tubes base (section 2.2.2); however, it is important to point out all systems shall support human presences and interface with it.
Some major requirements includes: need to provide real time communication between Earth and/or others lunar bases during whole travel, need to include life support equipment in every manned transfer vehicle (Lunar transfer vehicle, Descend Module, Rovers, etc…) and presence of emergency protocol and equipment on transportation systems. As said for habitat systems, it is needed to think about systems to allow access to base inside cave. This represents another open point for this study.

New Crew settlement	
Short description	This phase includes all activities related to transportation of crew from Earth to Lava Tubes base
Starting point	Crew is ready for mission on launch site
Final point	Crew is settled inside Lava Tube base
Interfaces	- Humans - Ground station system - Lunar bases - Landers - Pressurized Rovers
High Level Requirements & Constraints	- Transportation systems shall provide real time communication during whole length of travel - Transportation systems shall have life support equipment - Transportation systems shall have emergency equipment in case of failures - Transportation systems shall provide a user friendly interface with human - Transportation systems shall protect crew from radiation environment
Challenges	- To transport crew inside Lava tubes

Table 2.2.4-1 New crew settlement phase resume

2.2.5 Operations

This scenario includes all activities performed during missions inside the habitat, included extravehicular activities. As said before, we will consider a standard mission length of 180 days per crew but habitat will be considered to be used for at least 5 years since the built up.
As in aerospace field safety is the most important drivers, it is necessary habitat can guarantee an high level of safety during the mission. As Lava Tube represents a more isolated scenario than surface one (example South Pole) due to struggle in accessibility and so in escaping during emergency situation, it is necessary to design systems in such a way as to be able to work also in suitably described emergency scenario. This kind of design includes the possibility to have real time communication in every moment of the mission at least with Earth and mission control centre both to receive commands from Earth and to send housekeeping data to terrestrial ground segment. What is more, it is necessary to store a sufficient quantity of utilities (as water, air, energy, food, etc…) to make up in case of unavailability for a short period of time or for the period needed to leave the habitat and come back to South pole base or Earth.
Beside emergency situation, habitat shall support crew for all activities performed during the mission which includes: daily experiments and laboratory work, EVAs, sleeping, free time and relax and workouts. All these activities require specific systems which shall interface with humans and allow them to survive and work; in the following Table is reported a list of high level requirements for all major systems.

Operations	
Short description	This phase includes all activities performed by crews during their permanence inside habitat
Starting point	Crew is settled inside Lava Tube base
Final point	Crew is ready to leave Lava Tube base
Interfaces	- Extravehicular Mobility Unit - Habitat systems - Pressurized Rovers
High Level Requirements & Constraints	- Habitat shall support up to 4 crew members for 180 days - Habitat shall provide structural protection to crews during missions - Habitat shall support human wellbeing - Habitat power system shall provide electrical power to crew - Habitat communication system shall provide communication capabilities with: Earth, EMU suit, Pressurized rovers and other surface Lunar bases - Habitat shall support human life - Habitat shall provide hygiene and medical capabilities - Habitat shall support EVA activities - Real time communication with Earth shall be guaranteed 24h per day - A minimum reserve of utilities (energy, water, air, etc…) shall be guaranteed in every time
Challenges	- To guarantee high level of crew safety - To allow crew to carry out EVA activities on Surface without use of pressurized rovers - To generate electrical power - To provide communication both inside and outside habitat

Table 2.2.5-1 Operation phase resume

2.2.6 Crew back to Earth

This phase includes all activities needed to allow safe return to Earth of crew after the mission on Lava Tubes.

This scenario is very similar to one described in section 1.2.3 and so it will be given only a resume on key points on following Table.

Crew back to Earth	
Short description	This phase includes all activities related to transportation of crew from Lava Tubes to Earth
Starting point	Crew is ready for return inside Lava Tubes habitat
Final point	Crew is landed on Earth
Interfaces	- Humans - Ground station system - Lunar bases - Ascend Module - Pressurized Rovers
High Level Requirements & Constraints	- Transportation systems shall provide real time communication during whole length of travel - Transportation systems shall have life support equipment - Transportation systems shall have emergency equipment in case of failures - Transportation systems shall provide a user friendly interface with human - Transportation systems shall protect crew from radiation environment
Challenges	-

Table 2.2.6-1 Crew back to Earth phase resume

2.2.7 Cargo

This phase includes all activities needed to transport cargo from Earth to Lava Tubes habitat. Cargo includes all utilities need to support life of crews (water, air, food) and their operations (experiments, devices, back up systems, etc…).
In this study it has been assumed to perform one resupply cargo mission per missions; that means cargo will resupply habitat once every 6 months. Based on this consideration, in following chapters it will be considered 180 days as reference duration to estimate the mass and volume of consumables to be launched during cargo missions.
All aspects related to requirements, interfaces, constraints and challenges has been shown in section 2.2.2 for habitat transportation and are very similar to cargo's one.

Cargo	
Short description	This phase includes all activities related to transportation of cargos from Earth to Lava Tubes habitat
Starting point	Cargo packaged inside launcher
Final point	Cargo arrived at Lava Tubes habitat
Interfaces	- Launcher Adapter and Fairing - Service module of transportation vehicle - Lander - Ground station system
High Level Requirements & Constraints	- Cargo shall withstand to launch loads - Cargo shall withstand radiation environment during transfer to moon - Cargo shall withstand to thermal environment during transfer to moon - Cargo shall land not far from expected landing site - Cargo shall able to communicate to ground station system all housekeeping data - Cargo shall withstand to landing loads on Moon
Challenges	- To transport cargo inside cave

Table 2.2.7-1 Cargo phase resume

2.3 Mission Architecture

The Mission Architecture describes all elements which will support the mission from first stages to operation phases, comprising communication architecture and ground support elements.
All elements summarized in Table 2.3-1 represent the state of arts or elements which will be available in near future, in any case these will be described in depth in the following sections.

Element	Choices
Subject of Mission	Moon
Location	Lava Tubes on Marius Hills (14.100°N; 303.262°E)
Space Segment	- Lunar Transfer Vehicle - Human Descend & Ascend Vehicle - Lunar Gateway (not mandatory)
Launch Element	Space Launch System (SLS) or Super Heavy Rocket+ Starship (by SpaceX)
Communication Element	- On Earth: Nasa Deep Space Network or ESA Deep Space Antennas (or LGS Lunar Ground Stations) - In Moon Orbit: Satellite constellation in Moon orbit for Communication purpose - On the Surface: Wide range antennas for communication with satellite constellation in Moon Orbit
Ground Element on Moon	- Pressurized Habitability Module for 4 people - Airlock - Power Plant - Communication - ISRU Plant - Pressurized rovers (auxiliary)
Ground Element on Earth	Mission control centre network

Table 2.3-1 Mission Architecture for Lava Tubes scenario

2.3.1 Location

Large part of information we currently have about Lava Tubes on moon come from the Japanese mission SELENE-Kaguya [5] which for the first time detected a big hole on lunar surface (called 'skylight' opening) where it has supposed the existence of a Lava Tubes. This skylight was found on Marius Hills region and his opening is 65 meters in diameter [7]. Studies of SELENE were followed by other missions like NASA-LRO and Indian-Chandrayaan-1 which gathered more photos and data about possible new lava tubes on equatorial region.
The most important studied and detected surface holes known at the moment are listed in Table 2.3.1-1 [7].

Location	Size of hole [m]	Depth	Angle to respect long axis
Marius Hills	65 x 65 meters	48 meters	-
Mare Tranquillitatis	98 x 84 meters	107 meters	165 degrees
Mare Ingenii	118 x 68 meters	45 meters	36 degrees

Table 2.3.1-1 Lists of main skylight holes find on the Moon

After having detected holes, it was carried out further studies on regions near these to understand if it is possible these holes could be collapsed spot of underground caves or Lava Tubes. The main study has been carried out in 2011 by NASA spacecraft GRAIL, which studied the region around Marius Hills Hole by using gravity field data [8]. GRAIL is able to analyse information about change in gravity field to identify lack of mass under the surface, which could be caverns or lava tube, in particular change of gravity field could be generated only by large lack of mass in the order of several tens of kilometres. Results of mission showed change in gravity and so it allows to suggest the existence of long tubes in that zone.
Over the time other missions, including SELENE itself, confirmed this theory using other technic of analysis as radar echo pattern of Lunar Radar Sounder instrument on SELENE. [5]
Thanks to all these studies, it is possible to conclude with an high level of confidence, that Marius Hills region hosts several possible lava tube which would extend for at least several tens of meters (data not sure).
In this study the reference location will be the Marius Hills (Fig. 2.3.1-1) region in particular the region which have been studied by SELENE and GRAIL as the high probability to find there a Lava Tube.

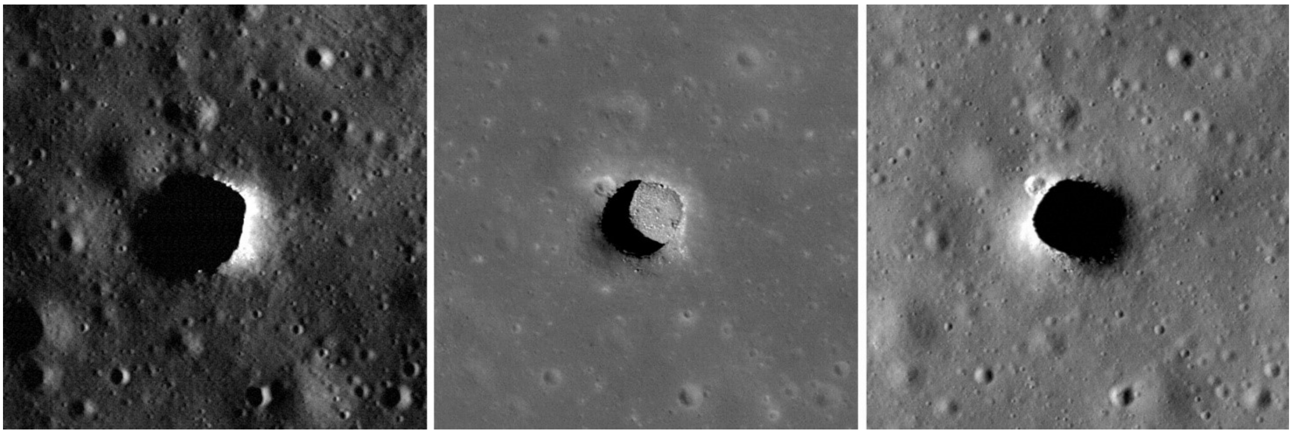


Figure 2.3.1-1 Marius Hills Hole by NASA's Lunar Reconnaissance Orbiter (credit by ESA)

2.3.2 Space Segment

Space segment in this study includes all elements and systems which will support all phases of the mission from set-up to habitability. Main systems discussed in this sub-section include: Lunar Transfer Vehicle, Human Descend & Ascend Vehicle and Space Lunar Gateway.

Lunar Transfer Vehicle

The Lunar Transfer Vehicle is a critical system to both transport crew from Earth to Moon and also cargo needed for the mission. This system shall be able to support crews for the whole duration of travel between Earth and Moon providing it power capabilities, communication capabilities and life support.
Currently main plans for LTV concerns the capability to transport humans and cargo from low Earth orbit to

Lunar orbit and in particular to Lunar Gateway which will be orbiting around in a cis-lunar orbit starting to 2020s. For these purposes, It will be able to dock the Gateway and provide cargo from Earth or transport samples gathered on Moon surface to Earth, in addition to crew transport capabilities.
The main reference LTV for Artemis programme is Orion capsule which include the crew module designed by Lockheed Martin and the European service module designed by Airbus. Orion will be able to support a maximum of 4 crew members for the whole mission and up to 21 days in deep space condition. It will be launched by the new American launch system (SLS-Space Launch System) and it is able to dock the Lunar Gateway. What is more, it is also able to re-entry to Earth landing safely on terrestrial oceans (thanks to heat-shield). Orion is on the final stages of test campaign and it will be launched in the first Artemis mission to support the exploration programme in future. [3]
In addition to Orion, also two new CLTV (Cis-Lunar Transfer Vehicle) have been designed by European companies: Airbus and Thales Alenia Space. These two Transfer Vehicle will be designed to support future European exploration programmes and increase cargo capabilities between Earth and Moon not only for scientific purpose but also for commercial one [9,10].

Fig 2.3.2-1 Rendering of Orion Capsule (credit to Wikipedia)

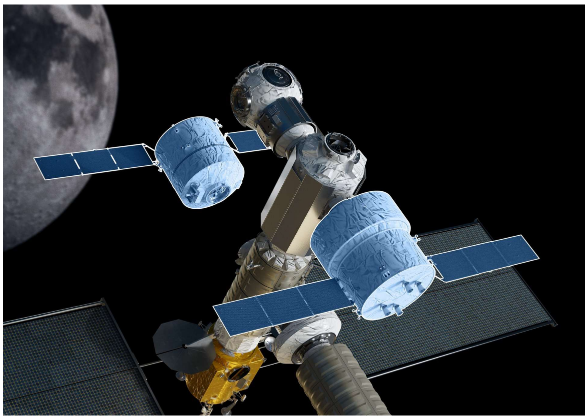


Fig 2.3.2-2 Concept of Lunar Transfer Vehicle by Airbus Space

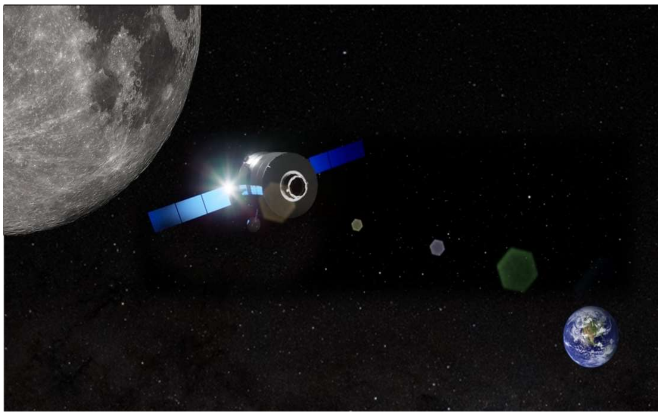


Fig. 2.3.2-3 Concept of Lunar Transfer Vehicle by Thales Alenia

Human Descend & Ascend Vehicle

The human descend and ascend vehicle is a fundament system needed to allow crews access to lunar surface; it will be used to transport humans from lunar orbit to Moon surface and to transport astronauts

from surface to lunar orbit ready for back trip to Earth. According current plans, Human Landing System (HLS) will be able to bring humans both from Lunar Gateway and from launcher vehicles (except Starship which represents the HLS itself).
NASA has promoted a challenge between three big companies to choice the best option and design of the human landing systems used for ARTEMIS program; the three companies are: Dynetics, Blue origin and Space X. The aim of challenge is to identify the best option between the three suggested system, using as choice criteria: cost, safety and sustainability. In April 2021, NASA has chosen Space X Starship as winner of competition due to reduced costs and more sustainable project.
As concept of SpaceX and Blue origin represents the major two different model of HLS, they will be introduced below; Dynetics system represents a single module solution similar to both the other ones and so it will not introduced in this work.

The Blue Origin human landing system is the solution developed by Blue Origin and partners to allow access to lunar surface both to humans and cargos; all information available in this paragraph are directly taken by Blue Origin and partners website and ARTEMIS program document [11].
Blue Origin HLS is able to interface and dock both Orion capsule and Lunar Gateway station, this makes solution very flexible in utilization and allow multiple different usages for the following missions to the Moon. The HLS is composed by three different modules designed and built by important partners: Descend Element, Ascend Element and the Transfer Module.
The Descend element is designed and built by Blue Origin itself. The element is a lander capable of landing in every part of lunar surface, included South Pole transferring both crew and cargo from lunar orbit to Moon soil. The element uses autonomous guidance algorithms which allow it to safe land on Moon. At the moment, there are not available data about mass and volume constraints. During land phase, this element and Ascend one separate from transfer element and land autonomously.
The Ascend element is designed and built by Lockheed Martin. This element will be used to bring back crew or cargo from Moon surface to lunar orbit. The system will have ECLSS capabilities as well all needed software capability to perform ascend from Moon.
The transfer element is designed and built by Northrop Grumman. This element will bring the descend and ascend elements to lunar orbit maximizing as well the mass to be transported to the Moon (both crew and cargos). The reference concept for this element is the Cygnus cargo module, used to resupply the ISS, this guarantee a great amount of experience on design of this element.
All elements use liquid hydrogen and oxygen thrust solution which allow to generate and control thrust during all phases of mission.
At the moment there are not available data about mass and volume constraint for this system.

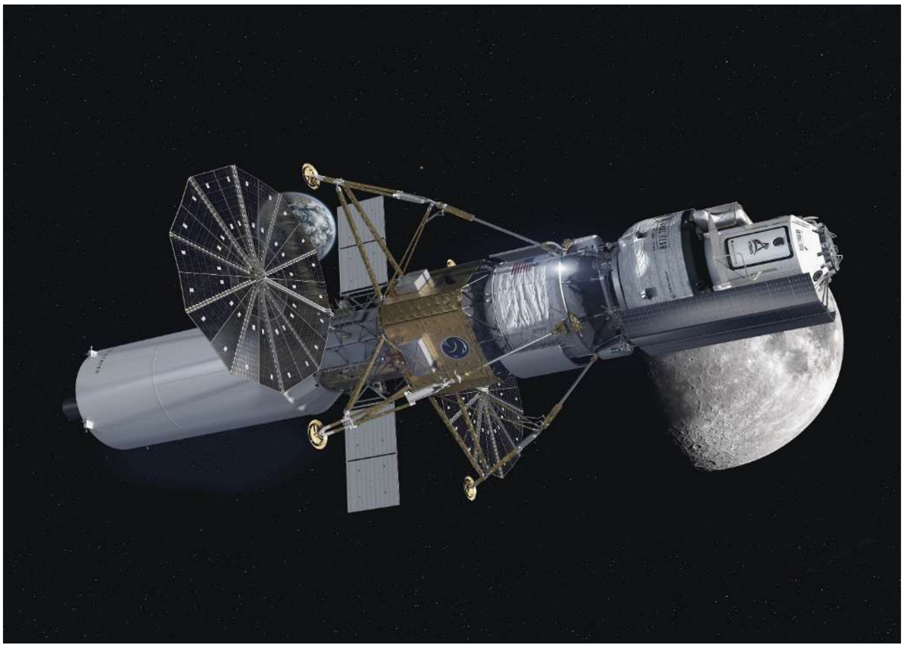


Fig. 2.3.2-4 The Blue Origin and partners three elements HLS architecture (Credit by Blue Origin)

SpaceX proposed to use the Starship as landing solution to bring both crews and cargos to the Moon. It represents a single module solution which will be launched with the SpaceX Heavy Rocket. According to Space X this solution should be able to dock both the Lunar Gateway and Orion capsule allowing a great level of flexibility for future missions.

According to SpaceX plans, it will be possible to launch Orion capsule and crews with SLS from Earth, perform a docking maneuvers on low lunar orbit with Starship, transfer crew from Orion to Spaceship and then land the SpaceX vehicle, this will allow multiple and more sustainable mission to lunar surface.

Constraints related to mass and volume for Starship systems have been reported on paragraph 2.3.3.

Lunar Gateway

Lunar Gateway is the international project proposed by NASA and with support of other space agencies to build an orbiting space station in lunar orbit, all information presented in this section come from the official NASA Artemis overview document [3, 11]. The Lunar gateway has a double role within the space exploration framework; first it would be an orbiting laboratory to study deep space condition on humans and avionics and sun activities. Experiments carried out during permanence inside gateway includes study of Sun activities measuring the frequency and magnitude of solar events thanks to payload as NASA HERMES (Heliophysics Environmental and Radiation Measurement Experiment Suite), in addition to that it will be studied the dose and effect of radiation outside the Van Allen belt, it represents an optimal simulation of deep space condition similar to one experimented on Lunar surface. The second main objective of Gateway is the Logistic role inside the exploration program; it would be a supporting element for Lunar surface activities included communications capabilities and ascent/descend activities of human crews.

The reference orbit chosen for Lunar Gateway will be a HALO orbit with apogee of 70,000 km and perigee of 1500 km, this has been chosen both for orbital stability reason and to allow simpler access to different parts of Moon, especially Poles.

According to Artemis framework, Lunar Gateway assembly will start in 2023 when the first two modules, PPE and HALO, will be assembled together on Earth and launched, will be realised on target orbit. After that other 5 modules designed and produced by several international partners will be launched within the 2028

and assembled directly on lunar orbit.
The final set up of Lunar gateway includes seven modules:

- *PPE*: The *Power and Propulsion Element* is the main module dedicated to electrical power handling and electrical propulsion capabilities. Thanks to solar arrays PPE is able to gather electrical power from Sun, control and distribute it to all Gateway loads. For Lunar Gateway project It has been estimated a power consumption above 60 kW. In addition, thanks to electric propulsion, it is able to carry out corrective maneuvers both to change orbit and to change attitude.
  Finally, it has also communication and thermal control capabilities.
  It will be assembled with HALO module and launched in a single rocket in 2023.

- *HALO*: *The Habitational and Logistic Outpost* is the first habitable module to be launched to host crews during permanence on Gateway.
  It has 55 m^3 habitable volume and ECLSS capabilities useful to support human activities in first stage of gateway building and Operations. The design is similar to Cygnus vehicle with the addition of communication and thermal control capabilities; also, it has docking ports useful to support robotics activities and lunar surface descends as well as docking of lunar transfer vehicle from Earth.
  It will be assembled with PPE module and launched in a single rocket in 2023.

- *ESPRIT*: The *European System Providing Refuelling, Infrastructure and Telecommunications* is the module design by Airbus and Thales Alena Space to provide further communications capabilities both with lunar surface outposts and Earth and to store xenon and hydrazine useful to perform corrective maneuvers of Gateway. It has docking ports to support cargo and refuelling missions to Gateway.

- *I-HAB*: The *International Habitation* Module is the central habitable module designed by Airbus and Thales Alenia Space to support human activities inside Gateway. It offers sleeping, galley and training spaces to crews; what is more, it represents the core module linked to the others module, that means it will allow movement inside the station.

- *U.S HAB*: The *U.S. Habitation* module is the habitable module designed in U.S which has similar functions to other habitable modules.

- *Gateway Airlock Module*: It is the module used to perform EVAs activities. It will be designed by Russia.

- *Gateway Logistics Modules*: They are independent systems used as cargo vehicles to refuel station, transport experiments to gateway and collect samples and needed items during lunar surface mission; In addition, they can be used also to dispose thrash from station.
  It willable to transport up to 5,000 kg of pressurized payload or 1,000-2,600 kg of unpressurized payload.

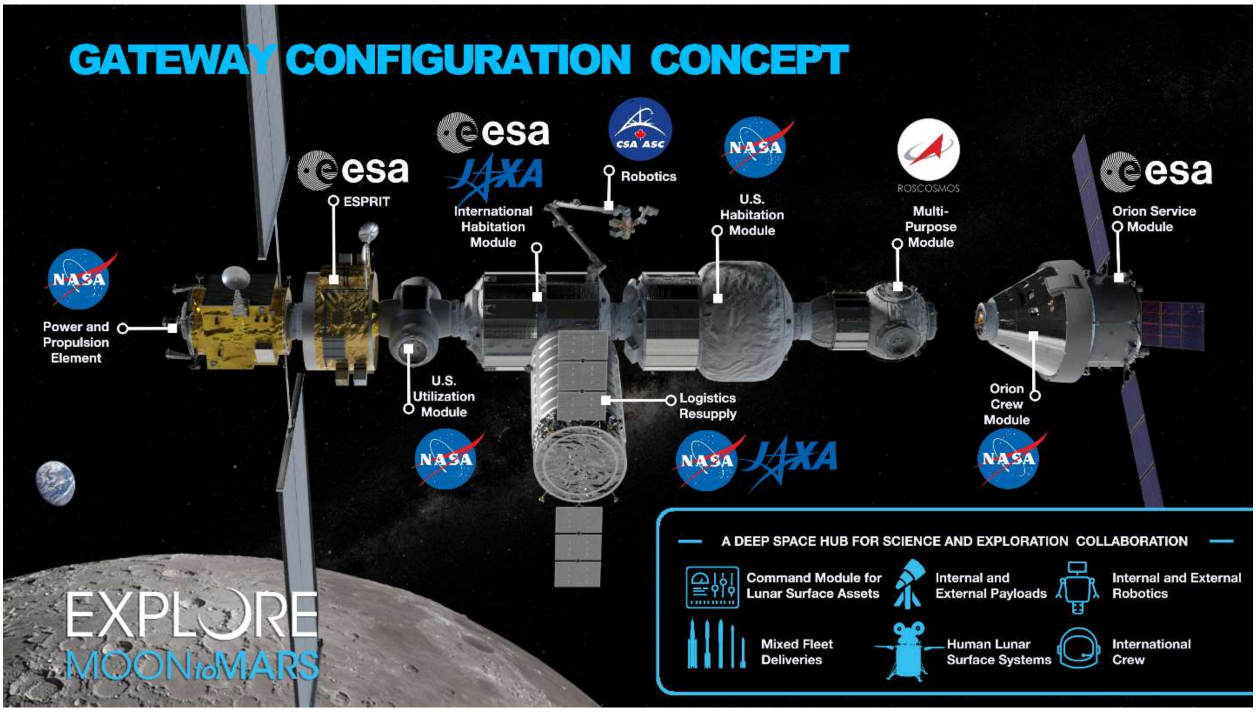


Fig 2.3.2-5 Lunar Gateway (credit to Wikipedia.org and NASA)

2.3.3 Launch Element

Nowadays there are two main launch systems which is considered to use for future exploration mission on the Moon: Space Launch Systems (SLS) by NASA and Super Heavy Rocket + Spaceship solution by SpaceX. Both these two solutions are currently in final stage of testing and they will be used in the following years for Artemis program missions and even Mars missions.
In this sub chapter it will be given a short review of both solution and it will be given some high level requirements useful to plan transportation of all systems and habitat and cargo missions too; what is more, requirements will also be a constraint for some choices made during the design phase (i.g size of main habitat module).

Space Launch System (SLS)

The Space Launch System is the reference system used by NASA to put into both LEO and LLO satellites and exploration payloads during Artemis program; all information reported below are officially provided by NASA [13].
SLS comprises a family of 6 main rockets, divided into 3 different version: Block 1, Block 1B and Block 2. Each version represents a bigger and more powerful version of previous and it is able to transport larger payloads increasing also costs for single launch; as each version is an evolution of others, it will be possible to use it starting from different years (Block 1B starting from mid 20' and Block 2 starting the end of 20'). For each version it is available two different rockets one used only for cargo purposes and one rocket used also to transport humans (inside the space transfer system).

Whatever version it is considered, the basic set up of rocket includes:

- *Core stage*: it represents the main stage used in the first moment of launch to bring rocket from Earth surface up to high atmosphere. It includes all tanks of LOX and LH2, motors (RL 25 engines for Block 1) and all supporting systems for this stage.

- *Launch adapter/Interstage*: It represents the main component which connect the core stage with the fairing or exploration upper stage (for lastest version). Both first version it will contain the Interim Cryogenic Propulsion Stage which is connected with Orion capsule or with the Payload adapter.
- *Boosters*: Each version has two solid rocket boosters able to increase the power of rockets.
- *Integrated Spacecraft/Payload Element*: It includes the Payload adapter, the Payload and Fairing for cargos version of Block 1, whereas it incudes Orion capsule for Crew version.
  Version Block 1B and Block 2 include a new element called "Exploration upper stage" which uses RL 10 motors to bring payload and crew in LLO in a more efficient way. Also for Block 1B and 2 versions are present the Payload adapter, the payload and fairing.

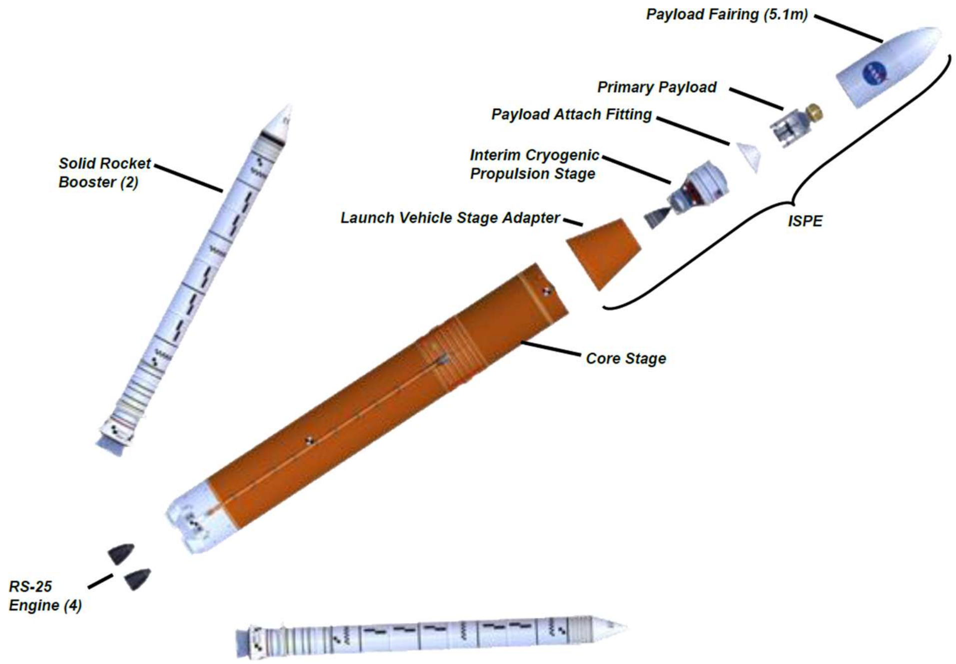


Fig. 2.3.3-1 Reference configuration of Block 1 rocket [13]

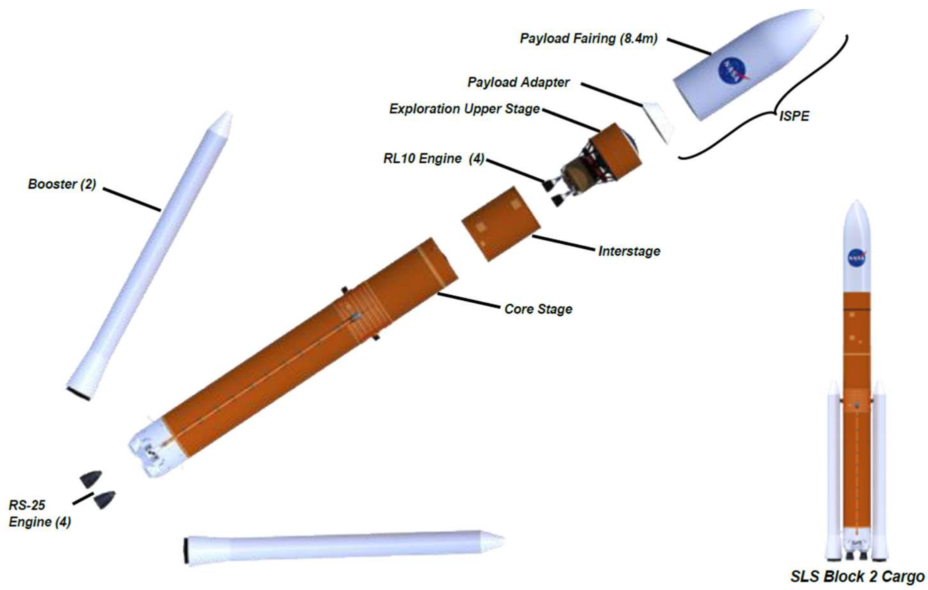


Fig 2.3.3-2 Reference configuration of Block 2 cargo version of rocket [13]

One of the most important feature of launch system is the list of requirements in terms of size and mass. While the mass is basically dependent of rocket configuration (thrust generated by engines) and final destination, size depends on fairing used for the mission; in fact, there are several different fairings which can adapted with the rocket and which allow to carry out different type of mission.
Major requirements for volume and mass of each rocket is reported on Fig. 2.3.3-2 provided by NASA [14] in figure below.

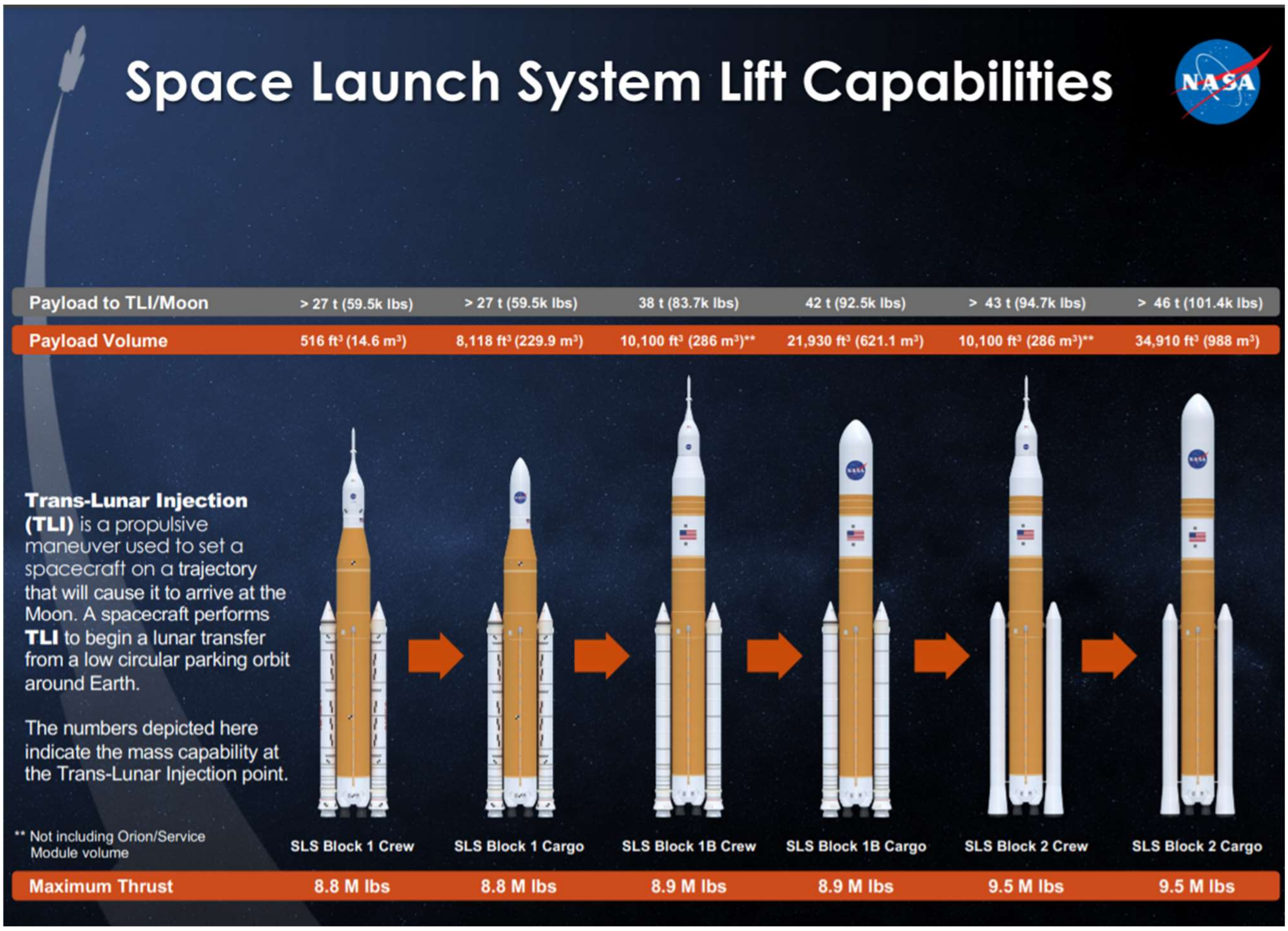
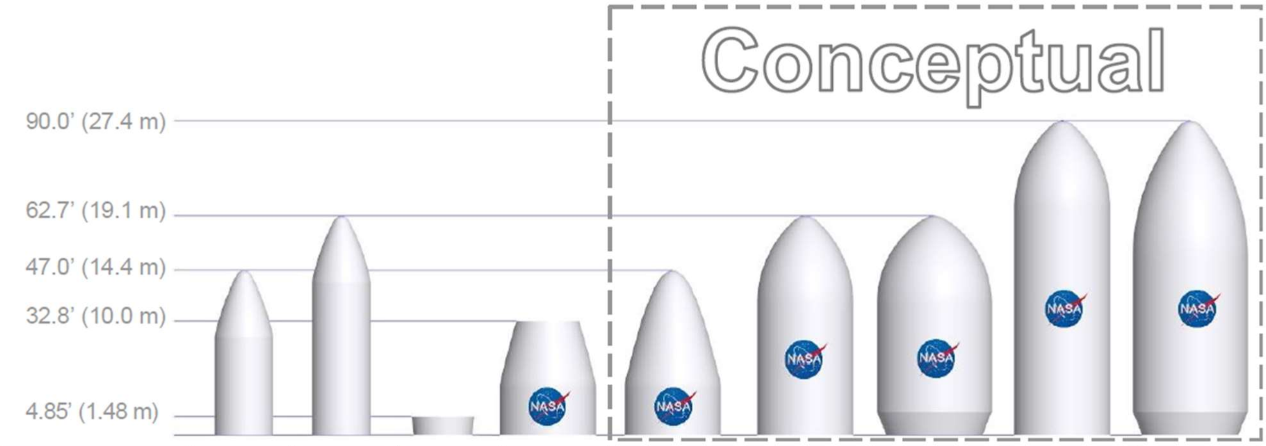


Fig 2.3.3-2 Resume of rockets capabilities (mass and volume) for each SLS version and configuration [14]

All different configurations of fairing are shown in figure 2.3.3-1.
It is important to point out it is not possible to adapt all fairings to all rockets, in particular for Block 1 it is possible to adapt only the first 2 versions (5.1m Short or Long version) and "10m" versions are available only for Block 2 version.
It is also important to enlighten the size shown in the picture are not the real size used to write requirements; in fact due to payload adapter (not considered in the data of pictures) both height and diameters are reduced and it must be taken into account. In addition, as the diameter of fairing decreased moving up top (due to fairing conic shape), it has been decided to consider only cylindric portion of fairing to host the payload it represents a conservative assumptions which decreased again the available height of fairing.



Enclosure	5.1m PLF	5.1m PLF	OSA	8.4m USA	8.4m USA PLF	8.4m PLF, Short	10m PLF Short	8.4m PLF, Long	10m PLF Long
Payload Type	5m PPL	5m PPL	5m SPL	8.4m CPL	8.4m PPL	8.4m PPL	10m PPL	8.4m PPL	10m PPL
Length	47.0 ft	62.7 ft	4.85 ft	32.8 ft	47.2 ft	62.7 ft	62.7 ft	90 ft	90 ft
	14.3 m	19.1 m	1.48 m	10.0 m	14.4 m	19.1 m	19.1 m	27.4 m	27.4 m
Diameter	16.7 ft	16.7 ft	17.7 ft	27.6 ft	27.6 ft	27.6 ft	33.0 ft	27.6 ft	33.0 ft
	5.1 m	5.1 m	5.4 m	8.4 m	8.4 m	8.4 m	10.0 m	8.4 m	10.0 m
Internal Diameter	15.1 ft	15.1 ft	16.7 ft	24.6 ft	24.6 ft	24.6 ft	29.9 ft	24.6 ft	29.9 ft
	4.6 m	4.6 m	5.1 m	7.5 m	7.5 m	7.5 m	9.1 m	7.5 m	9.1 m
Available Volume	5,358 ft^3	8,118 ft^3	516 ft^3	10,100 ft^3	11,260 ft^3	21,930 ft^3	32,470 ft^3	34,910 ft^3	46,610 ft^3
	151.7 m^3	229.9 m^3	14.6 m^3	286.0 m^3	319 m^3	621 m^3	919 m^3	988 m^3	1,320 m^3
Potential Availability *(No Earlier Than)*	COTS	COTS	2020	2024	2025	2025	2029	2029	2029

Fig. 2.3.3-3 Resume of possible fairing configurations for SLS [13]

In this study it will used only some of these configuration fairings to define size requirements for payloads. For Block 1 version it has been chosen the "5.1m PLF" version because it represents the biggest version available for this kind of rocket.

Parameters	Symbols	Values
Reference height	h_{ref} [m]	19.1
Maximum height	h_{max} [m]	NDA
Reference diameter	D_{ref} [m]	5.1
Internal diameter	D_i [m]	4.6

Table 2.3.3-1 Resume dimensions of 5.1m PLF long fairing

For Block 1B version it has been chosen either "8.4m short" or "8.4m long" version because their shape allow to maximize the available volume for payload.

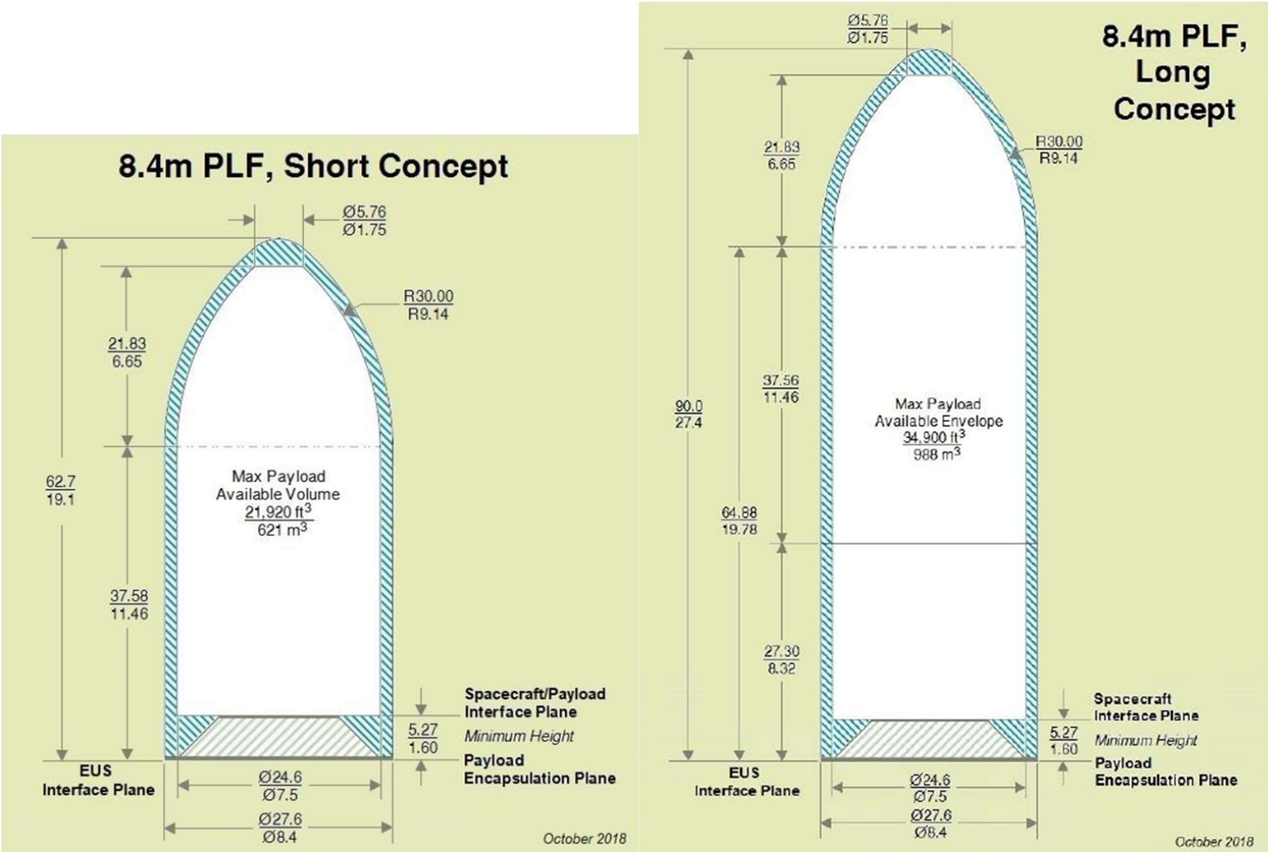


Fig. 2.3.3-4 8.4m PLF short and long concept [13]

Parameters	Symbols	Values
Reference height	h_{ref} [m]	19.1
Maximum height	h_{max} [m]	9.85
Reference diameter	D_{ref} [m]	8.4
Internal diameter	D_i [m]	7.5

Table 2.3.3-2 Resume dimensions of 8.4m PLF short fairing

Parameters	Symbols	Values
Reference height	h_{ref} [m]	27.4
Maximum height	h_{max} [m]	18.18
Reference diameter	D_{ref} [m]	8.4
Internal diameter	D_i [m]	7.5

Table 2.3.3-3 Resume dimensions of 8.4m PLF long fairing

In a similar way, for Block 2 version has been chosen the “10m” versions, also in this case to maximize the volume available for payload.

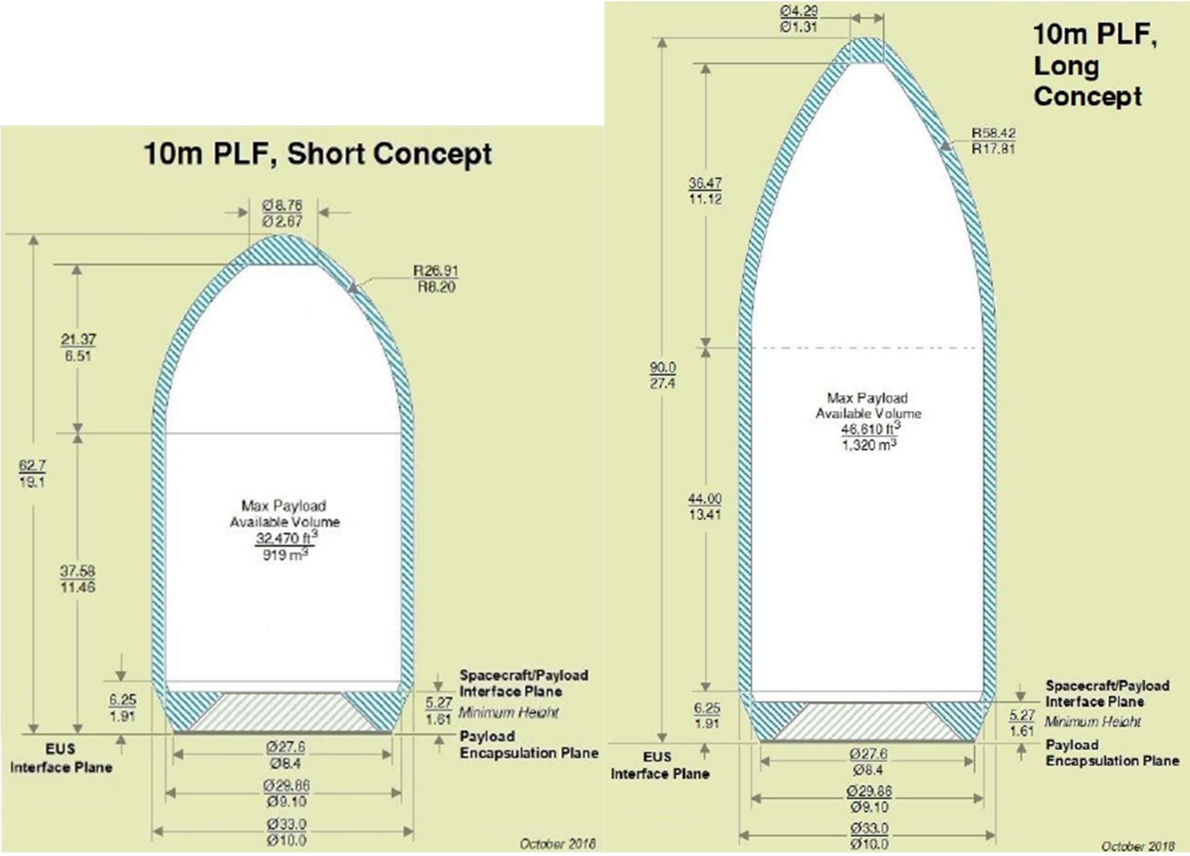


Fig 2.3.3-5 10m PLF short and long concept [13]

Parameters	Symbols	Values
Reference height	h_{ref} [m]	19.1
Maximum height	h_{max} [m]	9.85
Reference diameter	D_{ref} [m]	10
Internal diameter	D_i [m]	9.1

Table 2.3.3-3 Resume dimensions of 10m PLF short fairing

Parameters	Symbols	Values
Reference height	h_{ref} [m]	27.4
Maximum height	h_{max} [m]	11.8
Reference diameter	D_{ref} [m]	10
Internal diameter	D_i [m]	9.1

Table 2.3.3-4 Resume dimensions of 10m PLF long fairing

Super Heavy + Spaceship

The Super Heavy + Spaceship is a launch solution developed and in test phase by SpaceX and it is finalized to support future exploration mission both to the Moon and to Mars even transporting human crew.
One of the biggest feature of Spaceship solutions is the chance to reuse it for multiple mission getting the mission more sustainable both economically and ecologically.
As SLS, Spaceship solutions is able to bring up the Moon both cargo and human (there are two different version of spaceship), but differently from SLS, Spaceship is able to land on the Moon surface without needing a Landing and Ascending systems; this get the Space ship solution much more interesting of SLS. [15]
Spaceship solution uses a Super Heavy rocket as booster and Spaceship vehicle, powered by subcooled methane and oxygen, which transports payload.

Concerning requirements, Spaceship is able to transport up to 100 tonnes of payload up to Moon surface using the fairing of 8 meters of diameters and 8 of available height. In the figure below is shown the envelope of Spaceship fairing [15].

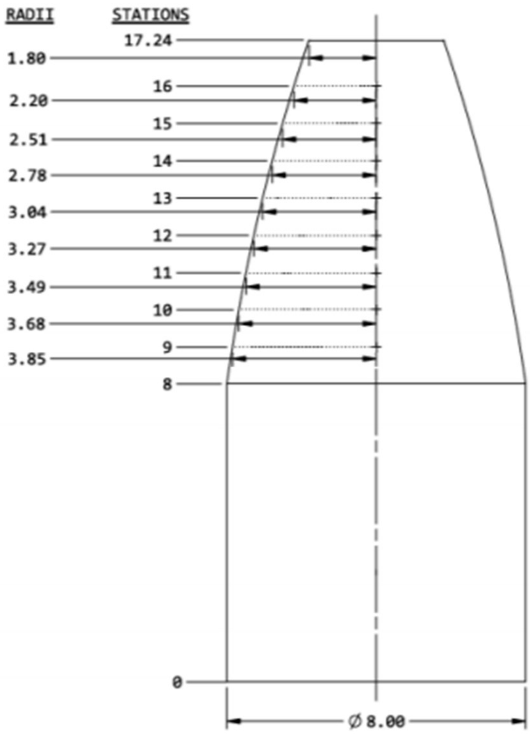


Fig. 2.3.3-6 Spaceship fairing envelope and dimensions [15]

It is important to point out there are in development other version capable to bring up to 22 m of height instead of 17.24m (reference height).

Parameters	Symbols	Values
Reference height	h_{ref} [m]	17.24
Maximum height	h_{max} [m]	8
Reference diameter	D_{ref} [m]	8
Internal diameter	D_i [m]	NDA

Table 2.3.3-5 Resume dimensions of Spaceship fairing

2.3.4 Communication Element

The Communication Architecture is a key element in design of Lunar bases due to high importance of communication and navigation on Moon surface. Nowadays, the major space agencies with support of big space companies have started preliminary studies to define the optimal communication architecture for future mission on the Moon; however, as these studies has been started recently, there are now much information available on literature. For this reason, in this sub-chapter will be introduced basic principia of major space agencies studies; these principia will represent the basis for a deeper analysis in following chapter of this study.
Currently the major projects to build a communication architecture to support navigation and communication functions in Moon missions are: LunaNet by NASA [3,16], LCNS by ESA [17] and communication architecture study from JAXA.

All these projects are agree on offering a complex and multilayer architecture; the multilayer architecture comprises: Earth Grounds Stations, On Earth orbit communication constellations, Lunar Gateway, On Lunar orbit communication constellations and Moon beacon/Moon Grounds Stations.
This architecture is really suited for different reasons:

- It allows real time communication between Moon-Earth and Moon-Moon users.

- It offers a great level of redundance; if an element of architecture fails It does not afflict others element and it is possible to readapt communication flow.

- It is flexible; that means it is possible to add new elements and layers in any moment depending on requests of users.

- It can be used by everyone; thanks to use of different communications bands, a wide range of satellite and users can use it.

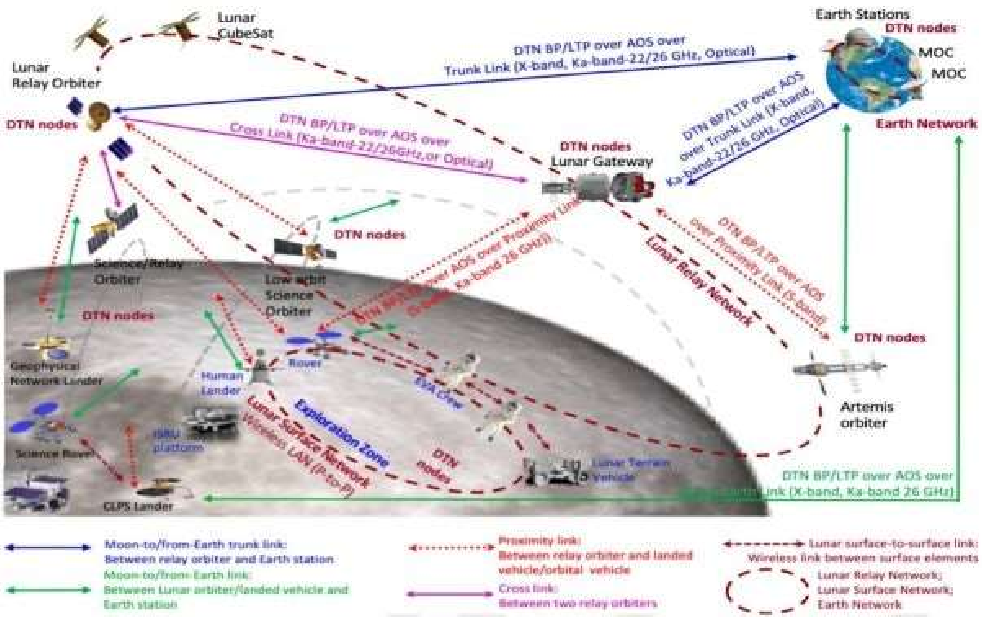


Fig 2.3.4 Communication Architecture by NASA (Credit: NASA)

Building the whole communication architecture will require several missions and, according to major space agencies roadmaps, it will be possible to have it about in the 30'.
As said, not specific information is currently available about the architecture for this reason in this study it will be considered a reduced version of multilayer architecture. It includes:

- Earth Ground stations network using Deep Space antennas
- Moon Relay constellation
- Moon Ground stations network represented by LavaTube beacon and South Pole beacon

A deeper analysis of these elements will be conducted in following chapter of this work.

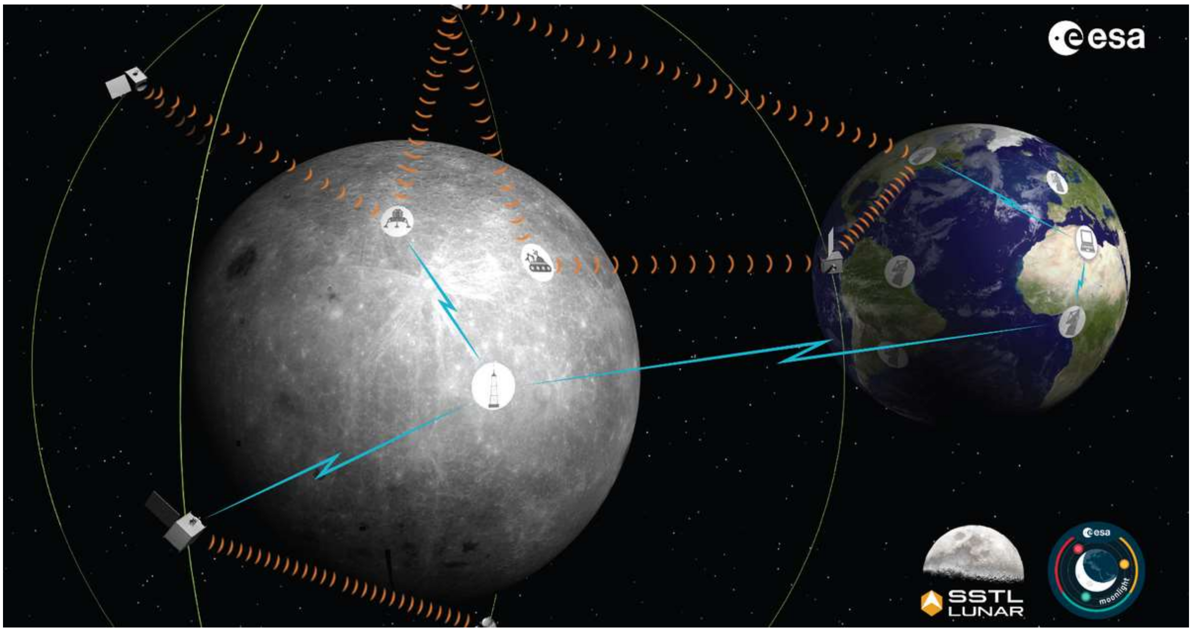


Fig. 2.3.4-2 Reference ESA Communication architecture used in this work (Credit by ESA)

2.3.5 Ground Element on the Moon

The ground element architecture on the Moon includes all systems needed to support on surface operations during exploration missions both pressurized and not pressurized.

The main system is certainly the habitable module which will host crews during the whole duration of mission and shall meet all needs of humans. Since the 90', it has been developed a huge number of papers which analysed the feasibility of habitat design and proposed some baseline to design it; however, large part of these publication is focused on design of habitat for Pole regions and there is not lot of projects focused on Lava Tubes.

Despite Lava Tubes and South Pole are different scenarios and design of habitat for these shall considered different assumptions and constraint, some common features can be taken from literature about South Pole habitat and adapted also for Lava Tube one. The main reference habitat used in this study is the ESA/MIT/SoM project [4], It is a feasibility study for a habitat on South Pole which will host up to 4 crew members for a length of 500 days. This project has been chosen because it represents a similar scenario in terms of duration and crew members capacity.

ESA/MIT/SoM habitat is a vertical cylindrical module composed by 4 floors; it is 15.5 meters tall and 10.5 meters in maximum diameter when deployed (8 meters when stowed). Structure is hybrid inflatable structure which allows to protect habitat from dangers of lunar environment (meteorites impact and rigid temperature); additional regolith and water layers are considered to protect from radiation. Airlock is also considered to allow EVAs. Concerning life support system, it considers a closed loop system which allow almost full recycle capabilities for atmosphere and water. Resupply consumables are sent from Earth as well as the greatest part of food needed (95%).

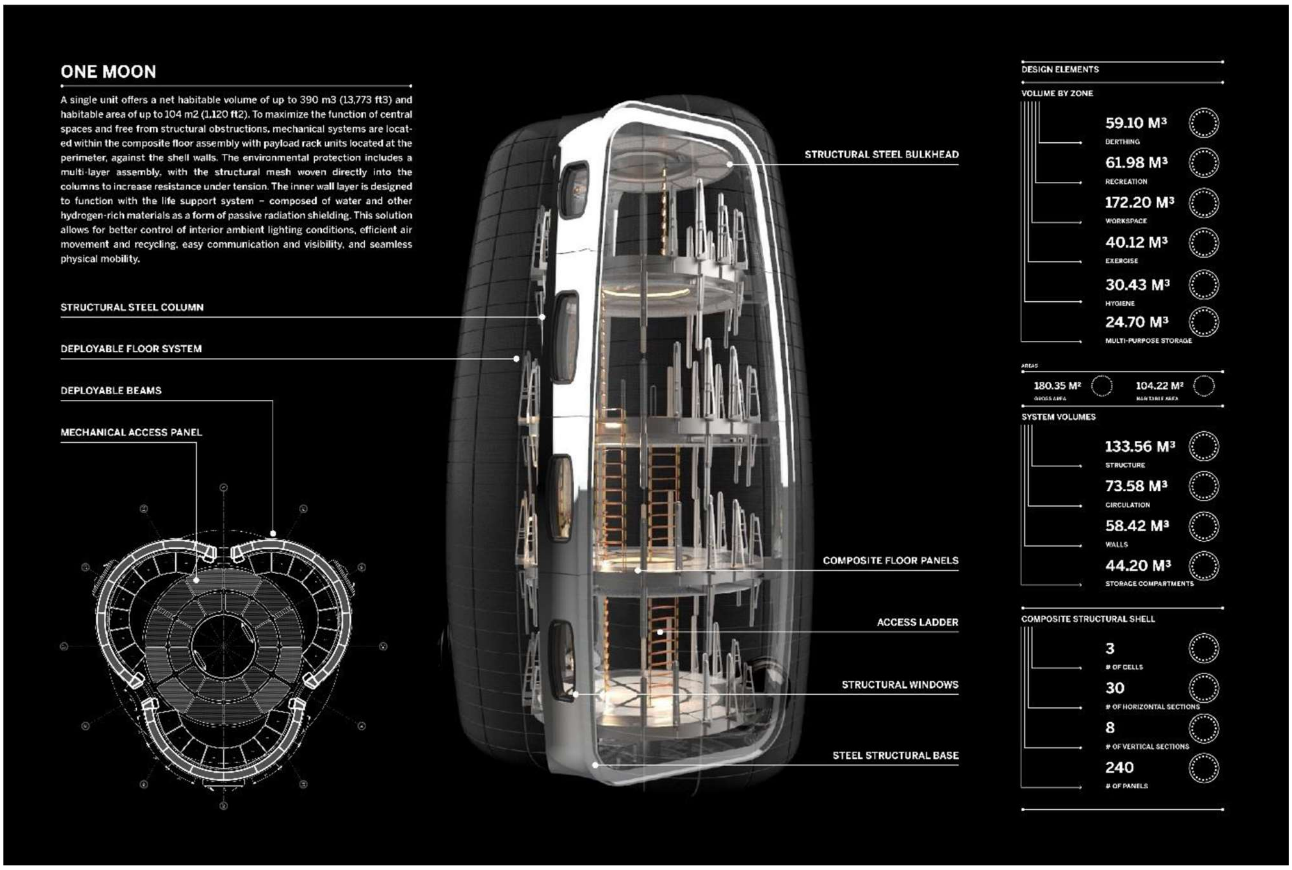


Fig 2.3.5-1 The ESA-MIT-SoM habitat proposal (Credit ESA) [4]

Pressurized rover is another key system to support human and robotic operation on Moon surface because allow humans to move for long distance on Moon surface and they can transport part of habitat and cargos from landing site to Moon camp. From a design point of view, pressurized rovers are very similar to habitat module as they should carry out the same functions as the habitat one; power generation, communication, accommodation, life support and radiation protection.

Different studies have already conduced preliminary analysis to design pressurized rovers. These projects are primary references for this study Lunar Cruiser by JAXA and Toyota [19] and Space Exploration Vehicle Concept by NASA [20].

Fig. 2.3.5-2 Lunar Cruiser by JAXA and Toyota (Credit Toyota.it)

Typical requirements for rover design are reported in table 2.3.5-1. Rovers will not been studied in this work but an overview about the main function is useful to understand similitudes with real habitat module.

ROV-001	The rover shall host from 2 up to 4 crew members
ROV-002	The rover shall support crew up for 14 days
ROV-003	The rover shall allow EVAs activities
ROV-004	The rover shall provide communication capabilities
ROV-005	The rover shall provide electrical handling capabilities
ROV-006	The rover shall provide guidance and navigation capabilities
ROV-007	The rover shall provide ECLS capabilities
ROV-008	The rover shall protect humans from radiation
ROV-009	The rover shall be able to transport cargos

Table 2.3.5-1 List of assumptions and requirements for Pressurized rovers

The last element to be studied is Airlock; it is fundamental to allow extravehicular activities both because it prepares astronauts to a different environment, in terms of pressure and temperature and because it allows cleaning procedures when astronauts come back, removing lunar dust present of spacesuit. Also for Airlocks the new trend it to use inflatable module which can be launched in stowed version and deployed in situ, this is done to reduce mass and volume of payloads.
According to state of arts [21,22,23], Airlock is composed by two part; the first rigid part is the equipment lock which is used to prepare astronauts for activities and where spacesuit are kept. The second part, the crew lock, is typically a small module, to reduce atmosphere losses during the doors opening, where cleaning activities are performed to avoid dust contamination inside the habitat.

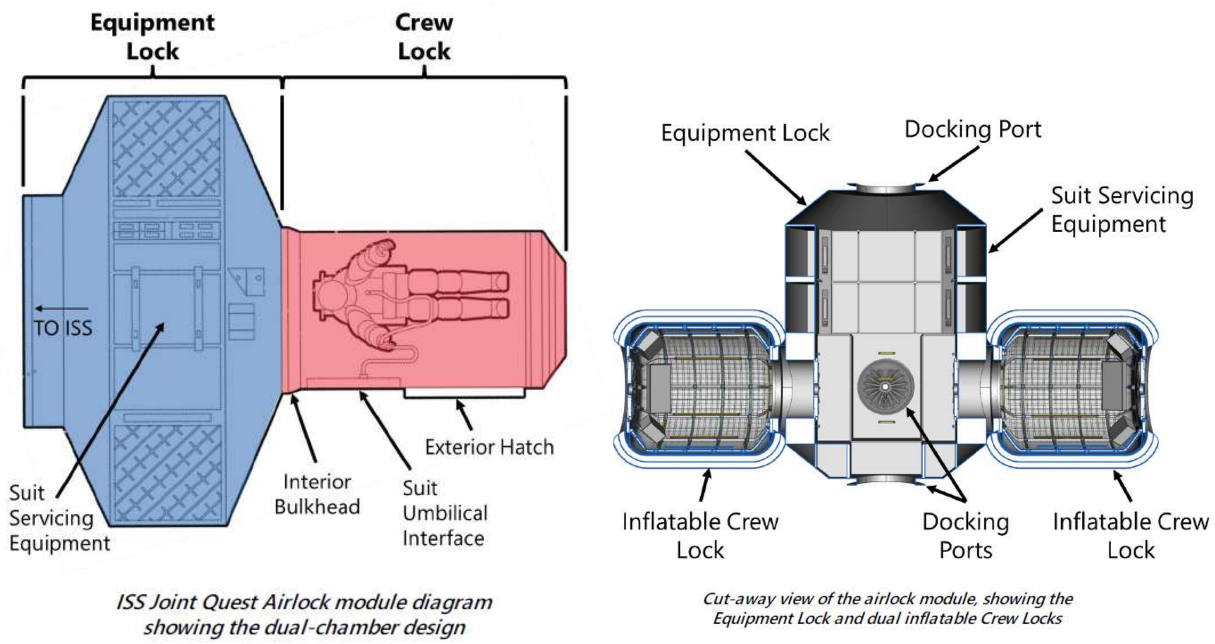


Fig . 2.3.5-3 and 2.3.5-4 Airlock of ISS (on the left) and Lunar Gateway (on the right) [22]

2.3.6 Ground Element on Earth

The ground element on Earth is basically represented by all mission control centres on Earth. For this study a detailed analysis of Ground element is not needed as it is possible to use the same centres already used for program Apollo, ARTEMIS and ISS.

2.4 High Level Requirements

In the following Table there are presented all high level requirements used as starting point to design the habitat. Some requirements come from assumptions made based on similar projects. Others come from the mission drivers for Moon exploration also these taken from similar projects.
Finally, other come from scenario analysis and they concern challenges and constraints of scenario.

ID	Description	Note
MIS-001	Lava tube lunar base shall support manned missions of maximum 4 people per crew	Reference crew capability for several Moon surface missions
MIS-002	Lava tube lunar base shall support human missions for a duration of maximum 180 days for one crew	Reference crew capability for several Moon surface missions
MIS-003	Lava tube lunar base shall support human mission for a total duration of 5 years	Reference habitat duration for similar lunar bases
MIS-004	Moon habitat shall be located inside Marius Hills Lava Tubes	Rationate from Chapter 2
MIS-005	Lava Tube lunar Base shall support human life and their functions for whole duration of mission including: communication, power generation and human life support	
MIS-006	Moon habitat shall support research activities on Moon surface including Extravehicular activities	Mission objectives
MIS-007	Habitat and other systems shall be launched from Earth using existing state-of-art launchers	Mission Architecture
MIS-008	On surface activities shall be carried out using state-of-art systems	Mission Architecture
MIS-009	Accessibility to Lava Tubes shall be considered as challenge into design	
MIS-010	Lunar Base shall be able to communicate with Earth in real time in any moment of mission	
MIS-011	Lunar Base shall be able to communicate with other Moon bases using relay satellite constellation	
MIS-012	Lunar Base shall be launched and built not before the 2030	

Table 2.4-1 High Level Requirements

In addition to these requirements, some other mission drivers have been defined:

- *Mass and Volume minimization*: Design shall reduce as much as possible masses and volumes of systems and consumables. This is done to reduce costs and complexity of both delivery missions and cargos. Reducing the number of launches and costs will get more sustainable the project.

- *Flexibility:* Design shall consider the possibility to expand habitat whit additional modules to grow the moon base. What is more, it should be possible to integrate inside the habitat new systems and modify the existing one. This is done to improve the reusability of system and get it more sustainable and reusable.

- *Modularity:* Design shall consider independent elements. This is done to allow separates launches of each element.

- *Possibility to refold habitat:* Habitat shall be designed to allow refold of module after the end of mission. This will get much easier the disposal activity getting possible to carry back to Earth the module or use is for other moon bases.
-

2.5 Conclusion

Mission overview is a key feature to design the whole habitat due to his influence in design choices.
In this chapter it has been introduced all aspects related how the missions work, defining the mission phases and scenario and the whole mission architecture, introducing all elements needed to operate the habitat. Finally, it has been presented mission requirements and drivers which will be the base to design the whole habitat.
Going into details, Mission architecture shows the need to design and build some elements which are currently not available. Communication architecture is still in study phase for a great part of space agencies and companies and so there are not much details about these elements. Together with communication, also navigation is still an open point which has not been developed yet. Human Landing system is another key element needed to allow access on Moon surface; also in this case there are only studied and not still prototypes (except Spaceship). Other big challenges concerns accessibility inside Lava Tube; currently there is not available information about Lava Tube and their access points; however, according to the few data gathered by SELENE, skylights are high and often partially covered by fallen rocks getting very hard to get access inside and to transport payloads.
All these aspects could be studied in future works which will be focused on particular elements of mission.

3. Functional Analysis

Functional Analysis represents the first step to identify all functions that shall be performed by habitat system. Identification of functions and sub-functions is important first to build how systems are made and how they work, secondary it is important to identify more specific requirements for each system. Based on functional analysis, in fact, it is possible to build the system architecture where functions are substituted by specific equipment which perform the identified function.
In this chapter it will be carried out the functional analysis which has the purpose to identify main functions of habitat and propose a baseline systems architecture.
First, it will be built the Functional Tree to identify the high level functions and sub functions. After that, Matrix Functions/Equipment will be built to identify the required equipment and devices needed to carry out all functions; using F/E matrix and functional tree, Product tree will be built.
Finally, it will be built the N2 matrix which shows the interface between systems.

3.1 Functional Tree

Functional Tree represents logical decomposition of functions from a top level (system) up to bottom level (units or components). In this work it has been decided to focus on only habitat leaving aside other functions as launch, transportation, disposal etc…Functional decomposition is fundamental to identify the system architecture and to design specifically each system.
Functional Tree in Fig. 3.1-1 shows the top level function breakdown up to the third level of decomposition; orange functions represents system function while the yellow represent the sub system functions.
In this analysis, for some systems, it has been developed a specific functional in the following chapters.

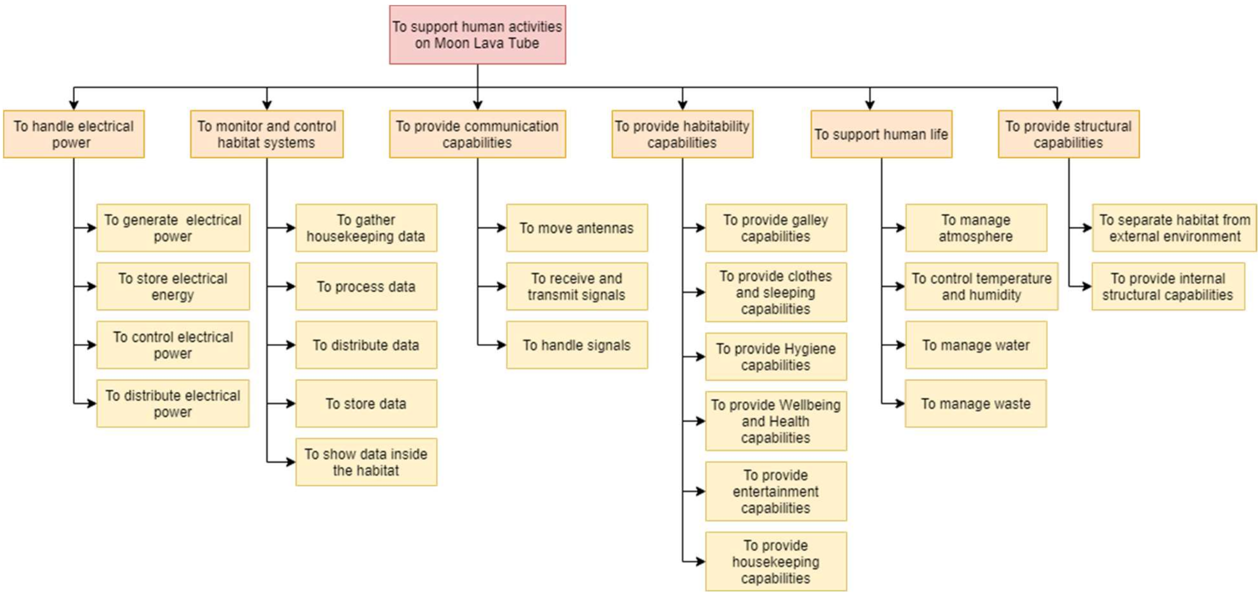


Fig 3.1-1 Functional Tree

3.2 F/E Matrix

Functions/Equipment Matrix allows to identify which equipment performs each identified function. In this way it is possible to pass from a functional point of view to an equipment point of view and to allocate functions to each equipment.

		Electrical Power System			Command & Data Handling					Communication system			Accomodation						Environmental Control and Life Support System				Structure	
		Power Generation Sub-system	Power Storage Sub-system	Power Management and Distribution Sub-system	Sensoring Sub-system	Data Processing Sub-system	Data Distribution Sub-system	Data Storage Sub-system	Data display Sub-system	Tracking Sub-System	Antennas Sub-system	RF handling Sub-system	Galley and Food quartier	Clothes and Sleeping quartier	Hygiene Sub-system	Wellbeing and Health Sub-system	Entertainment Sub-system	Housekeeping Sub-system	Atmosphere management Sub-system	Environmental thermal control Sub-system	Water management Sub-system	Waste management Sub-system	External Shell	Internal structure
To handle electrical power	To generate electrical power	X																						
	To store electrical energy		X																					
	To control electrical power			X																				
	To distribute electrical power			X																				
To monitor and control habitat systems	To gather housekeeping data				X																			
	To process data					X																		
	To distribute data						X																	
	To store data							X																
	To show data inside the habitat								X															
To provide communication capabilities	To move antennas									X														
	To receive and transmit signals										X													
	To handle signals											X												
To provide habitability capabilities	To provide galley capabilities												X											
	To provide clothes and sleeping capabilities													X										
	To provide Hygiene capabilities														X									
	To provide wellbeing and health capabilities															X								
	To provide entertainment capabilities																X							
	To provide housekeeping capabilities																	X						
To support human life	To manage atmosphere																		X					
	To control temperature and humidity																			X				
	To manage water																				X			
	To manage waste																					X		
To provide structural capabilities	To separate habitat from external environment																						X	
	To provide internal structural capabilities																							X

Table 3.2-1 F/E Matrix

3.3 Product Tree

Based on Functional Tree and F/E Matrix, it has been built the Product Tree which represents the Equipment composition from the top level up to subsystem level.

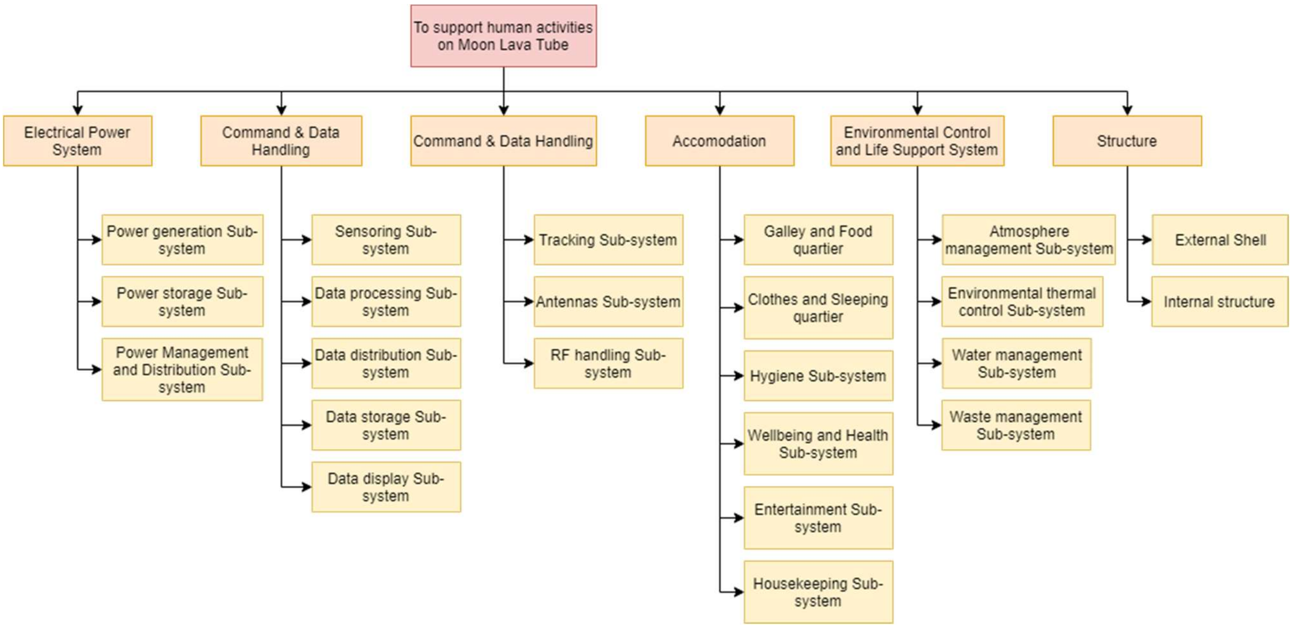


Fig. 3.3-1 Product Tree

Electrical power System: It is the system which handles the electrical power. First electrical power shall be generated using one of the available technologies, then it must be regulated to provide the right voltage to all loads and so distributed to all users. In addition, electrical power systems shall include addition protection system to avoid overloads on distribution bus. Power regulation and distribution are often functions performed by the same sub-system because regulation devices are included into the distribution network and they also manage the power allocation.

Command & Data Handling: It is the system which control and monitor in every moment the state of habitat and other systems and providing information both to crew, by means of a dedicated display sub-system, and to mission control, interfacing with communication system and sending the housekeeping data autonomously. As in space missions the new trend is to get everything autonomously and in order to reduce the workloads of crew, C&DH shall implement an artificial intelligence which can take decisions and command other systems without the participation of humans. Finally, it is necessary to have a storage sub-system which can save all data gathered during the mission, both housekeeping and scientific.

Communication System: It is the system which manage incoming and outgoing communication signals. Communication system, in this context, means only the external plant located outside the Lava Tube and all additional equipment which bring signals from the external communication plant up to habitat; it does not consider the relay communication satellite as part of it.
Communication system comprises equipment needed to move antennas to track satellites in low lunar orbit, equipment needed to modulate/demodulate and amplify signals and the antennas itself which generate/receive signals.

Accommodation: It includes all equipment needed to allow habitability capabilities.

Environmental and Life Support System: It includes all equipment needed to support life of crew. Life support include the management of atmosphere to provide the correct pressure and composition of air as well as the right temperature and humidity condition. Also water management is a function of life support as water is strongly needed for life and hygiene. In addition to that, also waste management is part of life support, as part of waste shall be recycled to obtain water and other useful biomass.

Structure: It includes all structure needed to protect humans from lunar environment and to sustain habitat. In particular, external shell is needed to contain habitat atmosphere and to protect crew from external threats (e.g moon dust, rocks impacts, low temperatures). Internal structure has the aim to give the shape to habitat and to separate internal spaces as well as to sustain vertical loads.

3.4 N2 Matrix

N2 is built to identify interfaces between systems and which kind of inputs/outputs each system provides to other ones [REFERENCE NASA SYSTEM HANDBOOK]. Representation of N2 matrix is shown in Fig. 3.4-1 where the interfaces are highlighted by circles with indicated the type of interface. Arrows show direction of interface, if it is an input (towards the block) or output (outgoing from block).
Interfaces are:

- **E: Electrical (Red circle):** It means electrical power is provided.
- **D: Data (Yellow circle):** It means data is provided. Data can be both commands and sensors data.
- **M: Mechanical (Blu circle):** It means there is a physical interface between to items.
- **I: General Interface:** It means there is occasional interface, usually between system and humans. It happens for example to repair the system or to interact with it.
- **SS: Supplied Services:** It means interface concerns other type of entities provided by systems. These entities comprise: air, water, food, entertainment, etc…

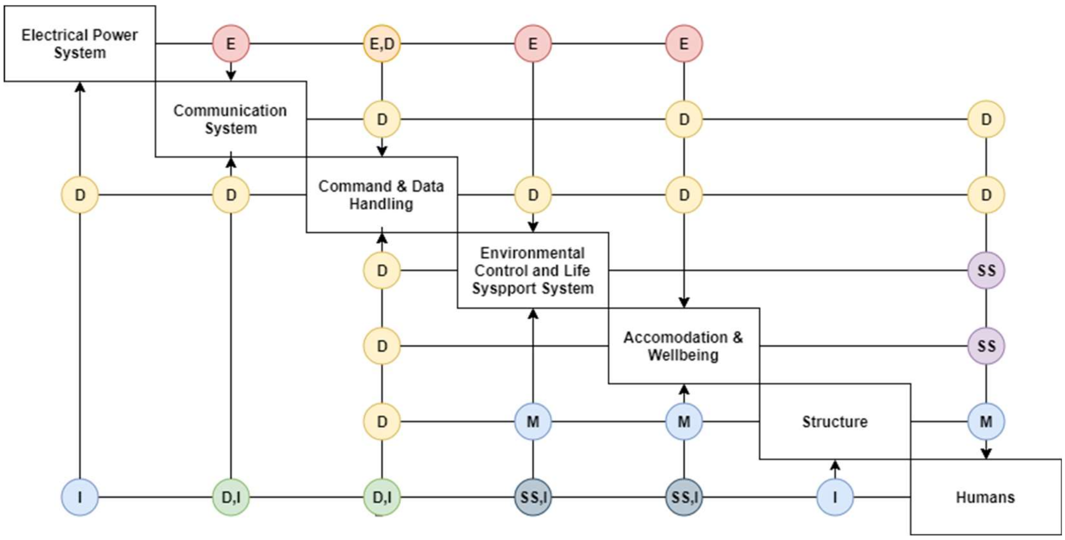


Fig. 3.4-1 N2 matrix

3.5 Conclusion

In this chapter it has been performed the functional analysis with the aim to identify the main functions to be performed up to sub-system level. In addition, it has been defined the main systems and their subsystems which shall be detailed in future works. Systems interface have been defined by means the N2 matrix.

Future work will be keeping on decomposition up to component level; in addition, it will be necessary to build the flow diagrams for both function and equipment defining a more detailed system architecture. Finally it will be necessary to detail interconnection between systems and define states of each system.

4. Structure

Structures for space application play a fundamental role into the mission design as they influence several aspects of mission as: human safety (against radiation and meteorites impacts), logistics and missions costs, design of others systems (e.g thermal system design),etc…The biggest gaps concerns the need to design a particular structure layout which is able to meet all human related requirements, in a Moon environment, keeping low the mass and volume
Over the years several solutions have been identified to optimize the design of effective structures which could solve this gap. Particular effective solutions have been identified on inflatable structures, which are particular multilayer structure that are launched form Earth and deployed once they arrive in deployment site. TransHab [24] and BEAM [25] projects are two of the most important studies of inflatable structure for space application, in particular both projects have been designed to be implemented on ISS. Concerning space habitat structure, most of studies focus on lunar habitat on south Pole and there are not studies conducted for habitat inside lava tube.
This study has the aim to develop a preliminary structure design considering as reference the Lava Tube scenario.
In this chapter first it will be conducted an analysis of habitat volume to estimate a starting value for following design. Then it will be proposed an initial baseline for habitat taking into account the launch requirements imposed by launchers and fairing. After that, it will be given a more detailed description of parts which compose structure. Following, inflatable shell will be analysed to identify material and specification for each layer. At the end, an additional short description of airlock will be provided.

4.1 Volume Estimation

The total volume considered in this study is composed by four different sub-volumes: habitable volume, systems volume, accommodation volume and structure volume (both primary and secondary). The sum of habitable volume and systems volume represents the total pressurized volume inside the habitat when it is deployed, while the sum of four represents the maximum size of module.

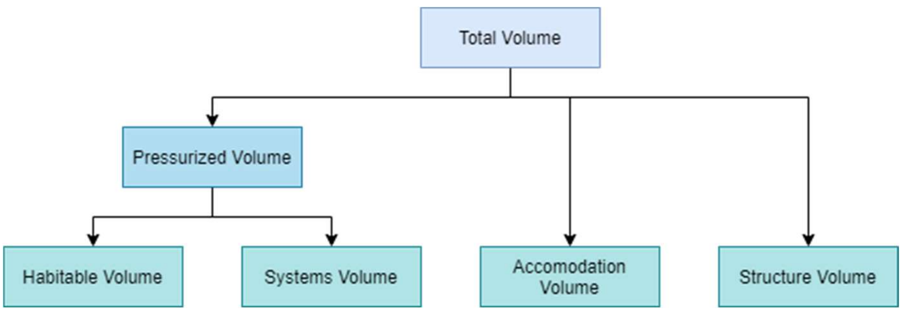


Fig 4.1-1 Volume Decomposition

Habitable Volume represents spaces which humans can use to live inside habitat free from devices and systems. In all past manned missions and on future works, the first selection of habitable volume depends on crew size and mission duration. That is made because according to psychological studies, the more an astronaut stays inside an isolated habitat the more habitable volume is needed to avoid stress and

uneasiness. As each astronaut need 'his own volume', the total habitable volume grows proportionally with number of crew members.
To select the habitable volume needed by each astronaut, Marianne Radisil et al. [26] analysed all past manned missions to get a mathematical law that return habitable volume per crew members over mission duration. Note that in Radisil et al. study, total pressured volume is considered as total Habitable one.

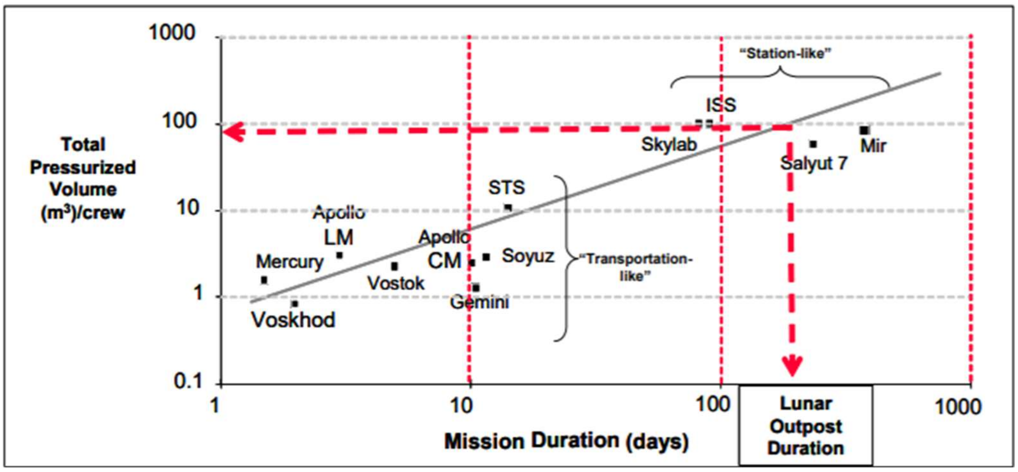


Fig 4.1-2 Mathematical law for habitable volume by M. Radisil et al. [26]

In this study it has been decided to use the value of 100 m^3/CM which is in accordance with currently projects (ISS) and new concept by ESA.
The total habitable volume for whole crew is so:

$$Habitable\ Volume = 100\frac{m^3}{CM} * 4\ CM = 400\ m^3$$

Accommodation value represents the total volume occupied by all systems and accommodation devices; it does not include ECLS System devices. The reference value taken has been obtained from the accommodation analysis carried out in the following chapters of this work (Chapter 5).

$$Accomodation\ Volume\ = 73\ m^3$$

Estimation of Systems volume, which includes all ECLS System devices, is not easy. In fact, currently data to carry out a good estimation are not available on bibliography. For this reason, in this work it has been decided to estimate the system volume as the 10% of habitable volume. This approximation has been made taking into account that systems volume depends on number of crew members; in fact, the more astronauts live on habitat the higher it will be the request of oxygen and water and so the higher will be the mass and volume of water and atmosphere processing units.
It is important to point out that this approximation is totally arbitrary and it will be necessary in following steps of design to identify more realistic value.

$$Systems\ Volume\ \approx 10\%\ of\ habitable\ volume = 40\ m^3$$

The total pressurized volume is so:

$$Total\ Pressurized\ Volume = 400 + 40 = 400\ m^3$$

Structure volume consists of primary structure and secondary structure values both depending on height and maximum diameter of module. Due the fact that in this step of analysis both height and diameter are not known, also for structure volume it has been decided to assume an arbitrary value. The value chosen is the 15% of the total volume free from structure itself.

$$Structure\ Volume = 10\% * (habitable + accomodation + systems\ volume) \approx 51.3\ m^3$$

Sum of the four sub-volumes above represents the total internal volume when habitat is not totally deployed.

$$Total\ internal\ volume = 400 + 73 + 40 + 51.3 = 564.3\ m^3$$

This value represents only a first estimation of volume as in this phase the structure and size have not been decided.
Once at least the secondary structure have been defined, and so it is possible to estimate analytically the volume of structure, it is possible to implement an iterative process which converges to the exact value of internal volume defining as well the reference maximum internal diameter, which represents the reference diameter to define different configuration (See next section).
In Appendix A is explained how iterative process is implemented, while in table below are resumed the results of process.

Parameters	Symbols	Values
Height	h [m]	12.8
Maximum Reference Internal Diameter	$D_{i,REF}$ [m]	7.52
Total internal volume	V_i [m^3]	568.92

Table 4.1-1 Results of structure iterative process

4.2 General Overview

The main habitat module selected for this study is a vertical cylindrical body shielded by an inflatable structure. The choice has been made taking into account several aspects. First it has been decided to use an inflatable shell because it was wanted to decrease weight and volume of structure during launch and transportation to deployment site; this choice is more and more selected in several space manned projects as TransHab and BEAM which are references projects for this study.

Fig. 4.2-1 TransHab (credit to Wikipedia.org) **Fig. 4.2-2 BEAM inflatable module (credit to Wikipedia.org)**

The second aspect took into account into this design is about the shape. According to state of art, the most used shapes for habitable module on Moon are spherical, torus and cylindrical shape [27]. These three shapes are widely used in projects because they do not require complex and heavy additional structure to be kept on. Considering the ratio between weight and load to be sustained, sphere is the best shape followed by torus and finally cylinder. However, sphere and torus are the worst choice in an habitable space point of view due to double curvature which decrease the usable volume of habitat; cylinder does not have this problem because of only curvature in one dimension. Because of need to use as possible whole the space available and as the difference of loads between different configuration is not very high, it has decided to select the cylindrical shape.

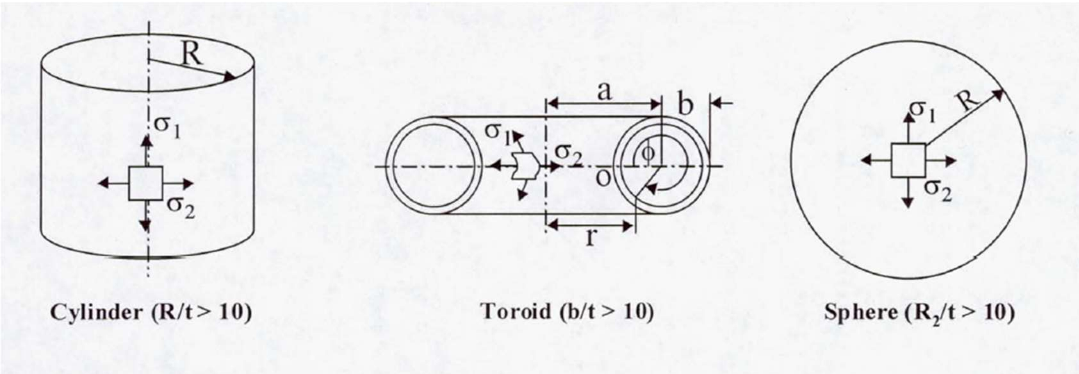


Fig. 4.2-3 Possible habitat shapes. [27]

Finally, it has been decided to develop habitat in vertical direction to keep base area low maintaining fix the volume chosen (see next chapter to discover how volume has been assumed); having a base area low allow to make possible the launch of whole module in a single time respecting the launcher requirements. In addition to that it is necessary to point out that several other projects consider vertical development of module as the best choice in Moon environment because due to lower gravity, access to high zone of habitat is easier and it is possible to use efficiently whole space. Figures below show the final set up of habitat module; where: green external shell, blue composite floors and red rigid structure.

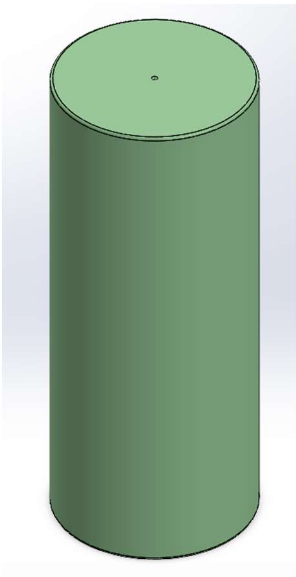


Fig. 4.2-4 Habitat module

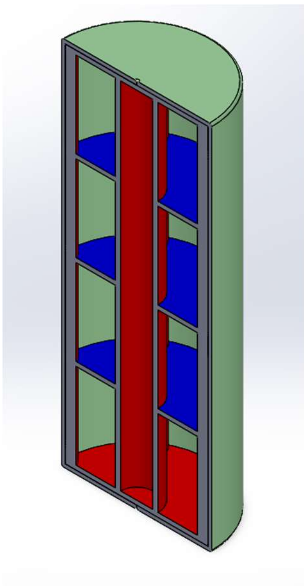


Fig.4.2-5 Habitat module section view

4.3 Configurations

Habitat design starts from the launcher requirements which impose strict constraints to habitat size.
In table 2 It is reported a short resume of launch options.

	Maximum Heigh [m]	Maximum Diameter [m]
SLS BLOCK 1	19.1 (TBC)	4.6
SLS BLOCK 1B – FAIRING 1	9.85	7.5
SLS BLOCK 1B – FAIRING 2	18.18	7.5
SLS BLOCK 2 – FAIRING 1	9.85	9.1
SLS BLOCK 2 – FAIRING 2	11.8	9.1
SPACEX - STARSHIP	8	8

Table 4.3-1 Resume of reference launch options requirements for this work

From a preliminary feasibility analysis, conducted in Appendix B, only three launch options could host the habitat: SLS Block 1B-Fearing 2 and both fairings of SLS Block 2.
Tanking these options as reference, the following four configuration have been considered. It is important to point out that these are only four possible configurations, chosen arbitrary. Other hybrid configurations could be also chosen in conformance with launchers requirements.

	Option 1	Option 2	Option 3	Option 4
Floors	3	3	2.5	4
Maximum Height [m]	12.8	11.8	11.3	17
Floor Height [m]	4	3.6	4-4-2.5	4
Maximum Diameter [m]	6.5 (Undeployed)	7.84 (Deployed)	8.02 (Deployed)	6.56 (Deployed)
Habitable Area [m^2]	133.34	145	101.12	134.09
Launcher Available	Block 1B (vrs2)	Block 2B (vrs 2)	SLS BLOCK 2 – vrs 2	SLS BLOCK 1B - vrs 2

Table 4.3-2 Resume of habitat options

Option 1 considers a floor stowed configuration; flooring will be deployed when habitat will arrive at deployment site. This option is a modification of a basic 3 floors configuration, where the internal diameter (7.52 meters) would not have allowed the transportation with any launchers. To estimate the new maximum internal diameter, it has been assumed that flooring can be "close" up to a closing angle of 60°.
This configuration allow to have a more compact structure during launch which bring to both save space and increase the resistance due to load in launch phase; however, the need of deployment mechanisms for flooring get this solution more complex than others and for this reason, in this study option 1 is not considered the best solution.
Option 2 is a modification of option 1 where the height has been reduced to fit it inside the Block 2- Fairing 2 launcher. This option allows to have all features of first model without change a lot the basic idea, in fact the number of floors remains the same and others size as well. The big problem of this solution is the floor height. Due to the low gravity on Moon, astronauts on Moon usually walk on surface 'jumping' and reaching usually higher distances from floor; for this reason, usually the high of floors is bigger than on Earth. According to some NASA study on Space Architecture [28], on Moon it should be considered as minimum floor height 4 meters. Despite that, many studies have proposed solutions using floor heights lower than 4 meters, which represents a suggested value but not a constrain. On basis of these consideration, in this study it has been decided to consider configuration using floor heights lower than 4 meters but taking into account that these configurations are not the best options. Finally, it must be said that the total height of option 2 configuration is exactly the limit for SLS Block 2, that means module can be launched with this rocket but reducing the total height is strongly suggested for safety reasons.

To overcome the problem of floor height it has been decided to change option 2, reducing the number habitable floors from 3 to 2. In this way, the two habitable floors have height of 4 meters in conformance with NASA suggestion. In addition to habitable floors, option 3 includes an additional floor having height of 2.5 meters; this floor, due to the very low height, is not used for living activities but it can be fully used by supporting systems and tanks saving a large volume on habitable floors. In conclusion, it must be said that the total height of module is reduced solving another problem of option 2 (integration on SLS Block 2 rocket). The major disadvantage of option 3 solution is the reduced habitable area which pass from an average of 135 m^2 of other options to 101 m^2; however, this do not represent a huge problem as this value represents a sufficient area for a 6 months staying on moon habitat.
The last configuration option takes advantage of huge SLS Block 1B fairing height to develop a 4 floors solution which maintains the same habitable area while it reduces a lot the maximum diameter of module making possible to launch it using SLS Block 1B rocket.
All options above are valid solutions, each one with advantages and disadvantages; however, the main aim of this preliminary work is not to select the best option, study which will be carried out in a second stage of design, but a valid one which can be reference for the rest of work. In this study it has been decided to use the option 4 configuration because this does not present huge challenges to be face up in following design phases and so it represents one of the most valid solution.

The final features of habitat module are listed in following Table.

Parameters	Symbols	Values
Maximum Height	h [m]	17
Maximum Internal diameter	D_i [m]	6.56
Maximum External diameter (when deployed)	D_e [m]	6.91
Total Area (when deployed)	A_i [m^2]	135.01
Habitable volume	V_{hab} [m^3]	400
Total internal volume	V_{tot} [m^3]	574.15

Table 4.3-3 List of final features of habitat module

4.4 Structure

The habitat module structure is composed by two main structural elements: the external shell which is a deployable element and the internal sustain structure which is a cylindrical rigid element set in the middle of module. This configuration is very similar to TransHab one and it is preferable because his simplicity and reduced additional structural elements.

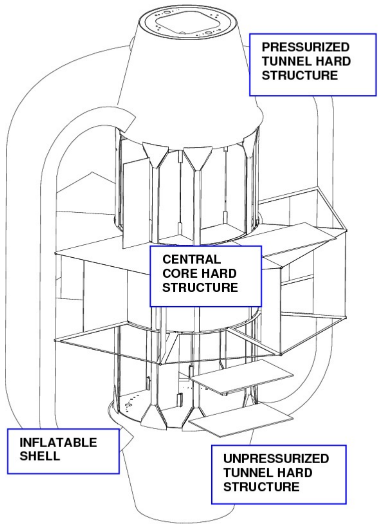


Fig. 4.4-1 Detail of TransHab structural elements [27]

4.4.1 Central Core

The central core, which is the secondary structure element, has the function to sustain the inflatable shell and to host part of systems. What is more, this element has also the function to floors access to astronauts by means an internal staircase.

The secondary structure is composed by a central circular element, hollow inside to allow astronaut get access to all floors, and 6 longerons arranged on the outer circumference. In addition, it has been added elements which connect the longerons and the central cylinder; these elements are thin layers arranged below flooring and they are used to allow loads to distribute on both main structural elements.

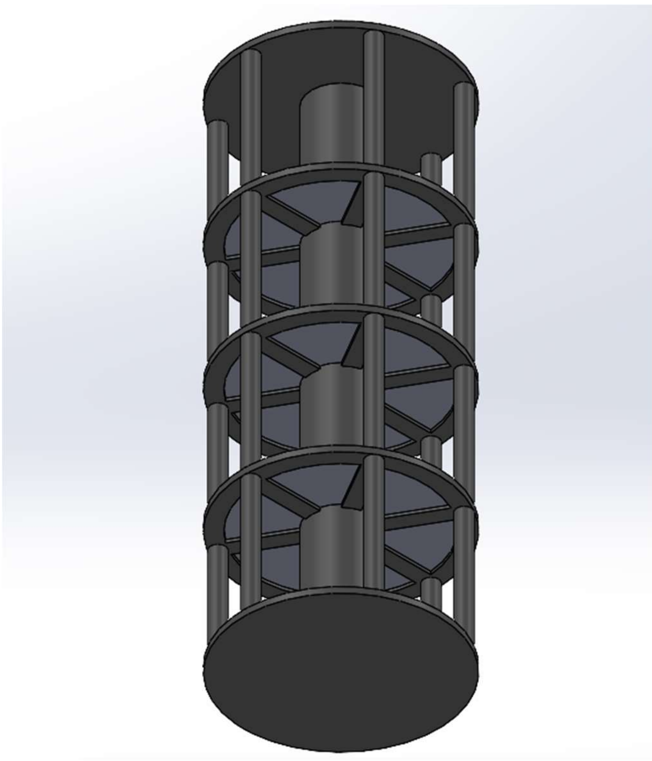


Fig 4.4.1-1 Secondary Structure of habitable module

Flooring is composed by two parts, a thin metallic layer which connect longerons and central element and a second thin layer of composite material where astronauts can walk on. The choice of composite material has been made to reduce weight of structure and give it a sufficient stiffness to sustain vertical loads.
It is necessary to point out that, as in this step of design it has not been carried out structural analysis, some values and choices has been chosen arbitrary taking as reference the few data available on bibliography. In future it will be necessary to carry out these analysis and re design the overall structure to optimize the weight, costs and volumes ensuring a good level of safety and structural reliability.

4.4.2 Deployable Shell

The deployable shell represents one of the most important element of structure system because it shall protect astronauts from a lot of hazards of Moon environment. In particular the design shall take into account both the danger of rock falls and the extreme temperature inside the Lava Tube as well as the pressurized loads. It is important to point out that the Lava Tube scenario represents one of the most convenient scenarios on Moon surface from structural point of view; in fact, thanks to regolith cave roof, habitat inside the Lava Tube is shield from both radiation and meteors impacts that simplifies the shell design reducing the thickness of protection layers. In addition to that, the stable temperature inside the cave reduces the temperature gradient between inside and outside habitat decreasing the temperature loads and strain of structure reducing also fatigues effects.
In addition to main requirements, it is necessary to consider also other human safety and assembly requirements to select the best materials for the shell; the list of all high level requirements for secondary structure is shown in Table 4.4.2-1.

ID	Description	Note
STR-PRI-001	Primary structure shall protect habitat from rock slides	
STR-PRI-002	Primary structure shall protect habitat from Moon low temperature	
STR-PRI-003	Primary structure shall withstand pressurization loads of habitat	
STR-PRI-004	Primary structure shall be deployable	
STR-PRI-005	The inner layer of primary structure shall not be toxic for human	To reduce health hazards for humans
STR-PRI-006	The inner layer of primary structure shall resist puncture	
STR-PRI-007	The inner layer of primary structure shall resist flames	To withstand in case of fires
STR-PRI-008	The inner layer of primary structure shall have good sound suppression properties	To reduce sound disturbance inside habitat
STR-PRI-009	The Bladder layers shall reduce as possible air leakage	To reduce resupply of air
STR-PRI-010	The Bladder layers shall block as much as possible moisture	Moisture could damage shell layers (i.g Kevlar)
STR-PRI-011	The Bladder layers shall protect shell structure from oxygen	Free oxygen could damage shell layers
STR-PRI-012	The Bladder layers shall resist flames	To withstand in case of fires

STR-PRI-013	The Bladder layers shall resist very low temperature	To avoid creep and damage during launch and transportation
STR-PRI-014	The Restraint layer shall resist very low temperature	To avoid creep and damage during launch and transportation
STR-PRI-015	The Rock slides protection layer shall resist very low temperature	To avoid creep and damage during launch and transportation
STR-PRI-016	The External layer of primary structure shall withstand to outgassing phenomena	To avoid increase duration of shell life
STR-PRI-017	The External layer shall protect habitat from Moon dust	
STR-PRI-018	The Deployable system layer shall contain inflatable structure during all transportation phases	
STR-PRI-019	The safety factor used in shell design shall be 4	From *"Airship Design Criteria, FAA-P-81 00-2)"* applied for TransHab

Table 4.4.2-1 High level requirements for Primary Structure

The shell consists of six main units each of which is studied to meet one or more requirements; units are composed by one or more thin layers. The choice of materials, number of layers and thickness has been made taking into account similar existing projects [29,30]. Primary structure layers are:

- *External layer*: It represents the closest layer to the lunar environment. The main aim of this layer is to protect the shell from lunar dust which could damage the innermost layers of shell and provide a first resistance layer form mechanical loads.
  Following the example of TransHab, it has been decided to use Nexter AF-62 layer; this is a ceramic material which provide a good mechanical resistance and, thanks to his ceramic nature, has a weak magnetic attraction effect against charged particles of Moon dust.

- *Multi-Layers Insulation (MLI):* It represents the thermal protection unit to reduce the heat exchange between habitat and Moon environment; reducing heat exchanges would decrease the power required from heaters to keep a comfortable temperature inside the module.
  This unit is composed by several thin layers of Mylar and Kapton, this solution has been used for several space application and represents the best solution to insulate space modules.

- *Micro Meteorites & Orbital Debris (MMDO) protection*: It represents the protection unit against the impact of moving objects as rocks or other debris.
  Due to high mechanical resistance, it has been supposed to use again Nexter as main components for this layer; however, in order to provide the right separation between Nexter layers and to reduce weight of this unit, it has been decided to add light weight foam blocks as part of MMDO unit.
  It is important to point out that, as rocks do not reach high speeds as micro meteorites, resulting in low energy impacts, MMDO protection unit in this project is thinner than other similar projects.

- *Restraint Layer*: It represents the unit which should withstand to pressure loads. Thanks to high ratio between resistance and weight, it has been decided to use an aramid material for this unit. Other studies suggest different option as Vectran, Kevlar and Spectra; in this work it has been decided to use Kevlar which represents the most used material used for inflatable application in space.
  It is important to point out that Kevlar very sensitive to moisture and UV rays, for this reason, it is

necessary to protect it from both threats.

- *Bladder*: It has the aim to contain the atmosphere inside the habitat, avoiding air leakage and protecting outer layers from free oxygen and moisture. Reducing air leakage would decrease the need of oxygen and nitrogen supply from earth and so reduce costs and weight for transportation.
  Bladder, as MMDO protection unit, is composed by layers of different materials arranged alternately to provide the right separation between them. For bladder are typically used polymers fibres and silicones due the fact that ceramic and glass fibres would increase the risk of puncture. In this work it has been decided to use CEPAC HD200, material with very low permeability already used for BIGELOW, and again Kevlar to separate CEPAC layers.
  It is important to point out that, as the bladder is the innermost unit, human safe requirements are critical for design of this layer and in particular, it is necessary to ensure the low flammability and the high resistance to punctures of material used.

- *Inner Layer*: It represents the interface between habitat and shell. As for bladder, it is necessary to ensure high level of safety for human when material layer is chosen. In this work it has been selected NOMEX.

Once the material for each unit has been chosen, it is necessary to select the number of layers and the thickness. In this study it has been decided to select layers and thickness based on existing project and not to design new one; this because it would have required complex thermal and structural analysis which are not the main focus of this work, what is more, structural analysis for inflatable structures are not easy and it would have required complex assumptions and more in-depth study.

Name of Layer	Main Function	Material	# of Layers	Thickness of layer [mm]	Total Thickness [mm]	Density [kg/m^3]	Starting radius [m]	Final Radius [m]	Layer Weight [kg]
Internal Layer	Bladder protection	Nomex	2	0,5	1	962	3,5	3,501	363,382888
Bladder			4	0,954	3,816	859,7484	3,501	3,504816	1240,13169
Bladder main layers	Atmosphere and moisture cointment	CEPAD HD200	4	0,254	1,016	1300			
Bladder separation layers	Separation between main bladder layer	Kevlar	4	0,7	2,8	700			
Restraint Layer	Withstand pressure loads and give structural stiffness	Kevlar	2	5	10	1440	3,504816	3,514816	5453,8873
MMDO			3		156,2	119,3086	3,514816	3,671016	7225,35323
MMDO main layers	To withstand to rock slides impact	Nextel	3	1,4	4,2	2700			
MMDO separation layers	Foam separation between MMDO main layers	Polyurethen Foam	2	76	152	48			
Multi Layers Insulation	Thermal Insulation	Mylar/Kapton	20	0,025	0,5	1420	3,671016	3,671516	281,276545
External Layers	Dust protection	Nextel	2	1,4	2,8	2700	3,671516	3,674316	2996,34702
Totals			39		174,316		3,5	3,674316	17560,3787

Table 4.4.2-2 Resume of Shell layers

4.5 Airlock

Airlock is a fundamental component to allow access to habitat from outside and vice versa. The main reference to design the airlock have been the ISS Joint quest lock and the Gateway lock (developed by

NASA) [22].
The airlock is composed by three main part: an equipment lock and two inflatable crew lock.
Equipment lock is a rigid prismatic structure used to prepare crew for EVA, here spacesuits are stored. Inflatable crew locks are two cylindrical structure which represent the real interface between habitat and external; inside the crew locks, pressure is slowly decreased from the nominal value inside the habitat up to a very low pressure similar to the external one. In addition, inside the crew lock are installed all dust removal systems which clear the spacesuit from moon dust after the EVA.

The choice to use two part for airlock is made for two main reasons:

- *To reduce the air loss for each EVA*: as part of atmosphere is loss during the opening of airlock due to the higher pressure inside the airlock, it is necessary to reduce the volume of airlock. Using an equipment lock, which remain roughly at the same module pressure (in real a bit lower) and a crew lock, separated by an hatch from equipment lock, the air lost concerns only the crew lock which is smaller, in volume, than equipment loss.

- *To reduce as much as possible dust contamination*: using two airlock stage whit different pressure decreases the risk of dust ingress from external to habitat module. What is more, as the cleaning is mainly done inside the crew lock, when astronaut come to equipment lock, their suit is supposed to be clean; if they are not completely clean, dust would not enter anyway inside the main module because equipment lock would represent an additional block to dust ingress.

Using as reference the ISS airlock and some other studies [21,22,23], the sizes of airlock module are reported in the following tables.

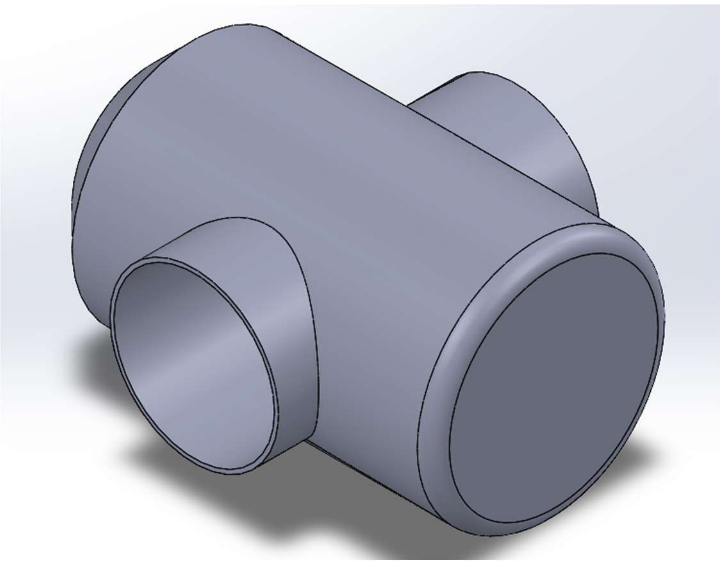


Fig. 4.5-1 Equipment lock

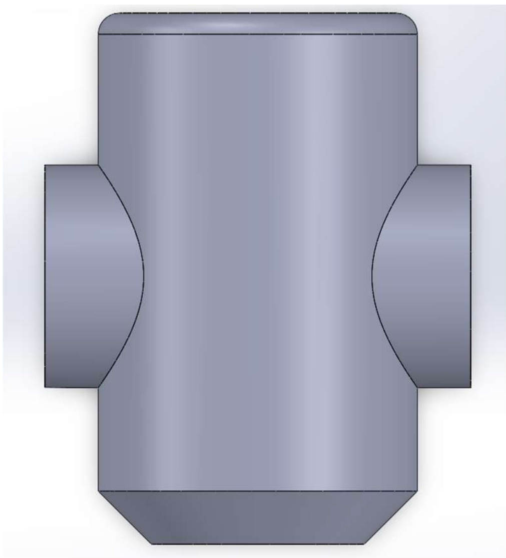


Fig. 4.5-2 Equipment lock view from above

Parameters	Symbols	Values
Crew lock length	L_{EL} [m]	5
Maximum Internal diameter	D_{i-EL} [m]	3
Crew lock volume	V_{CL} [m^3]	35.34

Table 4.5-1 Size of equipment lock

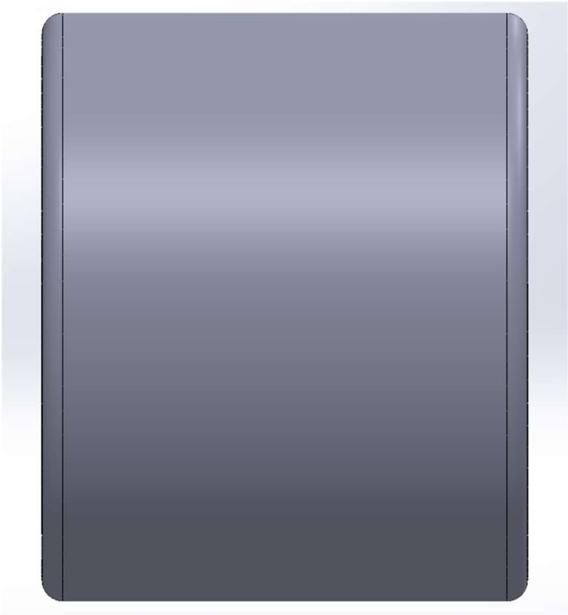


Fig. 4.5-2 Inflatable crew lock

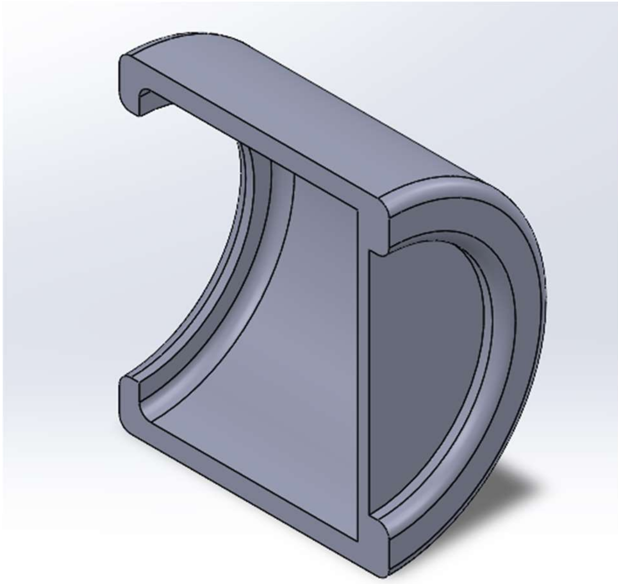


Fig.4.5-3 Section of inflatable crew lock

Parameters	Symbols	Values
Crew lock length	L_{CL} [m]	2
Maximum Internal diameter	$D_{i\text{-}CL}$ [m]	2.5
Maximum External diameter (when deployed)	$D_{e\text{-}CL}$ [m]	2.85
Crew lock volume	V_{CL} [m^3]	9.82

Table 4.5-2 Size of inflatable crew lock

An important issue to be face up concerns the moon dust. Lunar dust represents a huge hazard for both human health and systems. Unluckily due to low gravity and magnetic properties of dust, it easily bonds to spacesuit; when astronauts come back to habitat, after an EVA, their spacesuits are full of this dust. To avoid introducing inside the habitat lunar dust, it is necessary to considers some cleaning system inside the airlock which can remove dust from spacesuit before the ingress inside the main module.
Different approaches have been studied to remove dust from spacesuit:

- *Mechanical removal system* [31]: This system uses rotating brushes to mechanically remove moon dust from the space suit. This solution is very effective with big dust particles, but efficiency drops if smaller dust particles bond to spacesuit.

- *Electrostatic removal system* [32]: This system uses electromagnetic field to remove dust using their electrostatic properties.

- *Gas removal system*: This system uses high pressure gas jets to mechanically remove moon dust from space suit.

Based on these approaches, other hybrid solutions have been proposed. Studies are currently in progress and there are not a lot of data and information about systems used; however, a preliminary study of gas removal system with additional uses of mechanical devices conducted by K. Wood et al. [31], estimate a power consumption of some kilowatts, this value will be used in following part of this study.

4.6 Conclusions

Nowadays structure design is getting more and more importance due to it role within the mission design. Recent studies on lunar habitat suggest to use innovative deployable structure which allow to reduce weight and volume of habitat, decreasing the costs of mission.
In this study it has been considered an hybrid habitat module composed by two part: a primary inflatable shell structure, which separates inside of habitat from outside providing thermal and rock fall protection and a secondary rigid structure which gives the shape to module and withstands vertical loads. Habitat is a vertical cylindric in shape module composed by 4 floors with an overall area of 134.90 m^2 and 573.33 m^3 of internal volume; habitat has been developed in conformance with state-of-arts launchers requirements.
Additional airlock module have been designed to allow EVAs; airlock is composed by a rigid section (equipment lock) which prepares astronauts for activities and two, for redundancy reasons, inflatable section (crew locks) which connect inside with outside and clear spacesuit after the activities.
Some aspects remained unsolved and represents the future work: optimization of external shell design (design of thickness), structural analysis for each phase of mission and identification and design of dust removal system.
In addition to that the still opened gaps are: design of mechanical deployment system and characterization of material for space application (not all material considered have been proved in moon-like environment).

5. Accommodation

In the context of space exploration missions, especially for long duration missions, habitability and wellbeing of crew are fundamental to ensure the success of expedition because they play a key role in psychological aspects of astronauts.
Habitability and wellbeing, which can be gather together in the name of Accommodation, have been studied in several project which have the aim to identify all basic equipment needed to provide a sufficient level of wellbeing for the crew. Data currently available in bibliography have been gathered in some work as the Human Space Flight: Mission Analysis and Design by Wertz et al. [33] and NASA handbook and Advanced Life Support Baseline Values and Assumptions Document [34]. Also ESA/MIT/SoM in their work [4] propose a baseline for accommodation function, providing a list of basic equipment for their lunar habitat.
The main gap in this field concerns the correct identification of all needed equipment to provide a correct level of wellbeing for crew depending on mission duration and mission scenario; identification shall be done reducing as much as possible volume and mass of whole system.
This chapter will focus on the main accommodation functions needed to support human life and their wellbeing during whole the mission; the main functions identified on functional analysis are: food and dining, sleeping spaces and clothes, hygiene, medical care and wellbeing, entertainment and housekeeping.
In this chapter it will be introduced each high level function e provided a list of basic equipment needed, as well as an estimation of their mass and volume on the basis of bibliography data [33]. Equipment will be divided into dry mass and consumables; dry mass means all equipment which should be launched once and integrated directly with the structure while consumable means all equipment which shall be re supplied with cargo mission periodically or shall be launched in specifically in case of need.
The following main accommodation function have been found in functional analysis: food and galley, sleeping and clothing, hygiene, medical care and wellbeing, entertainment and free time and housekeeping.

5.1 Food and Galley

The galley has the aim to provide all needed equipment and spaces to store, process and eat food.
Freezers are needed to store and preserve food which is supplied from Earth once per mission.
Oven and Microowave oven are used to process food; typically, oven is used for more complex preparation while microwave oven is used for easy and quick preparations.
Dishwasher is needed to wash crockery; this is made to reduce both the mass and volume of cargo missions, which would supply new crockery for every new mission, and the water consumption, in case astronauts have to wash crockery by hand.
Sink and spigot are needed to provide water for food hydration and drinking needs; it can be provided both hot and cold water.
Finally, rack is needed to sustain all food equipment and to provide a support surface for eating.
The main consumable are food and other eating supplies, which are supplied by Earth once every 6 month (once per mission).

Galley and Food	Mass per item		Total Mass (incl. Margin)		Volume per item		Total Volume (incl. Margin)	
DRY MASS								
Freezers (x2)	50	kg	105	kg	0,25	m^3	0,525	m^3
Oven (x1)	50	kg	52,5	kg	0,25	m^3	0,2625	m^3
Microowave oven (x1)	70	kg	73,5	kg	0,3	m^3	0,315	m^3
Dishwasher (x1)	40	kg	42	kg	0,56	m^3	0,588	m^3
Sink, spigot and potable water dispenser	15	kg	18	kg	0,0135	m^3	0,0162	m^3
Rack (x1)	105	kg	115,5	kg	1,571	m^3	1,7281	m^3
CONSUMABLES								
Food and eating supplies	2,3	kg/(CM*day)	1738,8	kg	0,008	m^3/(CM*day)	6,048	m^3

Table 5.1-1 List of Galley and Food equipment

5.2 Sleep accommodation and clothing

The main component of sleeping and clothing accommodation is the private quarter (one per astronaut). Private quarter represents the personal space for each crew member used for sleeping, working and recreation. Due the high importance of this personal space and due the fact that astronaut will spend great part of his time, quarter shall provide all needed equipment as desk, bed personal stowage, etc…In addition to that, private quarter shall provide a sufficient level of privacy and sound isolation to avoid interference with other crew members.
Washing and drying machine are used to clear clothes. This choice is made to save mass and volume as not using these machine means having to supplies clothes from Earth with an increased mass and volume of cargo missions.
Racks are used to stow all personal equipment and to sustain all equipment; one personal rack for astronaut has been considered.
Consumables concern all personal stowage and personal crew member clothes, all these consumables are launched once per mission together with crew and depends on the number of astronauts.

Sleep Accomodation and Clothing	Mass per item		Total Mass (incl. Margin)		Volume per item		Total Volume (incl. Margin)	
DRY MASS								
Private quarter (bed+desk)	100	kg/CM	440	kg	2,5	m^3	15	m^3
Washing Machine (x1)	100	kg	105	kg	0,75	m^3	0,7875	m^3
Drying Machine (x1)	60	kg	63	kg	0,75	m^3	0,7875	m^3
Rack (x4)	105	kg	462	kg	1,571	m^3	6,9124	m^3
CONSUMABLES								
Personal stowage	25	kg/CM	120	kg	0,38	m^3/CM	1,824	m^3
Clothing	69	kg/CM	303,6	kg	0,224	m^3/CM	0,9856	m^3

Table 5.2-1 List of Sleep accommodation and clothing equipment

5.3 Waste collection and hygiene

Waste collection and hygiene function has the aim to provide all needed equipment to gather and collect biological waste and to provide hygiene capabilities for crew members.
Toilet systems are needed to gathers all body waste, it has been decided to use two systems (one for two crew members) to reduce as much as possible mass and volume but providing a great level of redundancy and allow astronauts to use at the same time the toilet (in case of need).

Shower is needed to clear astronauts body; as it has been made for toilet system, two showers have been implemented.
Handwasher/Mouthwasher faucet is used to allow hand and face washing as well as tooth brushing.
Ranks are used to stow personal hygiene kits and to sustain other equipment.
Consumables concern the personal hygiene kits (tooth brushes, toothpaste, etc…) and other hygiene supplies. Also in this case, supplies are sent once per mission together with crew.

Waste Collection and Hygiene	Mass per item		Total Mass (incl. Margin)		Volume per item		Total Volume (incl. Margin)	
DRY MASS								
Toilet system (x2)	45	kg	94,5	kg	2,15	m^3	4,515	m^3
Shower (x2)	75	kg	157,5	kg	1,41	m^3	2,961	m^3
Handwasher/Mouthwasher faucet	8	kg	16,8	kg	0,01	m^3	0,021	m^3
Rack (x2)	105	kg	231	kg	1,571	m^3	3,4562	m^3
CONSUMABLES								
Hygiene Supplies	0,075	kg/(CM*day)	64,8	kg	0,0015	m^3/(CM*day)	1,296	m^3
Personal hygiene kit	1,8	kg/CM	8,64	kg	0,005	m^3/CM	0,024	m^3

Table 5.3-1 List of Waste collection and hygiene equipment

5.4 Wellbeing and Heath

Wellbeing function is vital for astronauts to overcome partial gravity issues and to avoid problems when they return on Earth especially for long duration missions.
Exercises equipment includes two different exercises machine (treadmill or cyclette or rowing machine) plus additional equipment for free body exercises. Additional virtual reality devices could be used to improve the level of exercises experience. However, in this preliminary study virtual devices are not considered.
Additionally to wellbeing, also Heath function is critical for the success of mission; in particular the module shall provide spaces and equipment in case of urgent medical need. For this reason, medical suite shall be provided.
Rack ais used to support medical equipment.
Concerning consumables, medical additional equipment should be considered in case of need.

Well-Being and Heath	Mass per item		Total Mass (incl. Margin)		Volume per item		Total Volume (incl. Margin)	
DRY MASS								
Excercises equipment (x2)	290	kg	348	kg	0,19	m^3	0,456	m^3
Rack (x1)	105	kg	110,25	kg	1,571	m^3	1,64955	m^3
Medical/Surgical/Dental suite	500	kg	525	kg	2	m^3	2,1	m^3
CONSUMABLES								
Medical consumables	500	kg	600	kg	2,5	m^3	3	m^3

Table 5.4-1 List of Wellbeing and Heath equipment

5.5 Maintenance and housekeeping

Housekeeping function concerns the cleaning of habitat module and maintenance of it in case of failures or repairing needs.
Restraints and mobility aids are equipment needed to provide a certain level of safety for astronauts during potentially dangerous activity, as repairing in low access part of module or cleaning.

Racks are used to stow maintenance and housekeeping equipment and consumables.
Vacuum is used to gather particles from surfaces.
Trash compactor is used to process solid waste which cannot be recycled, reducing their volume.
In addition to these basic housekeeping equipment, other maintenance equipment is needed; this is considered as consumable equipment as it is supplied in case of need.
Trash bags are also considered as consumables and they are provided from Earth once per mission.

Maintenance and Housekeeping	Mass per item		Total Mass (incl. Margin)		Volume per item		Total Volume (incl. Margin)	
DRY MASS								
Restraints and mobility aids	50	kg	55	kg	0,27	m^3	0,297	m^3
Rack (x2)	105	kg	231	kg	1,571	m^3	3,4562	m^3
Vacuum (prime + 2 spares)	13	kg	14,3	kg	0,07	m^3	0,077	m^3
Trash compactor (x1)	150	kg	165	kg	0,3	m^3	0,33	m^3
CONSUMABLES								
Hand tools and accessories	300	kg	330	kg	1	m^3	1,1	m^3
Test equipment (oscilloscopes,	500	kg	550	kg	1,5	m^3	1,65	m^3
Fixtures, large machines tools, etc.	1000	kg	1100	kg	5	m^3	5,5	m^3
Trash bags	0,5	kg/(CM*day)	396	kg	0,001	m^3/(CM*day)	0,792	m^3

Table 5.5-1 List of Maintenance and housekeeping equipment

5.6 Free time and entertainment

Entertainment plays a key point to mental wellbeing of crew; a typical schedule includes slots for recreational activities which allow to avoid burnout due to heavy work cycles.
Free time entertainment includes spaces and equipment used by crew members to relax and spend some time.
Sofà and armchairs are used to sit and relax.
TV rack sustain screens and additional entertainment equipment.
Entertainment systems includes all electronical devices needed to watch movies or play videogames.
Consumables include additional entertainment equipment which can be provided to crew once per mission.
Movies are also considered consumables even if they are not physical entities.

Free Time and Entertainment	Mass per item		Total Mass (incl. Margin)		Volume per item		Total Volume (incl. Margin)	
DRY MASS								
Sofà	50	kg	60	kg	1,38	m^3	1,656	m^3
Armchair (x2)	43,7	kg	96,14	kg	0,78	m^3	1,716	m^3
TV rack	29,71	kg	32,681	kg	0,29	m^3	0,3045	m^3
Entertainment system	10	kg	12	kg	0,3	m^3	0,33	m^3
CONSUMABLES								
Films	0	kg	0	kg	0	m^3	0	m^3
Photography equipment, lenses	120	kg	126	kg	0,5	m^3	0,525	m^3

Table 5.6-1 List of Free time and entertainment equipment

5.7 Conclusion

During space human mission, especially for long duration mission, it is fundamental to consider habitability issue. This means to consider several functions connected to the presence of humans inside the spacecraft and their subsistence. Past studies highlight how duration and scenario condition influence the design of accommodation spaces and equipment; this represents still a gap that should be solved by analysing each mission separately and providing a reasonable list of equipment needed for that mission.
In this chapter it has been analysed the problem considering a Lava Tube scenario and a mission duration of 180 days per crew. For each function it has been provided a basic list of equipment and it has been provided an estimation of mass and volume using data available in bibliography.
The results in terms of mass and volume are reported into the following table.

Final Totals	**Total Mass (incl. Margin)**		**Total Volume (incl. Margin)**	
Dry Mass	3625.671	kg	50.24765	m^3
Consumables	5337.84	kg	22.7446	m^3
Complete	8963.511	kg	72.99225	m^3

Table 5.7-1 Masses and volumes of Accommodation equipment

This represents a very basic design of accommodation system. There are other aspects which have not been considered in this study as for example: illumination, internal space design, colours, etc… All these aspects should be investigated to provide a more complete overview of habitability problem.

6. Power generation system

The power management represents one of the most important function to allow exploration on the Moon as electrical power is required for mostly of human and robotic activities during missions.
Power consumption for habitability is usually higher than consumption for spacecraft and power generation technologies used for satellites or rovers are not usually applicable for habitat. What is more, usually scenarios impose constraints which make some technology unusable (for example the long eclipse periods on the Moon).
Over the years several preliminary studies focus on power generation for south pole habitat to provide a solution to this problem. Some studies, as the one proposed by Cassini [35] suggests to use an hybrid solution between solar panels and regenerative fuel cells to overcome the eclipse problem. Other studies, as the ESA/MIT/SoM one [36], explore different solutions considering both solar panels, and regenerative fuel cells and also nuclear reactors
In this chapter it will be given an overview of Power generation plant proposal to support activities of a Moon Lava Tube habitat. First, Power budget will be studied to get power consumptions requirements. After that it will be conduced a comparative analysis between two candidate power generation technology to select the best option.

6.1 Power Budget

The power budget is built considering the contribution of all the ground elements included in this mission. It includes:

- *Habitat*: Inside habitat electrical power is used for a large part of functions as life support, environmental control, illumination, habitat management, etc…
- *Airlock*: Airlock uses electrical power to feed dust removal system.
- *Communication Plant*: Communication Plant uses electrical power to transmit and receive packets. Typically, power is used to amplify signals.

Concerning habitat, a first estimation of power consumption has been made by comparison considering the most important projects developed in the past and planned for the future, where it will be considered the presence of human crew.
In order to make up a realistic estimation of power consumption, it has been considered projects which have made similar assumptions in terms of number of crew members and mission length.
Similar studies used as reference are shown in Table 6.1-1. [3, 36, 37]

Habitat	Number of Crew	Length of missions [days]	Minumum of Power Required [kW]	Maximum of Power Required [kW]
ISS	From 3 up to 6	180	84	110
ESA/SoM/MIT	4	500	57	60
Lunar Gateway	4	7 days	60 (based on PPE requirements)	More than 60

Table 6.1-1 Similar Space habitat projects and their Power consumptions

The typical power required for 4 crew members mission with length of about 6 months (each mission), is comprised between 60 kW and 110 kW with a mode value of about 60 kW.

Airlock power consumption depends on technology used to remove power dust. In this study it has not done in-depth analysis of dust removal system, for this reason it is not possible to have accurate value for power consumption. However, to estimate the power consumption of airlock it has been used as reference the study conducted by K.Wood et at. who made a preliminary analysis of dust removal system based on mechanical and gas jets technologies. [31]. Power consumption assumed is 4 kW.

Communication Plant consumption depends on several factors; a first estimation of power required has been made in chapter 7. Power consumption assumed is 2 kW.

Once the single contribution to power demand has been found, it is necessary to compute average power required both during nominal scenario and during emergency scenario. Based on experience of ISS, it has been assumed that peak of power can reach up to 30% more than average state (for example due to experiments). Concerning emergency state, it has been supposed that only vital and communication functions will be active during this condition; based again on experience of ISS [35], standby condition represents roughly the 42% of nominal power consumption.
Power requirements are reported in table 6.1-2 where an additional 15% of safety margin has been considered.

Parameters	Symbols	Values
Average Power	P_{AVG} [kW]	75.9
Peak Power	P_{PEAK} [kW]	98.7
Emergency Power	P_{MIN} [kW]	31.9

Table 6.1-2 Power requirements

6.2 Power Sources

In this study It has been considered two different options for Power Plant: Solar Panels + RFC solution and Nuclear Reactors. The two options have been selected using information about the power required and the length of habitat life. Once the power required has been known and assumed the length of mission, it has been used Fig. 6.3-1, taken from *Space Mission Analysis and Design.* by Larson and Pranker [38] to identify the best source options for Power Plant.

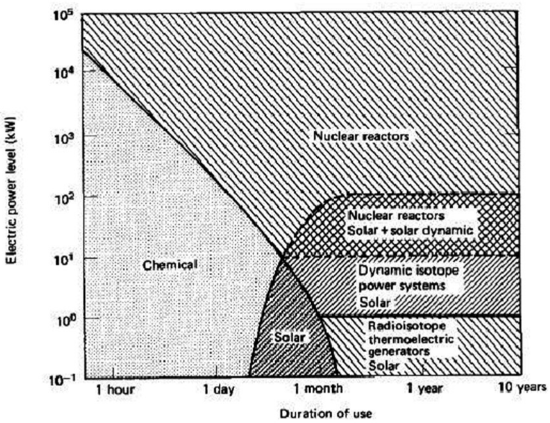


Fig. 6.2-1 Power source selection table [38]

6.2.1 Solar Panels + Regenerative Fuel Cells Option

The first option considers to use an hybrid Power Plant solution to produce electrical power both during daylight period thanks to solar panels and dark period thank to RFC and Electrolyser system.
The whole Power Plant is composed by: Solar Panels field outside the Lava Tube, RFC system near to habitat, Electrolyser near to habitat and tanks (where water, hydrogen and oxygen are stored) field placed nearby Habitat. Additional batteries are considered to supply peak of power demand.
During the daylight period, solar panels provide power to all users. In the meanwhile, Electrolyser take water from tanks and, using power provided by solar panels, transform into hydrogen and oxygen, which will be stored into tanks at high pressure. During the eclipse, RFC provides power using hydrogen and oxygen stored into tanks; the output water of RFC is stored again into water tanks.
Batteries are charged during sunlight period by Solar Panels.

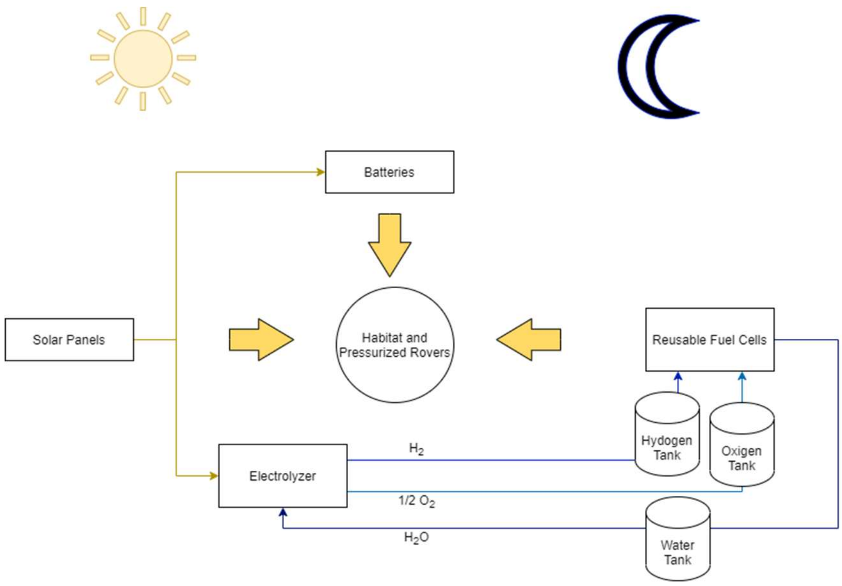


Fig. 6.2.1-1 Hybrid Power generation subsystem scheme

A first estimation of mass, volume and area required by the Power Plant has been carried out in Appendix C using method of Winley and Larson [38].
Before going on analysis of solar plants, it is necessary to briefly recall some features of moon environmental condition at equatorial region:

- *Daylight length*: According to studies conducted on illumination condition on equatorial region, the total length of daylight period is about half of Moon day. It means daylight periods last about 14.5 Earth days.
- *Absence of atmosphere*: On the Moon there is not quite atmosphere; that means whole sun radiation can reach the Moon surface without being blocked by gasses. For this reason, solar cells receive an higher amount of solar power than what they receive on earth.
- *Temperature*: On equatorial regions temperature can reach 120-150 °C due to long sun exposal periods and lack of heat transfer. That means it is necessary to protect solar plants units to avoid damages or decreasing of their performances.
- *Lunar dust*: It has been discovered that Moon dust is sharp in shape and has magnetic properties (due to sun radiation and presence of metals). These two facts make it hazard for solar panels because dust can easily bond to solar cells reducing the among of radiation absorbed. For this

reason, it is necessary to avoid dust can deposit on surfaces; this is made by elevating solar panels above Moon surface. It has been discovered that due to magnetic properties and reduced gravity, dust can float into the air up to 2 meters above the surface.

In this work it has been decided to use GaAs solar cells. GaAs cells are more and more used in space field due to their greater efficiency (about 20-30%) than Silicon (about 10-15%). What is more, thanks to use of multi layers and junctions, it is possible to protect cells reducing degradation due to radiation. Finally, it must be said that these cells have already been used in several space missions and so have a good level of qualification. In this work it has been used the AzurSpace GaAs cells [39].
In addition to solar panels, it is necessary to consider supporting units which can make solar cells possible to produce energy, these units are:

- *Structure and moving*: It comprises all components which withstand the weight of solar panels and allow them to point in sun direction during his path.
- *Thermal control*: It comprises all components which control temperature of solar panels keep it into the operating range. This is done because due to sun radiation, a huge among of heat is transferred to solar cells and it could bring temperature above the operating range.
- *Electrical equipment*: it comprises all devices which control and regulate the power generated by solar panels as well as electrical protection devices.
- *Integration equipment*: It includes all equipment needed to assembly and connect solar cells to solar panels.

Fig 6.2.1-2 Solar cell by AzurSpace

The final outputs of this preliminary analysis on Solar Panels plant are reported into the Table 6.2.1-1.

Parameters	Symbols	Values
Power produced by solar panels	P_{SA} [kW]	223.45
Area (only solar cells)	A_{sc} [m^2]	1465.12
Launch Volume	V [m^3]	73.26
Total Mass (only generation subsystem)	M_{TOT} [kg]	34234.31

Table 6.2.1-1 Resume of main outputs of Solar panels plant

Where:

- *Power produced by solar panels*: It takes into account both the power required by all users described above and power required by electrolyzer to transform water into hydrogen and oxygen.

- *Area (only solar cells)*: It is the whole area required by solar cells to produce the require power.

- *Launch volume*: It has been roughly computed, known total solar panels. Considering 0.05 m^3/m^2 [40].

- *Total mass*: It takes into account all supporting units needed to make sure that the solar panels can generate the required power as well as the solar cells weight.

Once the preliminary analysis has been done for solar plant, it Is necessary to carry out also for RFC plant. Detailed analysis has been carried out on Appendix D.

RFC plant comprises all units needed to: produce power, produce fuel starting from water, store fuel and transport it to RFC stack as well as all needed subunits useful to allow RFC stack to work (cooling units, pressure control units, etc…). In this analysis the focus will be only on: RFC stack, Electrolyser and Tanks.

In this work it has assumed to use an PEMFC. FC stack is composed by several cells connected in series. This kind of Fuel cells uses pure hydrogen and oxygen to produce electrical power, heat and water. Electrolytic membrane is made in Teflon so that ions of hydrogen can easily pass from anode to cathode; electron can not pass through the membrane and they are forced to pass through external circuit producing an electrical current. The advantages to use PEM fuel cells are:

- *Low reaction temperature*: PEMFC require lower temperature than others fuel cells type (for example SOFC). Temperature is about 50-200°C.
- *Quick start*: The time to start up the process is lower than other fuel cells type.
- *Higher energy density*: The ratio between energy produced and weight is lower than other solutions.

Typically, PEMFC has lower efficiency than others fuel cells and they require additional supporting units to avoid internal issues (ex. dehydration of membrane).
In this work it has used as reference Ballard FCveloCity-HD85 [41]; all data are reported above:

Parameters	Symbols	Values
Power produced per stack	P [kW]	85
Idle Power	P_{IDLE} [kW]	6
Weight of Fuel cell stack	M_{FC} [kg]	361
Volume of fuel cell stack	V_{FC} [m^3]	0.737
Feeding pressure	P_{FEED} [bar]	8

Table 6.2.1-2 Ballard Fuel cell data sheet

Concerning the system to produce oxygen and hydrogen starting from water, in this study it has been chosen to use high pressure electrolyser. This system is able to produce fuels at high pressure (from 10 up to 200 bars and above) without using dedicated compressors. This represents a huge advantage as hydrogen need to be stored at high pressure due to the low density and so the huge storing volume needed at normal pressure condition.
Studies about high pressure electrolysis for space or domestic application are still in progress and specification are not available. However, systems for high pressure hydrogen production for industrial application are already available and they represent a good reference for a preliminary estimation of system.
The reference project, from which specifications have been taken is PRETZEL project [42], managed by DLR and funded by European Union's H2020 program.

Parameters	Symbols	Values
Power consumption per volume flow unit	P_{CON} [kW/(m^3/hr)]	25
Hydrogen volume flow rate	V_{H2_dot} [Nm^3/hr]	4.5
Output pressure	p_{out} [bar]	100
Operating temperature	T_{oper} [°C]	90

Table 6.2.1-3 PRETZEL project specifications

Concerning sizes and weight of systems, there is not available data about it and so it is not possible to estimates these values.
In Appendix D has been performed analysis to estimate the high pressure electrolyzed power consumption; results are reported below.

Parameters	Symbols	Values
Required hydrogen volume flow rate	V_{H2_dot} [Nm^3/hr]	815.73
Electrolyzed power required	P_{ELEC} [kW]	251.49

Table 6.2.1-4 Power required by electrolyser

The last aspect to be studied about Solar panels + Regenerative Fuel cells option is storage. Storage is needed to deposit both fuels produced by electrolyzed during the daylight and water produced by Fuel cell stack during eclipse periods.
An important aspect to be studied concerns storage conditions. The biggest problem is related to the high volume required to store hydrogen; as hydrogen in standard condition has a very low density, it occupies large volume and so it is necessary to use big and heavy tanks. There are two options to decrease the storage volume of hydrogen: to liquefy hydrogen or to store it in high pressurized tanks. First option is quite complex to implement as the liquefy point of hydrogen is very low (about 50 K); this means it is necessary to have heavy and complex systems to decrease hydrogen temperature and maintain it inside the tanks. The second option is more feasible and so it is used in this study.
In particular it has used the following storage conditions:

Parameters	Symbols	Values
Hydrogen pressure	P_{H2} [bar]	100
Hydrogen Temperature	T_{H2} [°C]	43
Oxygen Pressure	P_{O2} [bar]	100
Oxygen Temperature	T_{O2} [°C]	43
Water Pressure	P_{H2O} [bar]	50
Water temperature	T_{H2O} [°C]	25

Table 6.2.1-5 Resume of storage conditions

While pressure values have been assumed on basis of electrolyser output values, temperature has assumed considering the values of temperature of water and temperature of Moon environment.
In this study it has been decided to consider inflatable tank structure. This has been chosen to decrease volume and weight and improving the mechanical and thermal properties of tanks.
In particular, inflatable structure is composed by a MLI part, composed by 10 layers of Mylar/Kapton material and a Restraint part, made of Kevlar, which has the role of withstand the pressure loads. Kevlar has been chosen thanks to high mechanical properties.
In table below have been reported the final total values of mass, volume and area of tanks (following analysis on Appendix D).

Parameters	Symbols	Values
Internal volume	V_{INT} [m^3]	289.62
Tank volume	V_{TANK} [m^3]	4.86
Tank mass	M_{TANK} [kg]	7002.41
Tank field area	A_{TANK} [m^2]	145.46

Table 6.2.1-6 Mass, volume and area specification values for storage plant

Finally, it is necessary to point out that values found using inflatable structure are higher than values found using Aluminium rigid structure. This counterintuitive fact is caused by the high safety factor used to size the Kevlar layer; in fact, according to structural requirements, it is required to use a safety factor of 4. This increases a lot the volume and mass of inflatable structure. In future, when inflatable structure will be studied better and tested in moon like environment, it will be probably possible to decrease the safety factor and so to reduce weight and volume of structure.
Another point to be enlighten concerns mechanical properties; differently from what was done for the main module, tank inflatable structure do not have any MMDO protection layer or bladder; also this has been made to reduce weight and volume. In future design, when it will be possible to reduce safety factor, MMDO and bladder layer should be considered into design.

6.2.2 Fission reactors option

The second option considers to use only nuclear fission reactors to produce electrical power both during daylight and eclipse periods.
Nuclear reactors give a great number of advantages compared to other solutions: lowest specific mass for high power requiring application, reduced volume of system, very long durability, independence by light condition and lack of storage needs. However, nuclear reactors have not been used up to now in real Moon environment and qualification of these prototypes is not comparable with other solutions. Nowadays, more and more projects are studied to design and improve the most safe and efficient nuclear systems as they will be the only solution for long terms mission on Moon and Mars especially for mission with higher number of humans inside habitat or village.
The reference project for this study is KRUSTY, developed by NASA [43]. KRUSTY has the aim to prove the feasibility of small nuclear reactor system which is able to produce 1 kW of electrical power for 12-15 years using as fuel uranium-235. Another bigger prototype, in development for future mars missions will produce up to 10 kW of electrical power using similar systems as the smaller versions. In 2018 KRUSTY project carried out successfully tests to obtain the TRL 5, testing material, control systems and simulating failure and off nominal conditions.

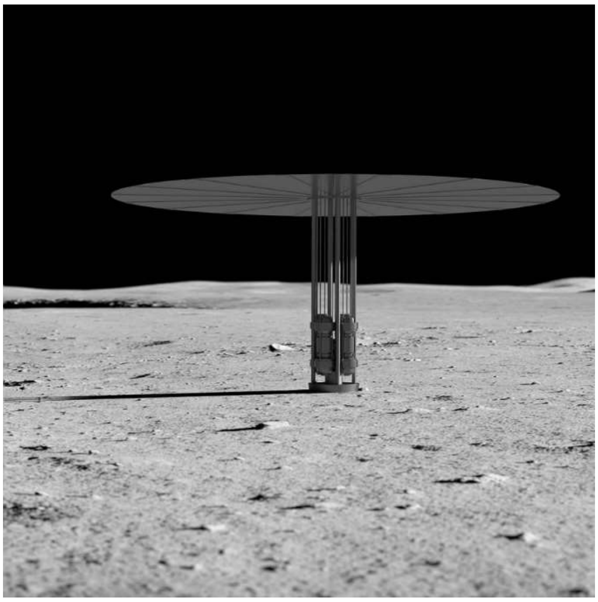


Fig. 6.2.2-1 KRUSTY system developed by NASA [42]

The following specification about KRUSTY system is reported into the following table.

Parameters	Symbols	Values
Electrical power generated	P_E [kW]	10
Thermal power generated	P_{TH} [kW]	40
Mass for each reactor	m_{REAC} [kg]	1500
Area of radiators	A_{RAD} [m_2]	20

Table 6.2.2-1 KRUSTY project specifications

Unlike the solar plant option, nuclear reactors shall cover also peak of power, that means it has been used, as power requirement the peak of power and not the average power.
Based on this requirement, it has been selected a nuclear reactors plant whit following features:

Parameters	Symbols	Values
Power required (peak)	P_{NR} [kW]	98.67
Number of reactor	N_{NR}	10
Total mass of plant	m_{NR} [kg]	15180
Area of radiators	A_{RAD} [m^2]	200

Table 6.2.2-2 Final feature of nuclear reactors plant

Nuclear reactors plant will be located at least 100 meters away from habitable module, this is done for two reasons: to reduce radiation risks for crew and to provide the sufficient surface area for thermal radiators.

Concerning the radiation issues, it must be said that there are not in-depth studies which provide data about the radiation risks for nuclear systems that means it is not possible at the moment to design specifically nuclear reactor field to provide the maximum of protection for humans. However, general precautions, as long distance between plant and module and burial of reactors on Regolith soil, would surely reduce the among of radiation emitted.

6.2.3 Power generation subsystem selection and challenges

Power generation option selected for this study is Nuclear reactors plant. The choice has been made considering the following criteria:

- *Mass and volume*: Nuclear reactors for this high power requiring application, are much more compact than solar panels + RFC solution as it can be seen in the following table. This means it is easier and less expensive transporting nuclear reactors from Earth than solar panels.

- *System complexity*: Unlike the nuclear reactors, solar panels and RFC system need of additional supporting units which get this solution, in terms of system architecture, more complex than the other. This represents a challenge during set up and maintenance operations.

- *Durability*: Nuclear reactors can work for long periods (decades) while solar panels typically can for shorter periods (mainly due to degradation over time). This means nuclear reactors are preferable if habitat is supposed to be used also after the end of mission, as the trend suggest to be.

6.3 Conclusion

Power generation is for space exploration mission a key feature as the huge importance of power on systems. Preliminary studies about lunar habitat showed a strong link between mission assumptions and power sources due to the influence of scenario constraints on systems.
While there are in bibliography a great number of studies about power generation for south pole habitat,

there is not studies who focused on Lava Tube habitat.
In this chapter it was designed the power sources for this kind of habitat. In particular, it has been decided to use nuclear reactors as sources due to his compactness and relative low mass as well as the possibility to use it for even decades, in preparation for future missions on Lava tubes.
Despite nuclear technology looks adapt for this use, it has not been used yet for real Moon application and so it has not been qualified yet for this use; this represents a technology need which should be solved before the real implementation inside the lava tube.
In additional to this, radiation containment issues represents still a gap that shall be studied in-depth to ensure safety of crew during the mission.

7. Communication

Communication is a key function for future exploration missions because provide the only link between crew on Moon surface and mission control centres on Earth or other Moonbases. In case of problems or need to take fast decision, communication between mission control and habitat shall be as quick as possible for this reason real time communication is often a critical requirement for space missions.
The main problem of communication is the distance and the full coverage of the Moon; using the current communication infrastructure on Earth it is not possible to reach all parts of the Moon and so to guarantee communication or even real time communication. What is more, distance increases the losses on communication link forcing to use higher wave frequencies and new technologies.
Over the years the communication problem has only face up partially due to the lack of a real need. Recently space agencies and companies started to tackle the communication problem trying to identify communication and navigation architecture to provide in future communication capabilities. NASA started to work on LunaNet, project which has the aim to build a network to provide communication on Moon between rovers, habitat e other users with Earth. Similarly, ESA is working on Lunar Communication and Navigation Services (LCNS) initiative which has the aim to study deeply the communication problem between Earth and Moon and built companies networks to solve it. Inside the LCNS, ESA Moonlight initiative is focused on relay satellite constellation problem and encourage companies to design and put into orbit their constellations.
The common idea behind all projects and initiative is the need to build a reliable, flexible and expandable communication architecture to provide communication for a wide range of users safely and fast.
In this chapter the communication problems will be faced up providing a general architecture which uses both elements already built and designing new elements. First it will be given a general overview of communication architecture. After that it will be estimate the data rate needed for both uplink and down link. Then, it will be designed a possible relay satellite constellation on lunar orbit and finally it will be designed the communication plant on Moon surface.

7.1 Communication Architecture

The communication architecture proposed in this study takes as example the reference ideas of NASA [16] and ESA [17]; It comprises three main elements: Moon communication plant, lunar relay satellite constellation and Earth ground stations.
Moon communication plant is basically a cluster of antennas and supporting units, located outside the Lava Tube and connected with it via cable. The main aim of Moon communication plant is to send and receive signals from Earth or other bases on Moon. Receiving signals will be amplified and sent to habitat where they will be processed to obtain the packets. Sending signals will be received by habitat and sent to users.
As the plant is located outside the Lava Tube, it is subject to solar radiation and the risk of impact from meteorites. This represents a challenge to be face up in following phases of design.

Lunar relay satellite constellation is a key element to provide communication between different parts of the Moon. In fact, due to the lack of physical infrastructure on the Moon and lack of line-of-sight between far moon bases, it is not always possible to establish a direct link between users on surface. In addition to that,

building a network of relay satellites on orbit allows to expand reliability of communication and accessibility with every part of Moon surface. Lunar relay satellite can also be used as bridge to support communication with Earth and so they represent redundant element for Moon-Earth communication.
Studies are currently focused on the best configuration of constellation; it has been proposed different configurations as explained by Kul B. Bhasin et al. in their work for NASA [44].

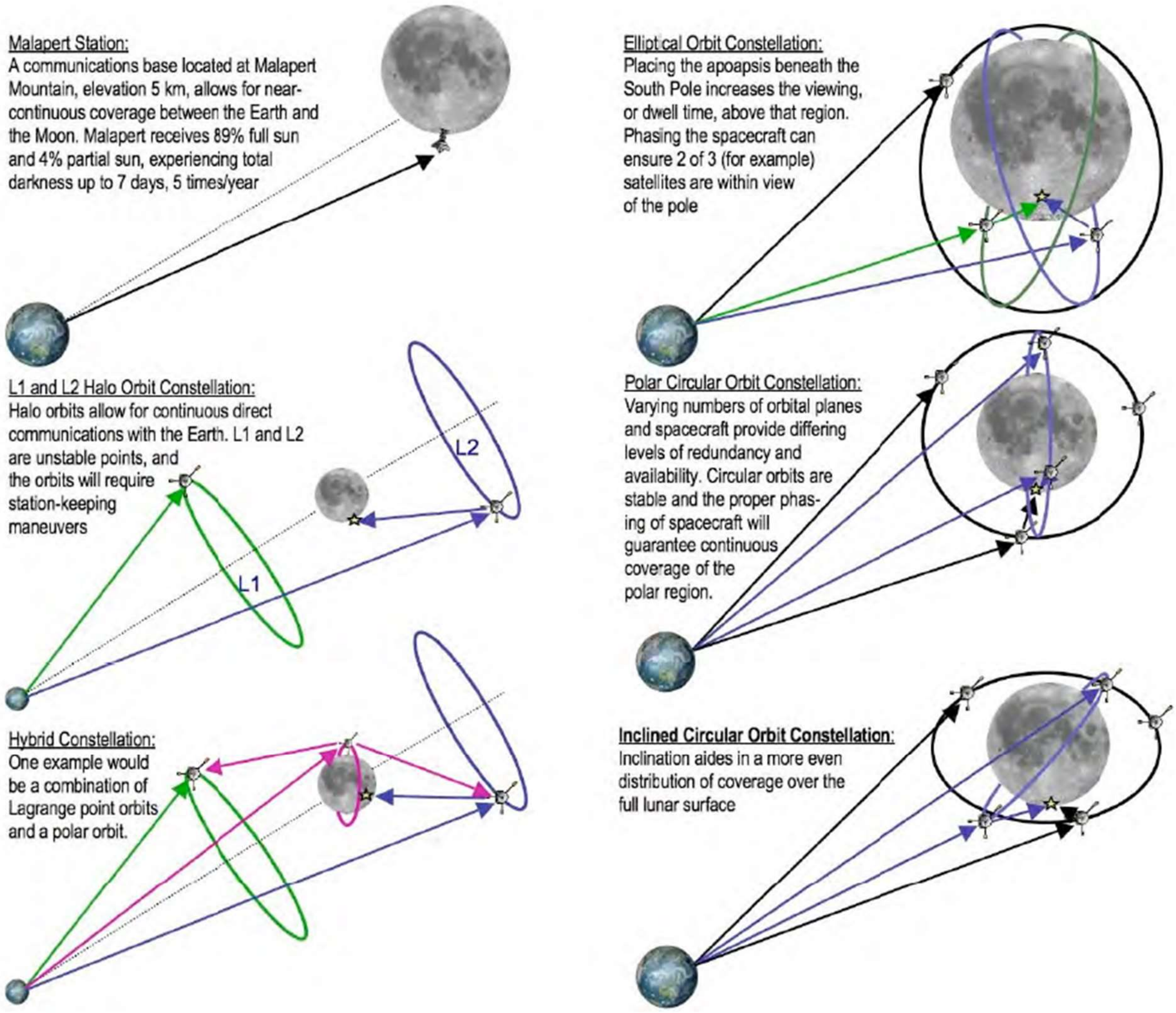


Fig. 7.1-1 Different satellite configuration studied by Kul B. Bhasin et al. [44]

The first configuration suggests putting satellite into HALO orbits to allow continuous communication between Earth and Moon; this is done because HALO orbits allow to have continuous direct communication both between earth and satellite and between Moon and satellite. This configuration is adapted to guarantee continuous communication between Earth and Moon but not to ensure communication in every part of the lunar surface. Also, as L1 and L2 points are instable, it is needed to perform continuously orbital station keeping maneuvers. As in this study, constellation has the main aim to support communication on Moon surface and secondary to support communication Mon-Earth, this configuration has been discarded.

Another interesting configuration suggests using elliptical orbits phasing it to increase the number of satellite in view with a moon base. The main advantage of this solution is the chance to design orbit, in terms of apoapsis so that it is possible to increase the time a satellite remains in view with a lunar base; in fact, the higher is apoapsis the slower will be the satellite increasing the visibility time with base. This solution guarantees a good coverage of Moon surface. This configuration is adapted for the purpose of this study.

Polar circular orbit constellation is mainly used to guarantee continuous communication between Earth and Poles region; however they are not adapt to cover all parts of Moon surface. As constellation shall provide communication between South pole and equatorial region, this configuration has been discarded.

The last configuration suggests using elliptical orbits placed in different inclination plane. This represents the option more adapt to increase the coverage of Moon surface as it is possible to perform cross-link communication between satellite in different orbital plane. Typically design of this configuration is more complex of the other options and the number of satellites involved higher; to keep the design of constellation as easy as possible, in this first phase of design, this solution has been, for the moment, discarded.

On Earth there are two main networks for deep space communication, NASA Deep Space Network (DSN) [45] and European Deep Space Antennas [46]. These networks use big antennas (up to 35 m of diameter) and high frequency band (typically S and X, but can support also higher frequencies) to communicate with spacecraft and probes outside the Earth orbit. Each network comprises three main ground stations separated by an angle of 120° in the Earth surface (longitudinal angle); this is done to guarantee the full coverage of Earth during communication. DSN ground stations are located in: Goldstone (California), Madrid (Spain) and Camberra (Australia).

Fig. 7.1-2 Antenna of Deep space network [Credit NASA]

In addition to Deep space networks, there are several other ground stations which can communicate with Moon; in this case typical value of antennas diameters is 12-15 meters and communication frequency band is typically S or X band. Space agencies plan to improve the ground stations network in following years to get easier communicate with moon during future exploration missions.

7.2 Data Budget

The first step to design the communication plant is estimation of data volume which shall be transmitted both in uplink and in downlink; this shall be done both to communicate with Earth and to communicate with other lunar bases.

In bibliography there is not information about typical volume requirements for moon communication, for this reason the procedure shown below to estimate this value is arbitrary and take the value from Earth application and the few information available in bibliography.

Before going on, it is necessary to define uplink and downlink:

- *Uplink:* Communication from Communication Plant to Earth Ground stations or Relay satellite.

- *Downlink:* Communication from Earth Ground Station or Relay satellite to Communication Plant.

In figures below are reported data volume estimation for both uplink and downlink with Earth and other lunar bases. Data volume estimation has been conducted first defining the purpose of communication, then estimating the period of utilization per day and finally, after the data consumption was know for each communication purpose, multiplying period and data consumption it is possible to obtain the total data volume for each communication purpose in one day.

Target of communication	Purpose of Communication	Type of data	Period of utilization in one day [s]	Data consumption [Mbps]	Data to be trasferred [Mb]
Earth					2895264
	Briefings	Video	7200	0,67	4800
	Housekeeping	Data Packets	86400	0,01	864
	Entertainment	Video	3600	2,67	9600
	Experiments	Data Packets	28800	100	2880000
Other Moonbases					3331,8
	Briefings	Video	3600	0,67	2400
	Vocal commun.	Voice	3600	0,0188	67,8
	Housekeeing	Data Packets	86400	0,01	864

Table 7.2-1 Uplink communication data budget

Concerning uplink with Earth, it has been assumed the following communication purpose:

- *Briefings:* It has been supposed crew and ground control centre would communicate using video call twice per day to update about progress of mission. Each call would last 1 hour and it would consume 2.6 Gb per hour of communication [47].

- *Housekeeping*: Data about housekeeping of systems is gathered continuously (24h per day). It has been assumed maximum of 10000 bps of data production [48].

- *Entertainment:* It has been supposed that once per day each astronaut can make a videocall of one hour with family, friends or media. Data consumption is the same as briefings.

- *Experiments:* Every day astronauts work for 8 hrs (28800 seconds per day). It has been assumed a data production of 100 Mbps [48]. It must be said that this represents the biggest contribute to data volume, however this data strongly depends on mission and so it is not possible to have at the moment an accurate value.

Concerning uplink with other bases, it has been assumed the following communication purposes:

- *Briefings:* It has been supposed crew inside moon bases would communicate using recorded video once per day to update about progress of mission. Duration of video would be 1 hour and it consume 2.6 Gb per hour of communication.

- *Vocal communication:* It has been assumed that crew would communicate also via vocal call for a total duration of 1 hour per day. Consume would be 67.8 Mb per hour.

- *Housekeeping*: Data about housekeeping of systems is gathered continuously (24h per day). It has been assumed maximum of 10000 bps of data production [48].

Target of communication	Purpose of Communication	Type of data	Period of utilization in one day [s]	Data consuption [Mbps]	Data to be trasferred [Mb]
Earth					293491,2
	Briefings	Video	7200	0,67	4800
	Commands	Data Packets	86400	0,008	691,2
	Experiments	Data Packets	28800	10	288000
Other Moonbases					3331,8
	Briefings	Video	3600	0,67	2400
	Vocal commun.	Voice	3600	0,0188	67,8
	Housekeeing	Data Packets	86400	0,01	864

Table 7.2-2 Uplink communication data budget

Concerning downlink with Earth, it has been assumed the following communication purposes:

- *Briefings:* It has been supposed crew and ground control centre would communicate using video call twice per day to update about progress of mission. Each call would last 1 hour and it would consume 2.6 Gb per hour of communication.

- *Commands*: Data about command of systems is gathered continuously (24h per day). It has been assumed maximum of 8000 bps of data production [48].

- *Experiments:* Data about experiments to be performed inside habitat can vary a lot depending on mission. As already said for uplink, also for downlink there is not information about data volume, for this reason it has been assumed to use 10 Mbps of data consumption [48] for a duration of 8 hours.

Concerning downlink with other bases, it has been assumed the following communication purposes:

- *Briefings:* It has been supposed crew inside moon bases would communicate using recorded video once per day to update about progress of mission. Duration of video would be 1 hour and it consume 2.6 Gb per hour of communication.

- *Vocal communication:* It has been assumed that crew would communicate also via vocal call for a total duration of 1 hour per day. Consume would be 67.8 Mb per hour.

- *Housekeeping*: Data about housekeeping of systems is gathered continuously (24h per day). It has been assumed maximum of 10000 bps of data production [48].

After total data volume have been estimated, it was computed the data rate. To do that the following assumptions have been made:

- *Antennas:* Communication plant include 1 antenna for each communication link (uplink and downlink of both Earth and other bases communication). Total of antennas is 4.

- *Period of transit:* Period of transit means the duration of linking for each communication establishment. As proved using STK simulation, Communication plant is able to establish a continuous communication link with at least on antenna of DSN. That means period of transit can be supposed be 24 for earth communication.
  Period of communication for moon base communication has been set on average of 2100 seconds as proven using STK simulation (see next section).

- *Number of transits*: It means the number of communication links established in one day. For Earth this value has been set to 1 because the communication is continuous while for moon bases it has been set to 8 because this is the minimum number of transits of relay satellites.

Based on these assumptions it has been found the following data rates for uplink and downlink:

Target of communication	Data to be trasferred [Mb]	Period in every transit [s]	Number of transit	Number of antennas	Data Rates [Mbps]
Earth	2895264	86400	1	1	33,5100
Surface	3331,8	2105,846667	8	1	0,1978

Table 7.2-3 Data Rates estimation for Uplink

Target of communication	Data to be trasferred [Mb]	Period in every transit [s]	Number of transit	Number of antennas	Data Rates [Mbps]
Earth	293491,2	86400	1	1	3,3969
Surface	3331,8	2105,846667	8	1	0,1978

Table 7.2-34Data Rates estimation for Downlink

7.3 Lunar relay satellite constellation

Lunar relay satellite constellation is considered to provide communication capabilities with other lunar bases and, in case of need, to provide crosslink communication between Earth and Moon.
As anticipated in section 7.1 of this work, the configuration chosen for satellite constellation considers circular orbits with different phasing, to allow a good coverage of Moon. In particular it has been decided to use 3 satellites which orbit on the same plane (same inclination of plan) but out of phase (different RAAN). The number of satellites has been chosen to guarantee at least one satellite in view with each bases roughly in every moment.

Parameters	Symbols	Values
Semimajor axis	a [km]	2437.68
Orbital period	T [s]	10800
Eccentricity	e	0
RAAN	[deg]	0-120-240
Inclination	i [deg]	60

Table 7.3-1 Resume of relay satellites constellation orbital parameters

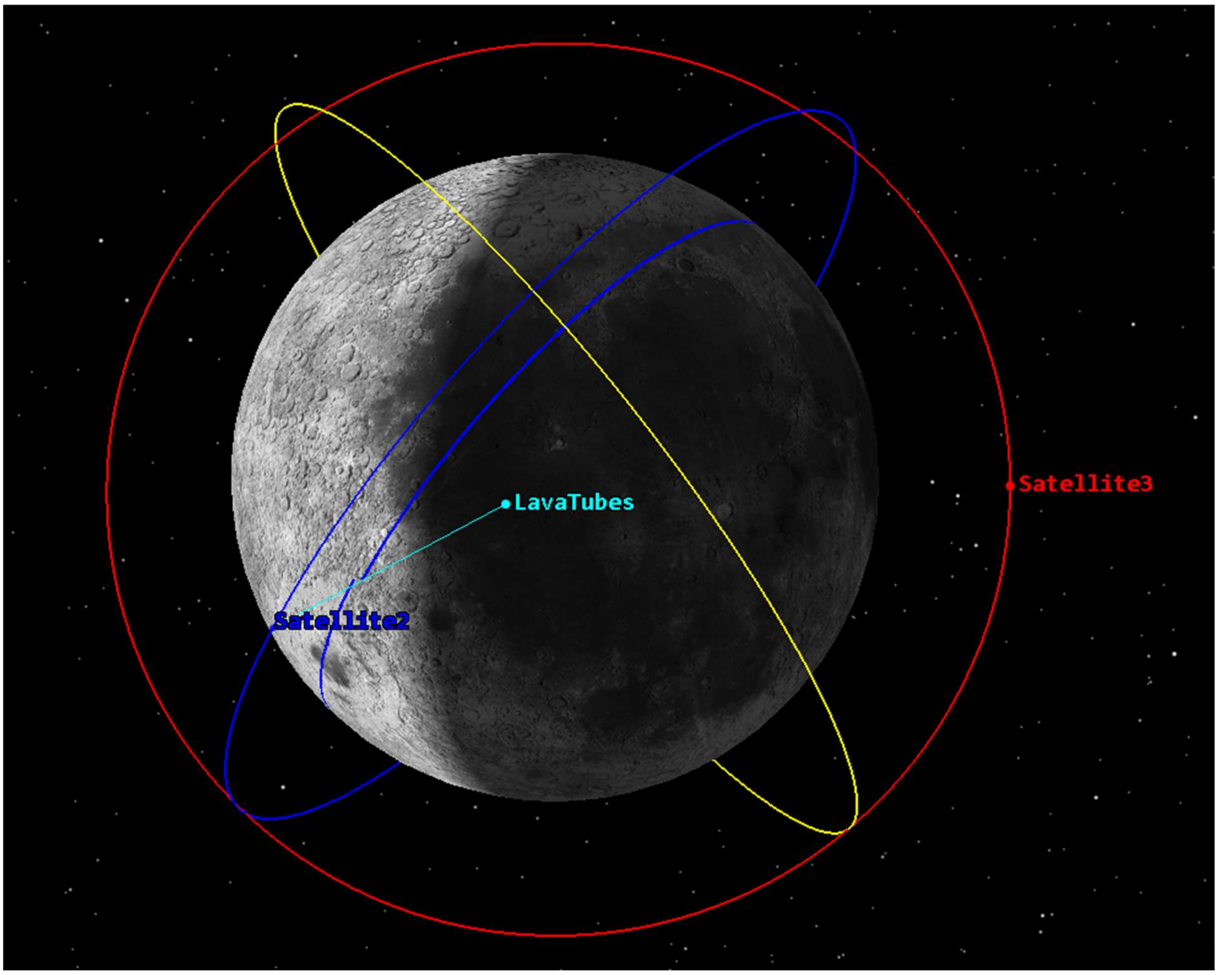


Fig. 7.3-1 Relay satellite constellation

Analysis has been carried out using System Tool Kit (STK). The period used to carry out analysis in STK starts from May 2030 and finishes in May 2035 (5 years of simulation as assumed life of habitat).
STK has been used for two reasons:

- To demonstrate that this configuration guarantee quasi continuous communication between bases and at least one satellite.

- To compute access and period of transit

Concerning the first purpose of analysis, as shown in figure 7.3-2, this configuration satisfies the request; in fact, it can be seen, superimposing the three lines that represent the transits of the satellites within the cone of visibility of the communication plant, that frequent transits are always guaranteed by at least one satellite. The different distribution of frequency transit over the time is caused by the moon rotation motion which, by moving the satellites ground tracks, at certain times, the satellite stops passing within the cone of visibility of the station. Simulation has been performed over one month because it is exactly the duration of lunar day; this means that the same condition repeats periodically every month.

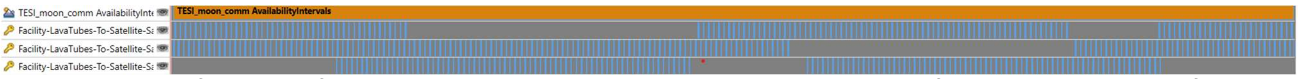


Fig 7.3-2 Transit frequency of satellites inside the Lava Tube communication plant cone of visibility (simulation performed over one month)

Concerning the second aspect, it has been computed using STK simulation, the following data about transits:

	Min duration [s]	Avg duration [s]	Number of accesses in one day
SAT 1	173,6	2067,8	8
SAT 2	220,4	2110,54	8
SAT 3	544,1	2139,2	8
Total [s]	173,6	2105,846667	8

Table 7.3-2 Accesses of satellite inside the Lava Tube Communication Plant cone of visibility

7.4 Communication Plant and Link Budget

Design of communication plant is strongly connected with Link Budget, as the feature of plant depends on closure or not of link budget.
To perform the link budget it has been made the following assumption:

- *Frequency bands:* As suggested by Wiley [48] it has been decided to use S band to communicate with relay satellites and X band to communicate with Earth. To avoid interference during communication, frequency used in uplink and down link is slightly different. Table below resume the frequencies used.

Parameters	Symbols	Values
Uplink Moon-Earth frequency (X band)	$f_{ME,\cancel{UP}}$ [GHz]	8
Downlink Moon-Earth frequency (X band)	$f_{ME,\cancel{DOWN}}$ [GHz]	8.1
Uplink Moon-Moon frequency (S band)	$f_{MM,\cancel{UP}}$ [GHz]	2.68
Downlink Moon-Moon frequency (S band)	$f_{MM,\cancel{DOWN}}$ [GHz]	2.58

Table 7.4-1 Frequencies used to communicate

- *Distance between habitat and communication plant:* As the loss due to cable depends on distance between Communication Plant and Habitat, it has been assumed in conservative way that this distance is 100 m.

- *Antennas:* It has been assumed to use only parabolic antennas both for communication plant and for relay satellite. In particular, relay satellites have parabolic antennas with diameter of 0.5 m.

- *Modulation:* It has been assumed to use digital phase modulation (D-QPSK) typically used for similar projects.

- *Minimum link margin required:* It has been set 6 dB as the minimum of link margin required to close link budget.

Based on these assumptions, it has been computed the link budget using Wiley procedure and considerations [48].
Results and size of plants antennas are reported in tables below.

COMMUNICATION ON MOON SURFACE			
UPLINK		**DOWNLINK**	
Power transmitting (COM Plant) [W]	20	Power transmitting (Satellites) [W]	10
Signal Frequency [GHz]	2,68	Signal Frequency [GHz]	2,58
Modulation	D-QPSK	Modulation	D-QPSK
Transmitting antenna diameter [m]	2,5	Receiving antenna diameter [m]	2,5
Receiving antenna diameter [m]	0,5	Transmitting antenna diameter [m]	0,5
Type of antennas	Parabolic	Type of antenna	Parabolic
Pointing error [°]	1	Pointing error [°]	1
Link margin [dB]	35,08	Link margin [dB]	22,0739

Table 7.4-2 Results of Link budget and size of antennas for Moon surface communication

COMMUNICATION WITH EARTH			
UPLINK		**DOWNLINK**	
Power transmitting (COM Plant) [W]	500	Power transmitting (GS on Earth) [W]	1000
Signal Frequency [GHz]	8	Signal Frequency [GHz]	8,1
Modulation	D-QPSK	Modulation	D-QPSK
Transmitting antenna diameter [m]	5	Receiving antenna diameter [m]	5
Receiving antenna diameter [m]	15	Transmitting antenna diameter [m]	15
Type of antennas	Parabolic	Type of antenna	Parabolic
Pointing error [°]	0,1	Pointing error [°]	0,1
Link margin [dB]	7,12	Link margin [dB]	8,055024

Table 7.4-3 Results of Link budget and size of antennas for Earth communication

It can be seen that both link budget as closed, that means this configuration is feasible. Details about link budget are reported on Appendix E.

However, it is necessary to make a consideration concerning the pointing error. This parameter takes into account the loss generated by a mismatch between the receiving antenna and the transmitting antennas. This error depends on beamwidth of antennas and pointing error. Typically, for parabolic antennas, the bigger is antenna, the higher is the gain and lower is the beamwidth.

Loss pointing error can be expressed by following equation [48]:

$$L_{pointing} = -12 * \left(\frac{e}{\frac{\theta}{2}}\right)^2 \quad [dB]$$

Where:

- e= Pointing error [deg]
- θ= Antenna beamwidth [deg]

It is clear that considering big antennas, and so small beamwidth will require to consider also small pointing error to keep low pointing errors. That could represent a problem because typically pointing errors are limitated (roughly values are 1-0.1). That means using huge antennas as the DSN (35 m diameter) would get impossible to close link budget; for this reason, simulation has been carried out considering communication with smaller antennas (15 m) used by several ground bases all over the world. For future project is it suggested to investigate the role of pointing error on Moon communication design funding solutions to allow also communication with using bigger antennas.

7.5 Conclusion

Communication between Moon and Earth represents, nowadays, a huge challenge to be faced up due to the absence of a communication architecture which can ensure real time communication between ground stations and lunar bases. In addition to that, also communication on Moon surface is hard for the same reason.
This chapter tried to propose a simple communication architecture which ensures real time link between Lava Tube habitat and Earth and communication between this habitat and other Moon bases (for example South Pole base). Architecture comprises a communication plant located outside the cave, a relay satellite constellation in low lunar orbit and a network of grounds station on Earth; communication plant and relay constellation have been designed while ground stations have been assumed to use the existing infrastructure on Earth. Communication between Earth and Moon happens directly using the X band while communication between moon bases happens using the relay constellation on S band.
In this work, first the data volume has been estimated, after that it was build the data budget which allowed to design the communication plant and also it was designed relay constellation using STK software.
This work also highlight some gaps and open points; first, it has been noticed that there is not information about volume required for lunar mission, especially about experiments. Data volumes represent an important information to well design the whole communication architecture.
Another gap concerns the lack of public studies which investigate the design of relay constellation.
The third gap concerns the huge pointing errors; it has been notice that using high gain parabolic antennas, the pointing errors grows a lot; to fix it is necessary to design more precise pointing systems or modify antennas, both solutions affect the overall design of system and so, they must be studied.
Improvement of architecture could be represented by implementation of optical communication which will allow higher data rate, lighter systems, more reliability and safety on information transmission.

8. Environmental Control and Life Support System

One of the most important system in space exploration habitat with manned crew is the Environmental Control and Life Support System (ECLSS), which emulate the terrestrial condition and allow astronauts to live also in lunar scenarios where there is not atmosphere and temperature are often not compatible with human life.

ECLSS is a particular system whose design depends strongly on both habitat assumptions (duration of mission and number of astronauts) and scenario features. Up to now the examples of ECLSS concern both spacecrafts used for short duration mission as the manned capsule or Shuttle and the International Space Station; however, these designs cannot be fully used for habitat on the Moon due to the different mission durations and gravity conditions which affect the design. In addition to that, settlement bases require an high level of independence from Earth as bringing all consumables as water, oxygen, food, etc… from earth would be very expensive and sometimes also impossible due to distances (example of Mars).

Recent studies focused on design of ECLS for habitat on Mars and the Moon trying to design new systems which could provide a nice level of independence from Earth by improving recycle equipment and implementing green houses for food production [4,49].

In this chapter it will be given a general overview on ECLSS defining the system architecture, requirements and estimating budgets for atmosphere and water subsystem. What is more, it will be conducted a preliminary thermal analysis to estimate the power to be produced by heaters. Finally, it will be given an overview on the In-situ resource utilization equipment used to produce oxygen from Lunar Regolith.

8.1 System Architecture

Environmental and Life Support system comprises different functions connected with the human life inside the habitat: Atmosphere Management, Thermal and Moisture Management, Water management, Food Management and Waste management. In addition to these primary functions, another function is typically considered as part of life support function: human protection. In this study, as Lava Tube offers free protection from radiation and meteorite impacts, protection system has not been considered as sub-function.

Identified the main sub-function, it is possible to make a step-forward into the functional analysis, started in chapter 3 of this work, to identify the needed sub-function for each sub-system.

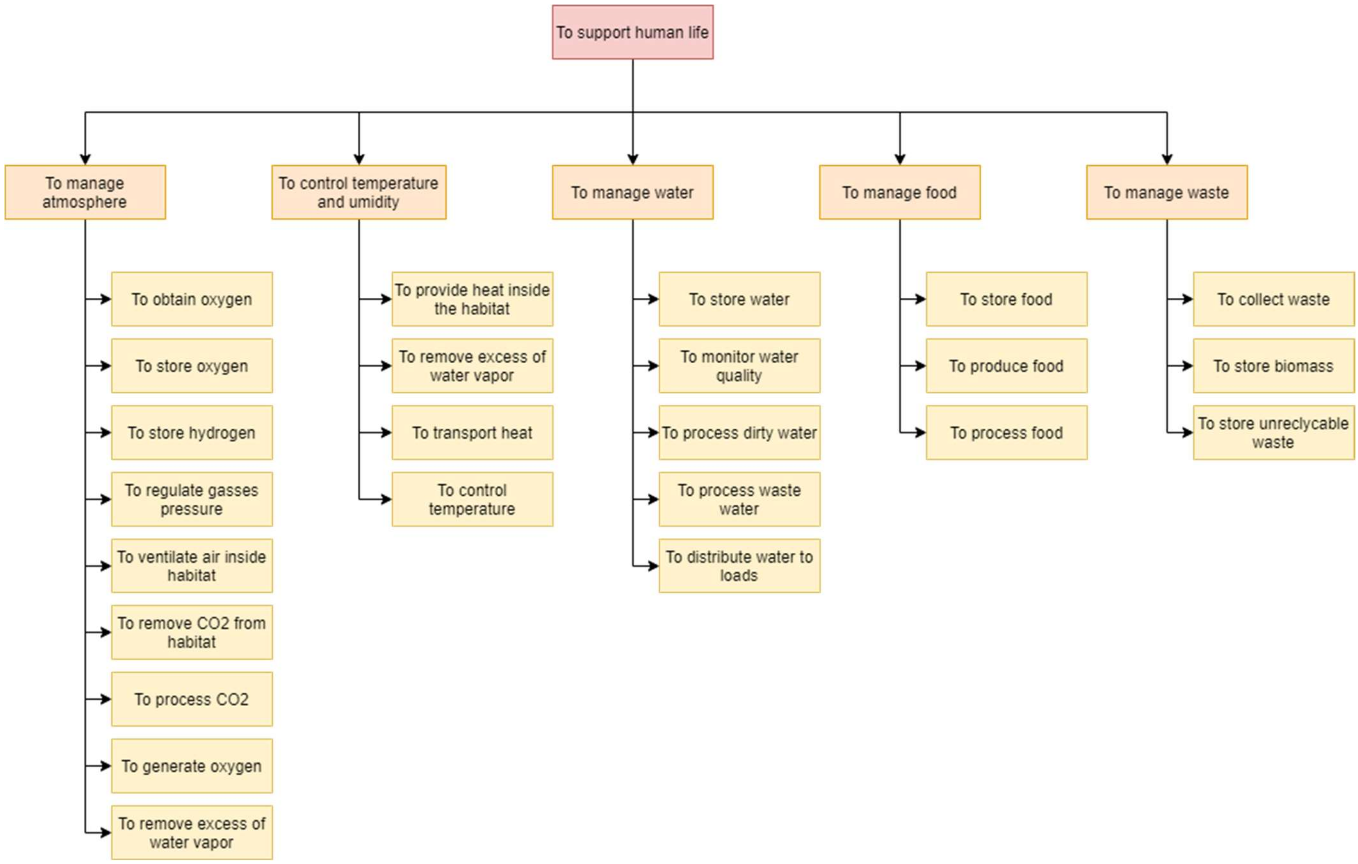


Fig. 8.1-1 Functional Analysis of ECLS System

One of the most important feature of ECLS system is the high interconnectivity of his sub-system; in fact, despite in Fig. 8.1-1 subsystems look to be independent from each other, in reality they constantly interface each other to exchange air or water and to allow a full recycle of consumables.
This happen because ECLS used for Moon and Mars mission should be the most possible independent by Earth and so it is necessary to recycle everything avoiding wastage.
This kind of system is called closed loop system because they don't need, or need only few, resupply from external and they are quite mandatory for long duration mission far away the Earth.
In the follow sub-section it will be introduced the system architecture for atmosphere & thermal functions, water & wet waste functions and food function.

8.1.1 Atmosphere and Thermal Management

Atmosphere and thermal management sub-system has the aim to provide atmosphere capabilities and control pressure, temperature, humidity and air quality. Architecture for this sub-system is shown in figure below, where rectangular boxes indicate an unit, while elliptical bubbles indicate consumables.

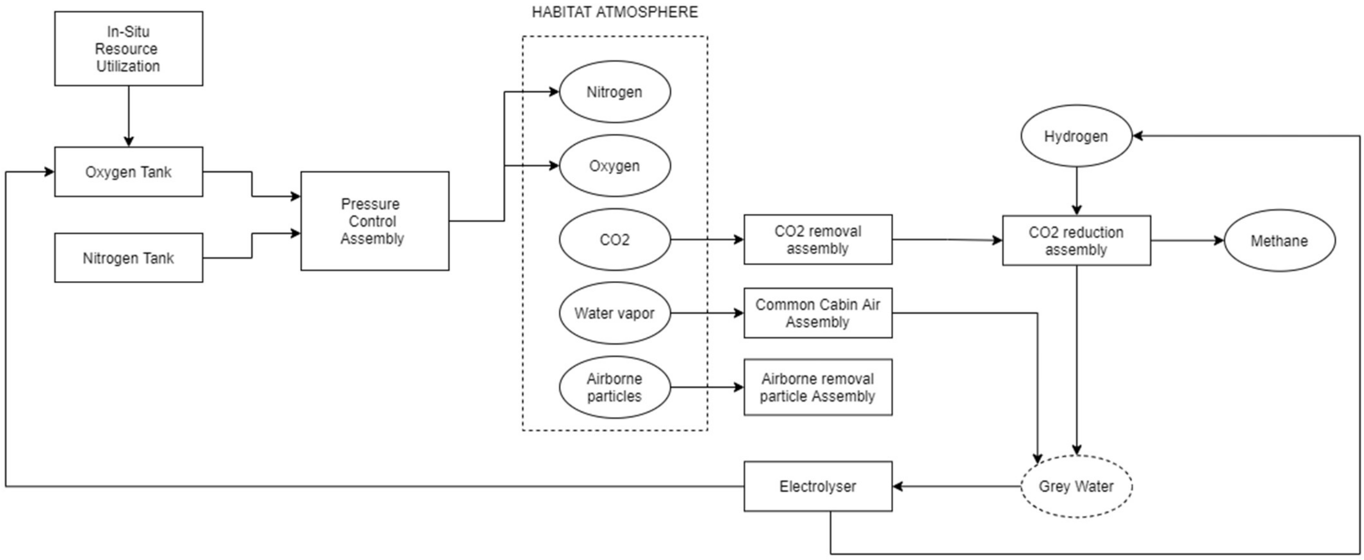


Fig. 8.1.1-1 Atmosphere and Thermal sub-systems architecture

First the basic gases which compose the atmosphere, oxygen and nitrogen, are stored inside specific tanks which monitor quality, pressure and temperature. Gases are sent to Pressure Control Assembly (PCA) which regulate pressure before putting them into habitat atmosphere.

Habitat atmosphere comprises five components: Oxygen and Nitrogen provided by PCA, CO_2 waste production of human respiratory process, Water vapor produced as well by respiratory process or other system and airborne particle as hair, skin waste, food waste, dust, etc…While oxygen and nitrogen can remain into atmosphere, the other three components shall be removed and processed (if possible).

CO_2 is removed from atmosphere by CO_2 Removal Assembly which sent it to CO_2 Reduction Assembly (CO_2RA). CO_2 Reduction Assembly use hydrogen to process the carbon dioxide and produce methane, which could be used as fuel for rockets, and grey water. Grey water is then electrolysed to obtain oxygen, stored inside the tanks and hydrogen, which will be used again to process new CO_2.

Water vapor is removed from atmosphere by Common Cabin Air Assembly which condensate it and sent I to grey water tank. It is important to point out that CCAA is used in general to control temperature of atmosphere, not only to remove water vapor from air.

Finally, airborne particles are removed from atmosphere by a dedicated removal assembly unit.

As it can see, the overall sub-system can be considered closed-loop because both carbon dioxide and vapor water are recycled to produce again oxygen (and water).

However, due to inevitable losses, part of atmosphere shall be resupplied periodically. Reasons of losses will be explained in following section, where they will be also estimated.

In this work, to resupply gases it has been decided to bring nitrogen from Earth using cargo mission and produce oxygen from Regolith using In-situ resource utilization (details will be explained in section 8.5). This has been decided to reduce costs of cargo missions (reducing the total mass) and to get more sustainable and independent the habitat from Earth. Unlucky, it is not possible to produce nitrogen as processing biomass is hard process still in study phase, but it could be implemented in future projects, when data and models will be available and prototype qualified.

8.1.2 Water and Wet Waste Management

Water and wet waste management sub-system has the aim to provide water to for all uses and to collect and process wet waste to obtain again water. In addition to that, subsystem should distribute water to all systems and should monitor and control water quality.

Water is used for different uses. Drinking and food preparation, hygiene, washing machines (when present), food production (when present) and medical needs.
Waste typically produced concerns: urinal fluids, waste water from hygiene, waste water from washing machine, waste medical water and water vapor present into atmosphere. Water consumption will estimate in following sections.
Architecture for this sub-system is shown in figure below.

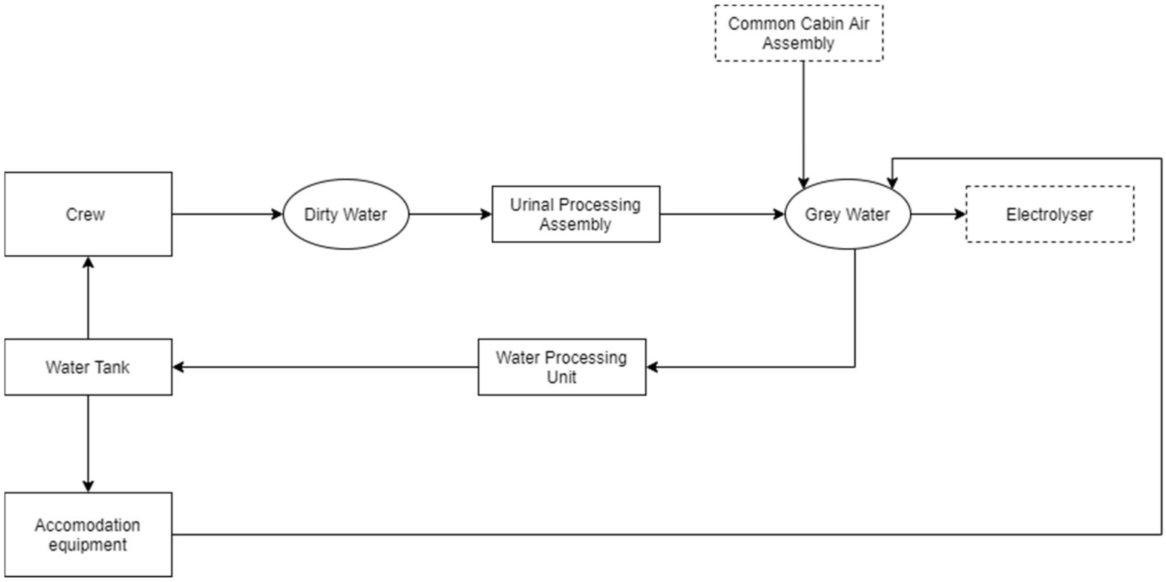


Fig. 8.1.2-1 Water and wet waste sub-systems architecture

Water loop starts when human and accommodation equipment request water. Humans consume water to drink and process food and produce as output dirty water whereas accommodation uses water for their systems (i.g hygiene, medical ,etc..) producing as output grey water. Another source of dirty water, as said before, is the condensed water coming from habitat atmosphere and processed by CCAA.
While grey water can be directly processed by Water processing Assembly (WPA) to produce utilizable water, dirty water needs an additional processing stage; this is done by Urinal processing Assembly which get as input dirty water and produce dirty water.
Water produced by WPA is stored into tanks where temperature and quality is constantly checked.
It is important to point out that, as said in previous section, grey water is also used to produce oxygen by means electrolysis; this represents a connecting point between the two different sub-system demonstrating the strong interconnectivity of ECLS system units.

Also for water sub-system it can be seen that the system is a closed-loop solution as water consumed by humans and accommodation equipment is fully recycled and not wasted. Also in this case, due to losses (explained in following sections), it is necessary to periodically the water. Unluckily, as there is not evidence of water presence near equatorial region of Moon, it is not possible to extract directly water from the Moon but it is necessary to bring it from Earth. Future work can consider the option to use the water extracted by other moon bases at the SouthPole here ice water has been found.

8.1.3 Food Management sub-system

Typically, food management sub-system gathers all functions related to production, storage and processing of food. However, it this study storage and processing functions have been included into accommodation function (Galley and Food) as it has been assumed that it is not necessary to perform particular processing

activities to transform food.
In this study it has been decided to use food brought from Earth and stocked into freezers. This choice has been made because setting a greenhouse for self food production would have been very difficult and not convenient for purpose of this study. In fact, as highlighted by De weck in his work based on Mars One mission [49], to produce the amount of food required by four astronauts it would be required additional CO_2 compared to the one produced by crew. While this is possible on Mars, where atmosphere is composed by roughly 95% of carbon dioxide and so it is possible to extract it using ISRU systems, on the Moon this is not possible due the absence of atmosphere. This would get more complex the food production system and so this choice has been discarded.
Possible other solutions could consider an hybrid system, where part of food is produced by greenhouse and part is brought by Earth [4]. This hybrid solution would have two main advantages: reduce costs of cargo mission and implement a reduce complexity and masses of atmosphere management system, as part of oxygen is produced by plants. Of course, the need to build a greenhouse module would represents an additional challenge both for concept of operations (because it is needed to build and design another module) and for power consumption as additional greenhouse would increase a lot the energy required as demonstrated by Babakhanova et. al [50] in a proposal study for building of greenhouse on mars.

8.2 Requirements and Assumptions

Environmental Control and Life Support System requirements are based on European ECSS document [51] and analogous American document [52]. Both documents impose requirements on functions to be performed and level of performance of units. Values are based either on already tested and operation system or on studies which imposes constraints (for example about atmosphere composition).
In table 8.2-1 are listed the major requirements, useful for a preliminary sizing of ECLS system. They are based on past missions, especially ISS, due to the similarity of it with this study.

ID	Subsystem	Description
ECLS-ATM-001	Atmosphere	The total pressure of atmosphere inside habitable areas shall be: - Minimum 97,9 kPa - Maximum 102,7 kPa
ECLS-ATM-002	Atmosphere	The partial pressure of oxygen inside habitable areas shall be: - Maximum 23,1 kPa - Minimum 19,5 kPa
ECLS-ATM-003	Atmosphere	The partial pressure of nitrogen inside habitable areas shall be minimum 80 kPa
ECLS-ATM-004	Atmosphere	The partial pressure of CO2 shall be lower than 1013 Pa
ECLS-ATM-005	Atmosphere	The airborne particle concentration (0,5-100 mm diameter) shall be lower than 0,05mg/m^3
ECLS-ATM-006	Atmosphere	The atmosphere inside habitable areas shall move within the following range of velocities: - Maximum of 0,051 m/s - Minimum of 0,203 m/s
ECLS-ATM-007	Atmosphere	The air leakage in nominal condition shall be lower than 2,04 kg/day

ECLS-ATM-008	Atmosphere	The air leakage of Airlock during each EVA cycle shall be lower than 10% of whole airlock volume
ECLS-TCS-009	Thermal Control	The temperature inside habitable areas shall be within the range of: - Minimum of 18,5°C - Maximum of 29,6°C
ECLS-TCS-010	Thermal Control	The surface temperature of habitable areas surface shall be within the range of: - Minimum of 4°C - Maximum of 45°C
ECLS-TCS-011	Thermal Control	The humidity inside habitable areas shall be with the range of: - Minimum of 25% - Maximum of 70%
ECLS-TCS-012	Thermal Control	The dew point inside habitable areas shall be within the following range: - Minimum of 4,6 - Maximum of 15,7
ECLS-WAT-013	Water Management	Water management systems shall provide water for the following uses: - Drinking and food preparation - Hygiene - Medical uses - Washing Machines (clothers and dishes)
ECLS-WAS-014	Waste management	Wet waste shall be recycled to obtain clean water
ECLS-WAS-015	Waste management	Solid waste shall be stored
ECLS-FOOD-016	Food management	Food shall support human energy needs considering an energy consumption, in nominal condition, of: - 2975 kcal for men - 2440 kcal for women per day
ECLS-FOOD-017	Food management	Food shall cover an additional 500 kcal consumption for each EVA cycle
ECLS-FOOD-018	Food management	Protein calories shall represent 12-15% of total calories intake
ECLS-FOOD-019	Food management	Carbohydrates calories shall represent 50-55% of total calories intake
ECLS-FOOD-020	Food management	Fat calories calories shall represent 30-35% of total calories intake

Table 8.2-1 List of major requirements for ECLS system

8.3 Budgets

A first estimation of needs and consumables resupply has been carried out for the following ECLSS elements: Oxygen, Nitrogen, Water and Wet Waste

Data and reference values have been taken from references NASA sheets and similar projects [51, 52, 53, 34].
It is important to point out that references values have often been taken from analogous scenarios analyzed by past works and not yet tested in real missions; for this reason values can vary a lot depending on assumptions made by each study. In this study, values found in reference documents could have been tailored depending on different assumption made and different scenario considered.

8.3.1 Atmosphere Budgets

The objective of Atmosphere budget is to estimate the total mass of gases required both to set up habitat and to fill up leakage.
Table 8.3.1-1 shows Oxygen and Nitrogen masses required to set-up habitat in worst condition; it means that highest value of pressure and lowest value of temperature, among those indicated in requirements, have been considered to estimate the value.
Estimation process have been carried out in Appendix F.

Parameters	Symbols	Values
Mass of O_2 required for habitat	$M_{O2,HAB}$ [kg]	134.21
Mass of O_2 required for habitat	$M_{O2,AIRLOCK}$ [kg]	16.77
Total mass of O_2	$M_{O2,tot}$ [kg]	150.97

Table 8.3.1-1 Oxygen required to setup habitat module and airlock

Parameters	Symbols	Values
Mass of N_2 required for habitat	$M_{N2,HAB}$ [kg]	483.37
Mass of N_2 required for habitat	$M_{N2,AIRLOCK}$ [kg]	60.4
Total mass of N_2	$M_{N2,tot}$ [kg]	543.77

Table 8.3.1-2 Nitrogen required to setup habitat module and airlock

To carry out budgets concerning resupply of gases, it has been assumed that loss of atmosphere happens due to the following reason: daily air leakage, loss due to regeneration of air and loss due to EVAs. Air leakage has been assumed equal to maximum allowed by requirements (see Table 8.2-1) as well as loss from Airlock during EVAs. Concerning loss due to regeneration of air, it has been assumed an efficiency of recycling system of 10%, value is similar to one used by ESA/MIT/SoM in their work [4]. Finally, concerning loss during EVAs, it has been assumed a loss of pure oxygen of 0.15 kg/(hrs*CM) considering a maximum of 20 hrs of EVAs per crew member per week [34]. Detailed analysis is reported in Appendix F.

Parameters	Symbols	Values
Total oxygen loss per day	$M_{O2,LOSS}$ [kg/day]	2.897
Total oxygen loss per mission	$M_{O2,LOSS}$ [kg/mission]	521.53
Total oxygen loss per year	$M_{O2,LOSS}$ [kg/year]	1057.55

Table 8.3.1-3 Oxygen loss

Parameters	Symbols	Values
Total nitrogen loss per day	$M_{N2,LOSS}$ [kg/day]	3.192
Total nitrogen loss per mission	$M_{N2,LOSS}$ [kg/mission]	574.49
Total nitrogen loss per year	$M_{N2,LOSS}$ [kg/year]	1164.93

Table 8.3.1-3 Nitrogen loss

8.2.2 Water and wet waste budget

As wet waste is partially recycled, water and wet waste budgets shall be considered together to estimate the needed water for mission and size of tanks useful to store water both in clean form and in waste one.
Estimation of water needed is made using requirements reference which define the user needs, values

used are taken or tailored by bibliography [4, 34]. Detailed estimation is reported in Appendix G.
Water consumption per day per activity is reported in Table 8.3.2-1.

Parameters	Symbols	Values
Drinking and food water	$M_{H2O,D\&F}$ [kg/day]	11.7
Hygiene water	$M_{H2O,HYGIENE}$ [kg/day]	30.6
Machines washing water	$M_{H2O,WASH}$ [kg/day]	71.64
Medical water	$M_{H2O,MEDICAL}$ [kg/day]	7

Table 8.3.2-1 Water intake total budgets

Based on water consumption, it is possible to estimate also water needed per mission and per year. Water needed per mission indicate the initial value of water to set up the habitat.

Parameters	Symbols	Values
Total Water required per day	M_{H2O} [kg/day]	126.99
Total Water required per mission	M_{H2O} [kg/mission]	22857.66
Total Water required per year	M_{H2O} [kg/year]	46350.25

Table 8.3.2-2 Water intake total budgets

Concerning Wet waste, it is necessary to say that it exists two different kinds of that because different processes are carried out on these. While Dirty water considers urinal water and faces wet part, waste water considers all other outputs like hygiene and washing machine water as well as water from atmosphere.
Dirty water is first treated by Urinal processing Assembly (UPA) getting waste water and then waste water is also processed by Water Processor Assembly (WPA) getting clean water ready for use.
Estimation of both dirty and waste water is made in Table 8.3.2-3.
It is necessary to point out that recycling of atmosphere water and faces wet part is not considered in this preliminary study due lack of knowledge and data about it.

WET WASTE BUDGET		
DIRTY WATER		
Urinal Water	[kg/(day*CM)]	1.5
Flush urinal water	[kg/(day*CM)]	0.5
Total dirty water	[kg/day]	8
WASTE WATER		
Oral Water	[kg/(day*CM)]	0.36
Hand and Face wash water	[kg/(day*CM)]	4.08
Shower	[kg/(day*CM)]	2.72
Dish Washer water	[kg/(day*CM)]	5.44
Clothing Washer Water	[kg/(day*CM)]	12.47
Dirty water	[kg/day]	6.8
Total waste water	[kg/day]	112.43

Table 8.3.2-3 Wet Water budget as output of various systems

Finally it is possible to estimate loss of water due to recovery. Loss of water in this study have been considered only as effects of small UPA and WPA inefficiency. Following studies conducted on UPA [54] it has been decided to consider an efficiency of 85%. WPA efficiency, as there are not relevant studies available in bibliography, it has been assumed to be 90%, similar to the one used by ESA/MIT/SoM [4].

To estimate the total dirty water lost due to UPA and the total water lost due to WPA, it has been used the following equations:

$$M_{dirty\ water\ lost} = Total\ dirty\ water * (1 - \eta_{UPA})$$

$$M_{waste\ water\ lost} = Total\ waste\ water * (1 - \eta_{WPA})$$

Results are reported in Table 8.3.2-4.

LOSS OF WATER		
Loss of Dirty Water due to UPA	[kg/day]	1.2
Loss of water due to WPA	[kg/day]	11.24
Total Water Loss per day	[kg/day]	12.44
Total Water Loss per mission	[kg/mission]	2239.81
Total Water Loss per year	[kg/year]	817531.38

Table 8.3.2-4 Water Losses

8.4 Thermal Analysis

The preliminary thermal analysis has been carried out to estimate, given as input the temperature requirements and data about structure, the thermal power to be provided by heaters to maintain the required temperature inside habitat.
Analysis has been conducted taking as reference study performed by Peresotti [55] who design the thermal control system for a Lunar habitat; in this study analysis has been conducted developing Matlab script, details about methods and scripts are reported in Appendix H.
To develop the script it has been necessary to make assumptions and to define requirements:

- *Habitat Temperature*: As shown in requirements table (8.2-1) the required temperature of habitat is comprises between 18.5 °C and 29.6°C. To carry out the analysis it has been used an average value of 23 °C.

- *External temperature*: As reported by Benaroya [7] the estimate average temperature, supposed to remain constant, inside Lava Tube is roughly -15°C.

- *Habitat size and volume*: As described in chapter 4, the reference habitat is a cylindrical module with maximum heigh of 17 m and maximum internal diameter of 6.55 m. The total volume is 573.77 m^3.

- *Inflatable shell*: As described in chapter 4, habitat is separated from outside by an inflatable shell composed by several layers of different material. The total thickness of shell is 17.43 cm. Detail of thickness and thermal conductivity of material is reported in Appendix H.

- *Heat internal sources*: In this analysis it has been assumed negligible the heat generated by systems. Heat produced by human has been assumed to be 400 W per crew member [34].

Based on these assumptions it has been found the following results.

Parameters	Symbols	Values
Internal temperature	T_I [°C]	23
Temperature internal wall	$T_{WALL,I}$ [°C]	22.21

Temperature external wall	$T_{WALL,E}$ [°C]	-0.87
External temperature	T_E [°C]	-15
Thermal power to be produced	W_{heat} [kW]	1.03

Table 8.4-1 Results of Thermal analysis

Where:

- *Temperature internal wall*: It is the temperature of the most internal layer of shell.

- *Temperature external wall*: It is the temperature of the most external layer of shell.

- *Thermal power to be produced*: It is equal to the thermal power lost through walls.

Results show that thermal power required by heaters is not so huge; it thanks to the design of shell structure which well insulate whole habitat. In fact, using MLI it is possible to reduce the emissivity of habitat and so reduce the power emitted by structure. In addition to that, using a thick layer of foam, which has very low thermal conductivity, increase greatly the thermal insulation effect of the structure.

It is interesting to point out that, according to simulation conducted by ESA/MIT/SoM for their habitat in south pole, the amount of power rejected during lunar nigh can reach up to 17.6 kW which is much higher value than the Lava Tube one. As this thermal power is typically produced using electrical power, this fact increases remarkably the power consumption of habitat.
This fact shows the huge advantage of setting the habitat inside the Lava Tube.

8.5 ISRU

In situ resource utilization (ISRU) concerns all activities which have the purpose to extract and utilize material from soil, atmosphere (when it is present in other planets) or dry waste. ISRU is a key equipment to get more sustainable exploration mission because, producing consumables directly on surface, it is possible to reduce cargos mission and their costs, especially for future Mars mission.
ISRU equipment at the moment exists only as study concepts or testing devices (e.g MOXIE system for mars oxygen production). The main gaps in design of ISRU concerns the power production, as typically the power required is high, the relative low efficiency and the mass and volume required for this equipment.
Focusing on ISRU equipment for Moon habitat, it is possible to produce oxygen, water and other metallic resources.
In this study It has been decided to use ISRU only to produce oxygen. In fact, while it has been demonstrated a great presence of oxygen in Regolith minerals, especially Ilmenite, [1] in Marius region, there is no evidence of ice water presence near the Lava Tube.

8.5.1 Oxygen production process review

Concerning ISRU for oxygen production, state-of-arts considers three different process to obtain oxygen gathering and processing lunar Regolith: Hydrogen Reduction of Ilemite, Carbothermal Reduction of Silicates and Molten Regolith Electrolysis. Each of this process is characterized by different level of power require, mass and volume and efficiency (mass of oxygen produced per mass of Regolith processed).

Hydrogen Reduction of Ilemite

Hydrogen Reduction of Ilemite [56] process starts gathering Regolith from soil and putting it into a heating chamber. Here Regolith is heated at 900°C while a flow of hydrogen is sent to the chamber; hydrogen, due to high temperature, reacts with oxygen present on oxides forming water vapor. Then water vapor is made to condense to obtain liquid water. After that, water is separated into hydrogen and oxygen by means of electrolysis. Oxygen is stored into tanks and can be used to resupply habitat atmosphere, while hydrogen will be used again to produce new oxygen.
According to studies Ilemite represents roughly the 25% of minerals on Regolith; as Ilemite ($FeTiO_3$) is composed by 8-10% of oxygen, it is possible to conclude that efficiency is about 2.5% (up to 4% in mare regions).
This solution represents one of the easiest processes to extract oxygen but efficiency is very low and so mass, power and volume expect could be very high.

Carbothermal Reduction of Silicates

Carbothermal Reduction of Silicates [57] process starts gathering regolith from soil and putting it into the heating chamber. Here Regolith is heated up to melting point while a flow of methane is sent to chamber toward the melted zone. Due to the high temperature, it happens pyrolysis both on methane and regolith; carbon produced by pyrolysis reacts with silicates forming carbon monoxide and hydrogen. Monoxide and hydrogen react with a catalysis element (typically nichel) so that it is possible to obtain water and carbon. By means electrolysis, hydrogen and oxygen is produced. Finally, methane is produced using hydrogen and carbon.
According to studies, it is possible to produce up to 10-20% of oxygen from regolith. However, this process is more complex and brings more risks linked to the higher temperature used.

Molten Regolith Electrolysis

Molten Regolith Electrolysis [57] process start melting regolith. Using two electrodes put into the reacted where regolith is melt and providing a sufficient voltage, electrolysis of silicates happens and current is generated inside the melt zone. In this way, oxygen goes toward the anode and other metals go to cathode. This process has theoretically higher efficiency value than other one (about 40%), but studies suggest the real efficiency is a bit lower. However, it represents a very simple solution due to the absence of additional gas to perform the process (hydrogen or methane) and his simplicity. It is important to point out that this solution uses all kind of silicates and not only some minerals as it happens with Hydrogen reduction of Ilemite this means, this process can be performed in every part of the Moon without huge changes in efficiency of oxygen production.

8.5.2 Preliminary design process of ISRU system to produce oxygen

In this study it has been decided to consider Molten Regolith Electrolysis (MRE) process to extract oxygen from the Lunar soil; decision has been made both thanks to system simplicity and for the high efficiency value.
This work has not the aim to design new prototype of MRE system, for this reason it has been decided to use as reference system model design developed by S.S Schreiner [57].
MRE model developed by Schreiner comprises the following equipment: structure, power system, reactor, storage unit and excavator units.

In his work, Schreiner developed two parametric equations which allow, given as input the oxygen generation rate [kg/year] the estimate value of mass and power consumption. Equations are reported below:

$$\mathrm{M} = 5.16[\pm 0.94]\mathrm{N}^{0.622[\pm 0.021]} \qquad (equation\ for\ mass\ estimation)$$

Where:

- M=Total mass of ISRU Plant. Expressed in [kg].
- N=Oxygen production rate. Expressed in [kg/year].
- Value in square backets represents uncertainty range.

$$\mathrm{P} = 0.253[\pm 0.033]\mathrm{N}^{0.588[\pm 0.015]} \qquad (equation\ for\ power\ estimation)$$

Where:

- P=Total power consumption of ISRU Plant. Expressed in [kW].
- N=Oxygen production rate. Expressed in [kg/year].
- Value in square backets represents uncertainty range.

Concerning power system, Schreiner's work considers a dedicate electrical power system, that means power is self generated by ISRU plant and it does not affect the habitat power budget. Schreiner suggests that every kind of power generation technology can be used to feed MRE system; in his work, he considered a solar panels solution for power generation.

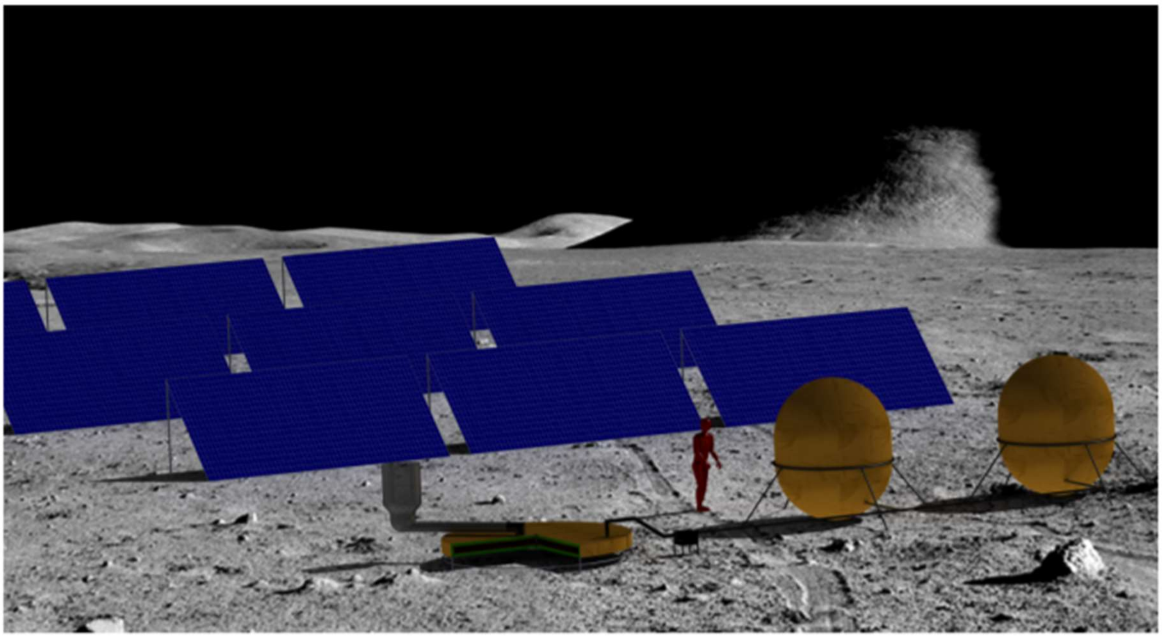


Fig. 8.5.2-1 CAD model of ISRU Plant developed by S.S. Schreiner [57]

Starting from the oxygen loss rate computed in previous sections, the following preliminary outputs of ISRU have been found.

Parameters	Symbols	Values
Oxygen loss per day	m_{O2} [kg/day]	2.897
Oxygen loss per year	N [kg/year]	1057.55
Total mass of ISRU plant	M_{ISRU} [kg]	392.46
ISRU Plant Power consumption	P_{ISRU} [kW]	15.18

Table 8.5.2-1 Resume of ISRU Plant requirements and specification

8.6 Conclusion

Environmental and Life support is a fundamental system to allow exploration mission both on surface and in orbit; without an ECLS system it is not possible for human to live due to lack of essential life conditions.
Up to now, ECLS systems have been developed and tested only for in orbit spacecraft as ISS or transportation capsule but no ECLS for surface habitat have been really built and tested in real environment. In addition, Environmental life and support system depends strongly on the mission assumptions and scenario, as increasing the number of crew members it will be necessary to consider higher consumables and more powerful systems.
One of the most challenging gaps in ECLS system design concern recycle and independence; in future mission on moon and Mars, due to distances, it will not be possible (or it will be possible but really expensive) to bring consumable as oxygen, water or food from Earth. It is necessary to design new ECLS system so that habitat will be independent by Earth and can produce themselves oxygen, water and food.
The aim of this chapter was to provide an overview of ECLS system designed to be partially independent from Earth. It was also studied the thermal problem sizing the heater; concerning thermal system it was shown the big advantage, in terms of electrical power required, to build an habitat inside a Lava Tube.
Finally, it was given an overview on In-situ Resource utilization system to extract oxygen from Regolith.
As result of this analysis it has been understood that the current technologies are not sufficient to provide life support capabilities, it is necessary to design new assemblies adapt for partial gravity in habitat surface.
Concerning the challenges still open we have; food production, water extraction from Regolith and smart habitat control.
All these function will be studied in future works.

9. Conclusions

In the following years space agencies and community around the world plan to bring again human on the Moon; starting from that, the next steps will be a first human settlement on lunar soil and following human settlement and exploration of Mars and near asteroids.
In this context, study and development of habitability solutions on Moon and Mars surface is fundamental to allow human exploration and settlement; several studies have been conducted to highlight the most critical gaps in habitability on Moon and to implement adequate solutions.
Some of the most important gaps identified by previous studies are: power generation, communication, mobility on Moon surface, In-situ resources utilization, radiation protection, meteorites protection, etc…
This study has the aim to make a review of state-of-arts technology used in Moon habitat design and to implement these in a very particular scenario, Moon Lava Tubes.
Moon Lava Tubes, discovered in equatorial region of Moon by SELENE mission, represents a very convenient scenario where to set up an habitat; in fact, it offers a free protection from both radiation and meteorites impact as well as a temperature stable environment. However, Lava Tubes pose some additional challenges as the accessibility into deployment site from outside and the risk of rock falls or terrain instability.
This study gave a general overview on all needed system architecture with a more in depth analysis on some systems. To solve the weight and launch costs of structure it has been designed an inflatable structure which can be launched using state-of-arts rocket and deployed directly inside the Lava Tube. Power generation gaps has been solved using nuclear reactors technologies, which represents the less heavy and more compact solution for the high power consumption of habitat.
Communication has been solved designed a dedicated communication plant outside the cave and an additional relay satellites constellation in low lunar orbit, which makes possible to communicate with Earth and other moon bases using the S-band. What is more, real time communication with Earth is also possible using X-band link.
Finally life support gaps have been solved implementing a closed-loop system which recycle quite the 100% of air and water; re filling products can be send from Earth using cargo mission or obtained in loco using ISRU (oxygen).
Although the study has shown that it is possible to build a lunar habitat inside the lava tubes consistently with the current state of the art, some ultra-high-tech technologies are still needed. Nuclear reactors are still a new technology never used in real Moon environment and it needs further studies to full qualify their use as well as studies to quantify the radiation problems. Communication architecture requires a dense network of relay satellites to ensure real time communication with all part of the Moon and Earth; this study is still in progress.
Life supports units exist only for in orbit application, it is necessary to design and qualify new life supports devices for Moon surface application.
In addition to these technology gaps, other challenges shall be face up; accessibility on Lava tube is still a relevant problem to be solved in future studies and mobility&navigation on Moon surface are still in development.
Futures works will deepen other main functions and system identified in this work and will try to solve the remaining gaps. Additional analysis and studies will be needed to better characterize the system architecture.

Appendix A: Iterative process for habitat volume estimation

As explained in Chapter 4, estimation of internal volume depends on four contributes: habitable volume, system volume, accommodation volume and structure volume. While the first three contributes can easily been estimated as it has been done there, structure volume cannot be determined independently. In fact, it depends on both the shape of structure and on size of habitat.
The problem is that to determinate the sie of habitat it is necessary to know as input the total internal value. It is clear that we should face up an iterative process which will bring to define both the structure volume and the total internal volume of habitat, as well as the reference internal diameter.
In this appendix first it will be recalled assumptions made in chapter 4, then it will be explained the iterative process and implemented.

A.1 Assumptions

The starting point to implement the iterative process is to recall all assumption made; the assumed volume we known are resumed in table below.

Parameters	Symbols	Values
Habitable volume	V_{HAB} [m^3]	400
System volume	V_{SYS} [m^3]	40
Accommodation volume	V_{ACC} [m^3]	73
Structure volume	V_{STR} [m^3]	51.3

Table A.1-1 Resume of habitat volumes

Where:

- *Habitable volume:* It is the volume where astronauts can live free from systems, equipment and structure. It has been assumed to use 100^3 per crew members (4 astronauts per mission).

- System volume: It includes all life support equipment installed inside the habitat. As precise values are not available, it has been assumed it is the 10% of habitable volume.

- *Accommodation volume:* It is the volume of all equipment considered as for accommodation functions. Value has been determined in chapter 5.

- *Structure volume:* It includes all volume occupied by secondary structure. At the beginning the shape of structure and size of habitat were known, for this reason it has been assumed the 10% of the sum of other three volume.

The total volume estimated is 564.3 m^3.

Another assumption to be made concerns the shape of secondary structure whose CAD is shown in Fig. A.1-1.

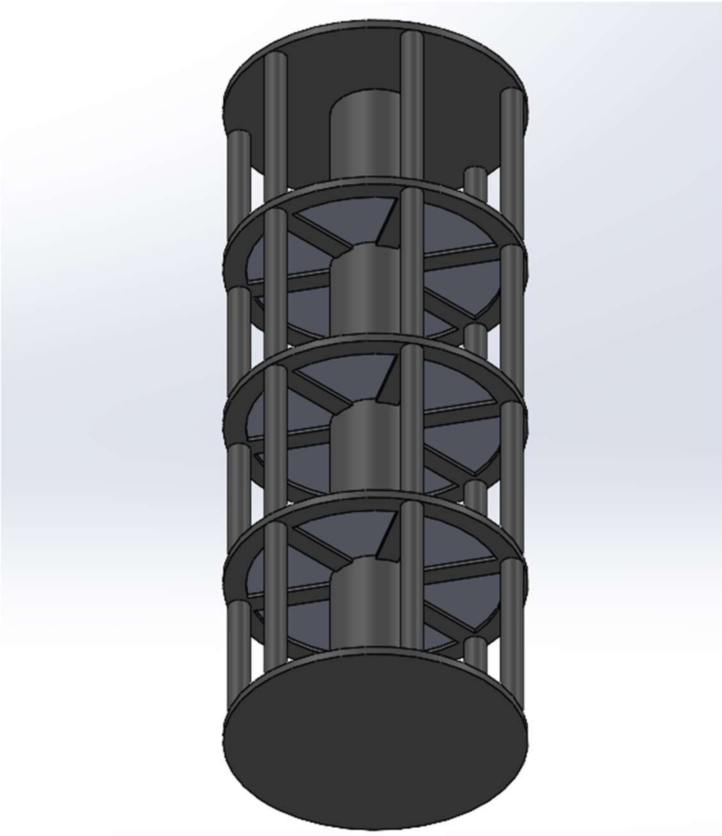


Fig A.1-1 Secondary Structure of habitable module

Secondary structure is composed by the following elements:

- *Longerons:* There are six longerons disposed along the external diameter to withstand the vertical loads. Longerons have cylindrical shape and they are characterized by an heigh of full habitat and diameter of 0.5 m. Material used is aluminium.

- *Central structure:* The central structure is a cave cylinder structure which supports longerons in withstand the vertical loads. Central structure has the same heigh as the habitat, internal diameter of 1.5 m and thickness of 0.25 m (external diameter of 2 m). Central structure is made in aluminium.

- *Floors and roof:* Habitat has four floors and one roof. Both floors and roof has a thickness of 0.2 m and the same diameter as the habitat. Technically floors are made by a composite layer of 10 cm and an aluminium structure of 10 cm too which connects longerons with central structure. For simplicity we will assume floor are full made on aluminium. It represents a conservative assumption from a weight point of view.

Final assumptions concern the habitat configuration. For the first estimation of volume it has been assumed the habitat has an height of 12.8 m and a total of 4 floors.
Once the final configuration will be chosen, it will be possible to repeat procedure with the new values of height and floors.

A.2 Structure volume mathematical volume

Structure volume can be seen as the sum of the three elements mentioned before.

$$V_{STR} = 6 * V_{LONG} + n * V_{FLOORS} + V_{CS} \; [A.2.1]$$

Where:

- V_{STR} = Total volume of structure [m^3]
- V_{LONG}= Volume of single longeron [m^3]
- V_{FLOOR}=Volume single floor [m^3]
- V_{CS}= Volume central structure [m^3]
- n= Number of floors +1 (because of roof)

Volume of longerons can be expressed considering a simple cylinder with habitat height and fixed diameter.

$$V_{LONG} = \left(\pi * \frac{D_{LONG}^2}{4}\right) * h \;\; [A.2.2]$$

Where:

- D_{LONG}= Diameter of longeron (assumed 0.5 m)
- h= Habitat height (assumed 17 m)

Volume of floor can be expressed as the volume of a cylinder with height equal to thickness and the same diameter as the habitat.

$$V_{FLOOR} = \pi * \left(\frac{D_{HAB}^2}{4} - \frac{D_{CS,e}^2}{4}\right) * t_{FLOOR} \;\; [A.2.3]$$

Where:

- D_{HAB}= Diameter of habitat (variable)
- $D_{CS,e}$= External diameter of central structure (assumed 2 m)
- t_{FLOOR}= Thickness of floor (assumed 0.2 m)

Volume of central structure can be expressed as the volume of a simple cave cylinder with the same height as the habitat.

$$V_{CS} = \pi * \left(\frac{D_{CS,e}^2}{4} - \frac{D_{cs,i}^2}{4}\right) * h \; [A.2.4]$$

Where:

- $D_{CS,e}$= External diameter of central structure (assumed 2 m)
- $D_{CS,i}$= Internal diameter of central structure (assumed 1.5 m)
- h= Habitat height (assumed 17 m)

Combining all equations above we obtain:

$$V_{STR} = 6 * \left(\pi * \frac{D_{LONG}^2}{4}\right) * h \; + 5 * \pi * \left(\frac{D_{HAB}^2}{4} - \frac{D_{CS,e}^2}{4}\right) * t_{FLOOR} \; + \pi * \left(\frac{D_{CS,e}^2}{4} - \frac{D_{cs,i}^2}{4}\right) * h \; [A.2.5]$$

A.3 Implementation of iterative process and Excel sheet description

Once the structure model has been defined, it is possible to implement in Excel the iterative process. Iterative process flow is really simple. Starting with a first estimation and volume, known the height of habitat, it is possible to find the diameter using the following equation:

$$D_i = 2 * \sqrt{\frac{Vol}{\pi * h}} \quad [A.3.1]$$

Found the diameter it is possible to estimate the structure volume using equation A.2.5. After that, known other volumes, it is possible to estimate the new total interna volume which will be the starting point for the next adept of process.

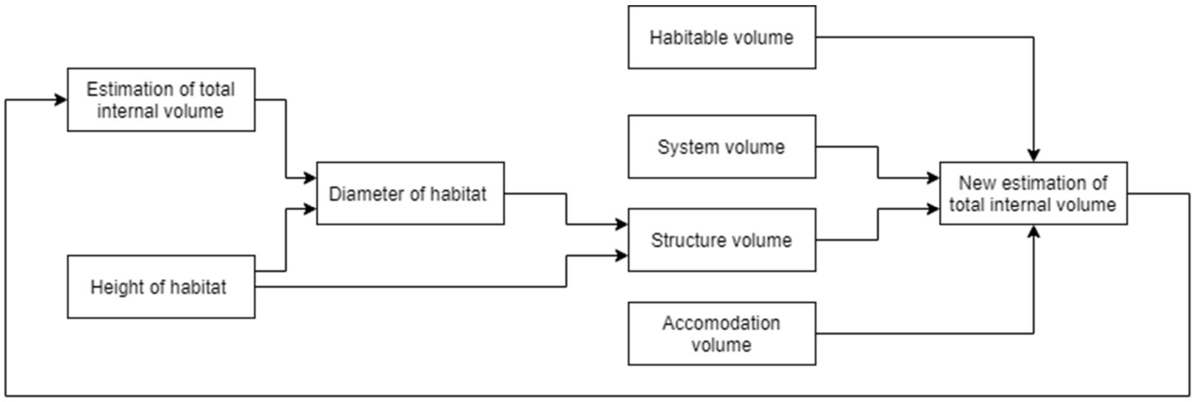


Fig. A.3-1 Iterative process for internal volume estimation

The first step to implement the process has been defining the assumptions. An image og assumption section of file Excel is shown is figure above where cells in heavenly indicate values that can be modified because they depend on assumption made about habitat configuration. Yellow cells can be modified is the secondary structure is modified too.

DATA AND ASSUMPTIONS		
MISSION		
Crew members	[CM]	4
Habitable Volume per crew member	[m^3/CM]	100
INITIAL ASSUMPTIONS		
Number of floors	[fl]	3
System Volume	[m^3]	40
Accomodation volume	[m^3]	73
Height	[m]	12,8
STRUCTURE ASSUMPTIONS		
Thickness of floor	[m]	0,2
Radius of longerons	[m]	0,25
Number of longerons		6
Number of flooring		4
Internal radius of central body	[m]	0,75
External radius of central body	[m]	1

Fig. A.3-2 Assumptions section on Excel sheet

In addition to assumption section, excel file comprise also the computational section shown in figure below.

ITERATION TO DETERMINATE FINAL VOLUME						
Internal Volume [m^3]	Height [m]	Diameter [m]	Accomodation Volume [m^3]	Habitable Volume [m^3]	Structure Volume [m^3]	System Volume [m^3]
564,000	12,8	7,49013	73	400	55,61283	40
568,613	12,8	7,52070	73	400	55,90113	40
568,901	12,8	7,52261	73	400	55,91915	40
568,919	12,8	7,52273	73	400	55,92028	40
568,920	12,8	7,52274	73	400	55,92035	40
568,920	12,8	7,52274	73	400	55,92035	40
568,920	12,8	7,52	73	400	55,92035	40

Fig. A.3-3 Computational section of excel file

This section is run up according to the procedure describes above.

A.4 Results and consideration

As it can be seen in Fig. A.3-3, the process converges quickly reaching a stable value (error lower than 0.1%) in just one step and an error lower than 0.0001% in 4 steps. This happens because we were lucky and estimated well the initial value of structure volume using the 10% of sum of other volumes. In general, more steps could be needed depending on the first estimation.
Finally, Fig. A.4-1 show the result after that final configuration has been chosen (4 floors and 17 m height).

ITERATION TO DETERMINATE FINAL VOLUME						
Internal Volume [m^3]	Height [m]	Diameter [m]	Accomodation Volume [m^3]	Habitable Volume [m^3]	Structure Volume [m^3]	System Volume [m^3]
564,000	17	6,49935	73	400	60,54898	40
573,549	17	6,55414	73	400	61,11068	40
574,111	17	6,55735	73	400	61,14373	40
574,144	17	6,55754	73	400	61,14567	40
574,146	17	6,55755	73	400	61,14578	40
574,146	17	6,55755	73	400	61,14579	40
574,146	17	6,56	73	400	61,14579	40

Fig. A.4-1 iteration process for final habitat configuration

Appendix B: Launcher analysis for habitat structure

In this appendix it will be analysed the habitat launch possibilities using the reference launcher identified in chapter 2.

B.1 Assumptions

Launchers identified as reference for this study have been introduced in chapter 2. The main requirements they impose to habitat size concerns the maximum height and diameter.
In the following table requirements have been recalled.

	Maximum Heigh [m]	Maximum Diameter [m]
SLS BLOCK 1	19.1 (TBC)	4.6
SLS BLOCK 1B – FAIRING 1	9.85	7.5
SLS BLOCK 1B – FAIRING 2	18.18	7.5
SLS BLOCK 2 – FAIRING 1	9.85	9.1
SLS BLOCK 2 – FAIRING 2	11.8	9.1
SPACEX - STARSHIP	8	8

Table B.1-1 Resume of reference launch options requirements for this work

Another assumption made concerns the initial estimated value for total internal volume. As described in chapter 4 and in appendix A, the initial assumed value is:

$$V_i = 568.92\ m^3$$

This value as been found after having applied the iterative process using the following assumptions:

- 3 Floors
- Height: 12.8 m

B.2 Launch feasibility

In order to verify if a rocket can host the habitat it has been implemented the following method.
First it has been computed the diameter that the habitat would have if it had the maximum height imposed by the habitat.

$$D_i = 2 * \sqrt{\frac{V_i}{h_{launcher} * \pi}}$$

Then the value found have been compared with the maximum diameter requirement imposed by the launcher; if the value found is greater than requirement, launcher cannot be used to launch habitat. Otherwise, launcher option ca be used.
Analysis have been conducted building an excel sheet. Details about results are reported below.

Launcher	Max. Height [m]	Base area [m^2]	Habitat diameter [m]	Requirements [m]
SLS BLOCK 1	19,1	29,786	6,158	4,6
SLS BLOCK 1B – FAIRING 1	9,85	57,758	8,576	7,5
SLS BLOCK 1B – FAIRING 2	18,18	31,294	6,312	7,5
SLS BLOCK 2 – FAIRING 1	9,85	57,758	8,576	9,1
SLS BLOCK 2 – FAIRING 2	11,8	48,214	7,835	9,1
SPACEX - STARSHIP	8	71,115	9,516	8

Fig. B.2-1 Results of analysis

According to results only three launch options are available.

Appendix C: Solar power plant design

The solar panels preliminary design has the aim to provide, given as input the required power to be produced, as outputs: area of solar panels and total mass of whole plant.
In this appendix it will be shown the whole process, proposed by Larson and Wertz [38], to carry out this estimation; process has been implemented by creating an Excel.
First it will be introduced all inputs and assumptions made in this work, then it will be shown and explained step by step the whole analysis process, as well as the Excel file and finally it will be shown the outputs.

C.1 Assumptions

The first data to be defined is the power consumption. As described in chapter 6, power consumption shall take into account all power needed by habitat and systems plus the power required by electrolyser to convert water into oxygen and hydrogen, elements needed by RFC to generate power during lunar night.
The power required by electrolyser has been found in ANNEX D.
Table C-1.1 resume average power consumption of all systems.

User	Symbols	Values
Habitat	P_{HAB} [kW]	60
Communication Plant	P_{COM} [kW]	2
Airlock	$P_{AIRLOCK}$ [kW]	4

Table C-1-1 Resume of power consumption by each user

The total power required, considering a margin of 15% is:

$$\boldsymbol{P_{SA} = 181{,}87\ kW}$$

Power generated by solar panels depends strongly by scenario features, for this reason, it is needed to define some reference values to carry out correctly analysis.

Concerning temperature, several studies [5, 58] conduced estimated a maximum temperature at the equator region of 400 K (or roughly 120-150 °C). As the more is the temperature the less efficient is the solar cell, it has been decided to use it as the worst case value for this analysis.

$$\boldsymbol{T_{op} = 400K\ (or\ 127\ °C)}$$

Concerning Solar irradiance Constant it has been decided to use the value used for GEO satellites orbiting around Earth, 1367 W/m^2; this because, due to absence of a relevant atmosphere on Moon, a great part of solar rays can reach the lunar surface without being blocked by atmosphere. This represents in good approximation the same deep space condition of far satellites as GEO or HEO.

$$\emptyset_{sun} = \boldsymbol{1367\frac{W}{m^2}}$$

Incidence angle, which is the angle between the solar cell plane and the direction of Sun radiation, represents a huge open point in this study, due to the small angle of Moon axes to the respect of Ecliptic plane (1.54°), Sun varies his incidence very little and the maximum angle of incidence experienced can be set as the maximum angle of Moon axes. This assumption does not take into account the conformation of soil, for this reason further studies shall be carried out in future.

$$\boldsymbol{\theta \approx 1.54°}$$

The last aspect to be analysed concerns the degradation of solar cells during years. Degradation on Moon surface is mainly caused by two reasons: Regolith dusk and Solar Radiation. Due to magnetic properties of Regolith and his s sharp shape, It tends to adhere very well on solar panels reducing the amount of solar radiation gathered. To reduce this effect, some studies suggest raising solar panels plant of 2 meters above surface soil [35] or carry out periodic maintenance during EVAs activities to clean the solar cells. Nowadays, the newest solar cells are designed with thin layer of protective material which allow to reduce solar radiation damage and improving degradation constant. In this study It has been used the following value, commonly used in similar studies and assuming to use GaAs cells, 2.75% per year. [38]

$$\boldsymbol{Degradation\ value\ per\ year = 2.75\%}$$

Other data needed to carry out preliminary analysis concerns solar cells performance. In this study it has been decided to use GaAs cells due to the high efficiency and the not much high cost. The reference model of cells is the AzurSpace TJ Solar Cell 3G30C model [39].

Definition	Symbols	Values
Efficiency at BOL	η_o	29.5%
Area of each cell	A_{cell} [m^2]	0.003018
Average weight	m_{cell} [kg/m^2]	0.86
Voltage at maximum power	V_{cell} [V]	1.44
Current at maximum power	I_{cell} [mA]	518

Table C.1-2 Data of GaAs cells

Other assumptions concern the Mass breakdown. Mass of whole plant is the sum of masses of each unit; these are:

- *Solar cells blanket mass*: It is the mass of only the solar cells without supporting structure or other units.
- *Structure and moving unit mass:* it is the mass of whole the infrastructure needed to sustain the solar cells and to move it towards the Sun direction.
- *Thermal control unit mass*: It is the mass of dedicated unit which will maintain the solar cells within the operating temperature.
- *Electrical equipment mass*: It is the mass of unit which will control and regulate the output power from solar cells before the transportation to the habitat.
- *Integrating equipment mass*: It is the mass of all needed equipment to integrate and connect solar panels.

As in bibliography dedicated parametric studies to estimate unit masses are not available, it has been decided to use ISS value as reference to roughly estimate these masses.

Specific Mass	Symbol	Values
Structure and gimbal specific mass	m_{STR} [kg/kW]	31.07
Thermal control specific mass	m_{THC} [kg/kW]	26.07
Electrical equipment specific mass	m_{ELE} [kg/kW]	21.79
Integration equipment specific mass	m_{INT} [kg/kW]	21.79

Table C.1-3 Resume of Specific mass on ISS

Specific masses are expressed as the ration between mass and power produced so that the value are independent of the solar panels area.

C.2 Preliminary sizing

C.2.1 Area of solar cells

The first step to estimate the solar panels area is finding the output power produced by each meter-square of solar cells, at the End-of-Life and under the defined environmental condition.
Normally the power output of solar cells is:

$$P_{out} = P_{in} * \eta$$

Where:

- P_{in}= Power received by solar cell, which is basically the solar constant (ϕ_{sun})
- η=Efficiency of solar cells

Efficiency take into account the fact that part of solar radiation is scattered or reflected and it is not absorbed by material; in addition, part of solar radiation can not be gathered because some wavelengths can not generate power. For this reason currently efficiency are not so high (typically from 10 up to 45 %, depending of the type of material and technology used).
Efficiency decreased when temperature increase, using the relation (20-3) of Wertz [40], it is possible to estimate the real efficiency at the assumed operating temperature.

$$\eta = \eta_0 * \left(1 + T_c * (T - T_0)\right) = 0.228$$

Where:

- η_0 =Efficiency at standard temperature (T_0)
- T_c = normalized temperature coefficient (for GaAs cells is roughly -0.0023/C°)
- T = Operating temperature
- T_0 = Reference temperature (=28°C)

Output power represents the maximum value which can be provided by solar cells; this value does not take into account corrections which reduce the real output power. In particular:

- Sun incidence: When the solar direction and normal to cells surface are not parallel, the among of solar radiation which reaches the cells surface is reduced.
- Design and assembly: According to test, solar array are less efficient than single cells by about 10%.
- Shadowing of cells: This effect is typically caused by bad assembly and positioning of solar panels on power site. In this study it has been assumed there is not negative effect of shadowing.

- Packing: This effect depends on the fact that usually cells does not cover the whole surface area required; typically it depends on both shape of cells and the way they are assembled. Typically value are roughly 80-85 %.
- Cable loss: It is caused by Joule effect on transportation cable between habitat and power plant. A rough estimation is made below.

Considering all these effects, the power at the beginning of life is:

$$P_{BOL} = P_{out} * I_d * \cos(\theta) * \eta_{cable}$$

Where:

- I_d=Inherent degradation: it gathers the assembly, shadowing and packing effects together. We supposed to use a compressive value of 0.77.
- Θ=Angle between solar cell normal and sun radiation direction
- η_{cables}= Efficiency of cables transmission

The cable efficiency can be found using the relation (20-11) of Werzt manual [40]:

$$\frac{P_{loss}}{P} = \frac{P * R}{V^2}$$

Where:

- P= Power produced by solar plant
- R=Resistence of cables
- V= Transportation voltage (assumed to be 120 V)

In this analysis it has been supposed to use a copper cable with features reported in table below:

Parameter	Symbols	Values
Length of cable	L [m]	2000
Cable section size	A [cm^2]	200
Standard condition resistivity	ρ_0 [ohm*m^2/m]	1.68 *10^8
Coefficient of resistivity	α [1/C°]	0.0043

Table C-2.2 Resume of Power transportation cables

Cable efficiency found is roughly 96%.

The final step to estimate the end-of-life power provided require to consider the degradation factor of solar cells over time. As said in the previous chapter, degradation on the moon is caused mainly by both the solar radiation damage and the lunar dust in addition to others factors as thermal cycles, meteorite strikes, etc… Power end-of-life is:

$$P_{EOL} = P_{BOL} * L_d$$

Where:

- L_d= Degradation factor

Degradation factor can be found using the relation below:

$$L_d = \left(1 - \frac{degradation}{year[\%]}\right)^{lifetime}$$

Where:

- Degradation/year: The value used often for GaAs cells is 2.75%

- Lifetime: It is the whole duration of mission. In our case 5 years.

Once output power produced by solar panels has been found, it is possible to estimate the required solar cells area needed to cover the power consumption.

$$A_{SP} = \frac{P_{SP}}{P_{EOL}}$$

In out case it has been found a required **area of 793.40 m^2**.

C.2.2 Masses estimation

Solar cells blanket mass can be found, known the required are, using equation below:

$$M_{SC} = A_{SP} * m_{cell}$$

Where:

- A_{SP}= Area of required solar panels.
- m_{cell}= Specific mass of solar cells [kg/m^2]

Mass estimation for additional units is carrying out using the reference specific mass values found for ISS [40].

$$M_{STC} = m_{stc} * P_{SP}$$

$$M_{THC} = m_{thc} * P_{SP}$$

$$M_{ELE} = m_{ele} * P_{SP}$$

$$M_{INT} = m_{int} * P_{SP}$$

The final total mass is the sum of all masses:

$$M_{tot} = M_{SC} + M_{STC} + M_{THC} + M_{ELE} + M_{INT}$$

Appendix D: Regenerative Fuel cells preliminary sizing

The regenerative fuel cells preliminary study has the aim to provide an estimation of both mass and volumes required to set up the RFC plant.
In this annex it will be shown the whole preliminary sizing process using data and handbooks available on bibliography. The process has been implemented using an Excel sheet.
First it will be given a general overview on RFC technology. Then it will be obtained the general equation to size PEMFC fuel masses. After that, it will be carried out the fuel analysis to estimate the masses and volumes of oxygen and hydrogen. Then it will be carried out the tank analysis to estimate masses and volumes of tanks.

D.1 Regenerative Fuel Cell description

Fuel cells is a useful technology which can generate electrical power in every light condition, both sunlight and eclipse. If it is added an electrolyser which can retransform water, output of fuel cells, into hydrogen and oxygen it is possible to define this system as Regenerative fuel cells. In this work it has been decided to use RFC only during eclipse period to make up for the impossibility of using solar energy.

The basic idea below the fuel cells is the conversion of chemical energy into electrical one by using electrochemical reaction. Considering a single fuel cell, it is composed by three parts: anode, cathode and an electrolyte membrane which separates and allow only the passage on hydrogen ions.
A continuous flow of pure hydrogen is sent to the anode, here hydrogen, thanks also to the catalyst, lose one electron per atom, getting a ion. While hydrogen positive ions can flow through the electrolytic membrane, electron are forced to pass through an external circuit that connects the two electrodes generating an electrical current. At cathode a continuous flow of oxygen arrives, here, oxygen combines with hydrogen ions and electrons generating as waste, only water (and oxygen or hydrogen which has not reacted).

Chemical reactions at anode and cathode are:

$$H_2 \rightarrow H_2^{++} + 2e \quad (at\ Anode)$$

$$\frac{1}{2}O_2 + H_2^{++} + 2e \rightarrow H_2O \ \ (at\ Cathode)$$

The overall reaction is:

$$H_2 + \frac{1}{2}O_2 \ \rightarrow H_2O + heat + electrical\ power$$

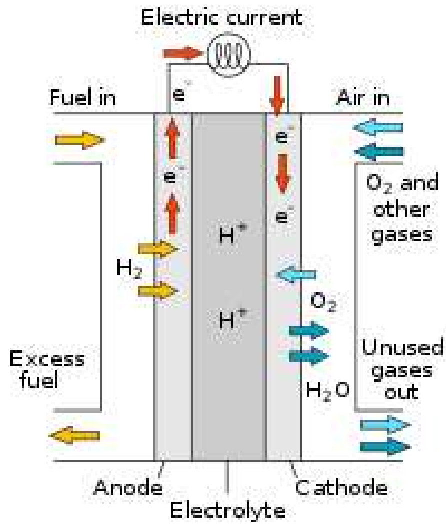


Fig. D.1-1 Detail of single fuel cell functioning (credit Wikipedia.org)

Typically fuel cells are classified depending on the electrolytic membrane and the fuel used. As the purpose of this work is not to describe all classes of fuel cells, it will be selected only one type and briefly described.
in out case it will be used a Proton exchange membrane fuel cell (PEMFC) which simply use a Teflon membrane to allow the passage of hydrogen ions and isolate electrons. This has different advantages: low operating temperature (roughly 50-250 °C), a quick start and an high energy density.
PEMFC use hydrogen and oxygen as fuel and so reaction are perfectly described above.
A single fuel cell is able to provide a voltage of 0.6-0.7 V; this value depends on current and specifically it decreases as the current increases due to some loss of voltage phenomena.

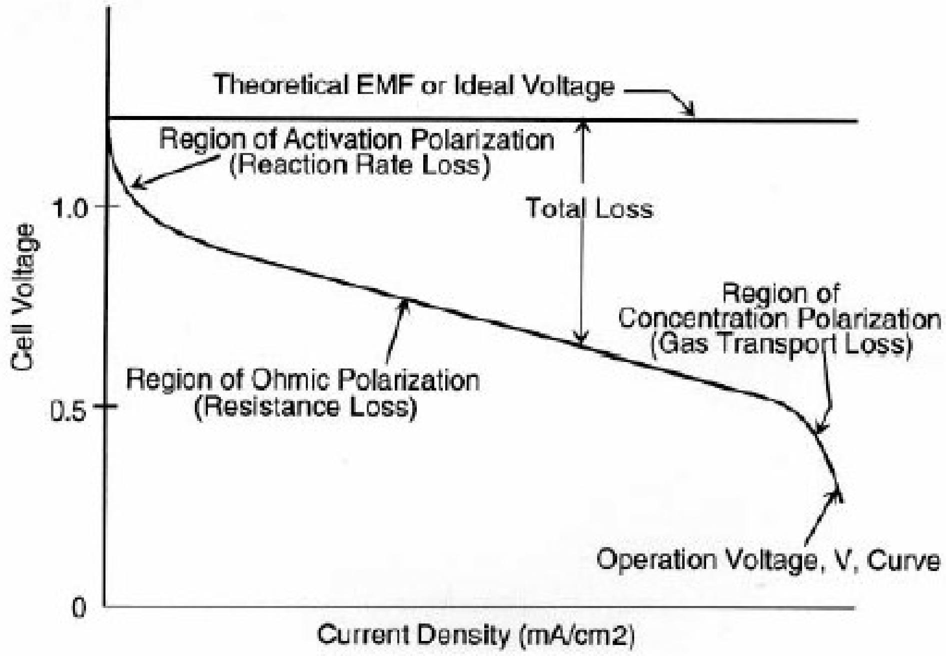


Fig D.1-2 Fuel cell polarization curve (credit by Xinhong Huang [RD D1]

If the cells are built to carry out the reverse process, producing oxygen and hydrogen, starting from water and electrical power, system can be defined as RFC. Typically RFC have a lower roundtrip efficiency than the simple FC due to the need of perform two different transformation (from oxygen and hydrogen to water and viceversa). According to state of art, efficiency of singles PEMFCs is roughly 45-55% which decrease up to 40% for RFC.

D.2 Fuel cells model

Before carrying out mass and volume estimation, it is necessary to introduce the general fuel cells equation which allow to estimate the number of reacting moles for both fuel reactant. General equations are taken from by Andrew L. Dicks et al. [59].

First it is necessary to recall the overall fuel cells equation valid for PEMFC:

$$2H_2 + O_2 \ \rightarrow 2H_2O$$

As it can be seen on reaction, two moles of hydrogen react with one mole of oxygen providing two moles of water. In this process, one electron for each atom of hydrogen is exchanged, so considering two moles of electrons are exchanged for each mole of hydrogen.

$$quantity\ of\ charge\ exchanged\ (Q) = 2 * F * N_{H_2}$$

Where:

- F= Faraday constant= 96485.33 C/mol
- N_{H2}=Total number of hydrogen moles

Dividing by time we have:

$$\frac{charge}{time} = Intensity\ of\ current = 2 * F * \dot{N}_{H_2}$$

As we are considering the moles of hydrogen of whole stack, we shall multiply this value for the number of cells (n):

$$\dot{N}_{H_2} = \frac{I * n}{2 * F}$$

Now we can express intensity of current, know the overall power provided by stack and voltage of the single cell:

$$P = I * V_C * n$$

Where:

- P=Power produced by whole stack
- V_C= Voltage of single cell. We assumed it to be 0.7 V
- n= number of cells

The moles flow rate of hydrogen can be expressed, combining the two relations above:

$$\dot{N}_{H_2} = \frac{P}{2 * F * V_C} = 7.403 * 10^{-6} * P\ \frac{mol}{s}$$

To find the total number of moles required it is necessary to multiply the mole flow rate with the total operating time (t_{op}) of FC. And so:

$$\boldsymbol{N_{H_2} = 7.403 * 10^{-6} * P * t_{op}} \qquad \boldsymbol{(D.2.1)}$$

As for each mole of hydrogen half mole of oxygen reacts, the number of oxygen moles will be the half of hydrogen moles.

$$\boldsymbol{N_{O_2} = 3.715 * 10^{-6} * P * t_{op}} \qquad \boldsymbol{(D.2.2)}$$

As for each mole of oxygen it is produce two moles of water, the number of water moles will be the same as the hydrogen.

D.3 Mass and volume estimation

D.3.1 Fuel masses and volume analysis

It is necessary to recall power requirements to estimate the total oxygen and hydrogen number of moles. As fuel cells shall produce only power for users and not also for electrolyzer, as it happens for solar panels, the power consumption is:

$$P_{FC} = 76.36\ kW$$

Which already take into account a 15% of margin.

Fuel cells shall produce power continuously for whole the eclipse period; as the night on the moon lasts roughly 14.82 Earth days, it means:

$$t_{op} = 14.82\ days * 24 \frac{hr}{day} * 3600 \frac{s}{hr} = 1280448\ s$$

Known the power required and the operation duration, it is possible to estimate the total number of moles for both fuels using equations (D.2.1 and D.2.2).

Obtained the moles of hydrogen, oxygen and water, it is possible to estimate masses using the molar masses:

$$\boldsymbol{m_{H_2}} = \boldsymbol{N_{H_2}} * \mathrm{M}_{\boldsymbol{H_2}}$$

$$\boldsymbol{m_{O_2}} = \boldsymbol{N_{H_2}} * \mathrm{M}_{\boldsymbol{H_2}}$$

$$\boldsymbol{m_{H_2O}} = \boldsymbol{N_{H_2O}} * \mathrm{M}_{\boldsymbol{H_2O}}$$

Where:

- M_{H2}=Molar mass of hydrogen=0.00202 kg/mol
- M_{O2}=Molar mass of oxygen=0.031998 kg/mol
- M_{H2O}=Molar mass of water=0.01802 kg/mol

Volumes estimation of fuels is quite a more complex topic; in fact, volumes depends on tank pressure and temperature. A particular case is the storing of hydrogen; as hydrogen has a very low gas transition point and his density when it is in gas form is very low, the volume required to store it is huge when it is at standard temperature. Usually to overcome this problem either the tank temperature are kept very low to allow to have hydrogen in liquid form or tank are heavily pressurized to reduce the volume.
In this study the following assumptions concerning the tank condition are made:

- Tank Temperature (T_{TANK})=-10°C
- Tank Pressure (P_{TANK})= 100 bar

Both conditions are plausible; temperature is close to moon environmental temperature inside lava tube (-15°C), that means it can be reached a thermal equilibrium between tank and environment very easily.
Concerning pressure, it is a reference value used in other projects. High pressure can be reached by using high pressure electrolyzed which produces directly hydrogen and oxygen at high pressure.

Volumes can be found by using the perfect gas law:

$$P_{TANK} * V_i = R * N_i * T_{TANK}$$

Reversing the equation:

$$\boldsymbol{V_i} = \frac{\boldsymbol{R} * \boldsymbol{N_i} * \boldsymbol{T_i}}{\boldsymbol{P_{TANK}}}$$

Where:

- R= Gas Constant=8.314 J/(mol*k)

- V_i= Volume of i-th component (either oxygen or hydrogen)
- N_i=Volume of i-th component (either oxygen or hydrogen)

Concerning water, volume can be estimate by using mass and density:

$$V_{H_2O} = m_{H_2O} * \rho_{H_2O}$$

Where:

- ρ_{H2O}=Density of water= 1000 kg/m^3

D.3.2 Mass and volume of tanks

Tanks has been designed using as reference inflatable structure to get more transportable and resistant. This study will consider a single inflatable structure made by two parts:

- The restraint layer: which will withstand the pressure loads inside the tanks.
- Multi Layers Insulation (MLI): which will allow thermal control inside the tank

While the restraint layer will be sized on the basis of each tank features (pressure and volume), MLI will be the same for all tanks. In particular, it will be used 10 Mylar/Kapton layers to get a total of 0.25 mm od thickness.
It is important to point out that this design will not use MMDO and Bladder layers, this is made to reduce as much as possible the weight and volume. However, these layers are strongly suggested to get more reliable the structures.

Before going on the analysis, in table below have been resumed all assumptions made concerning structural properties of tanks:

Parameter	Symbol	Values
Density of Kevlar	ρ_{kev} [kg/m^3]	1440
Tensile strength of Kevlar	σ_f [MPa]	3600
Safety factor	SF	4
Density of MLI layers	ρ_{MLI} [kg/m^3]	1420
Thickness of MLI layers	t_{MLI} [mm]	0.25

Table D.3.2-1 Resume of tank properties

The first step is to define the shape of tanks; in this work it has been decided to use the sphere shape because it is the best volumetrically efficient, that means that surface area, volume and mass are the lowest compared to other shapes (as cylinder or torus).

Once shape has been chosen, it is necessary to size it. The key parameter for sphere shape tank is radius. Radius can be found using the following equation:

$$R = \sqrt[3]{\frac{3}{4 * \pi} V_i}$$

The next step is to find the minimum thickness of restraint layer which is able to withstand the tank pressure.
Stress for sphere structure, when the ratio between radius and thickness is higher than 10, can be found using the following equation:

$$\sigma_f = SF * \frac{P_{TANK} * R}{2 * t}$$

Reversing to find the thickness:

$$\boldsymbol{t = SF * \frac{P_{TANK} * R}{2 * \sigma_f}}$$

Where:

- σ_F= Tensile strength of kevlar
- P_{TANK}= Tank pressure=100 bar
- t=Thickness of restraint layer
- R=Radius
- SF= Safety factor

The total thickness of structure IS the sum of MLI and Kevlar:

$$t_{tot} = t + t_{MLI}$$

The final step is to estimate the volume and mass of the only structure.
Concerning volume, as the thickness is very low compared to the radius it can be used the following equation:

$$\boldsymbol{V_{STC} = 4 * \pi * R^2 * t_{tot}}$$

Known the volume, also mass can be estimate using density.

$$\boldsymbol{m_{STC} = V_{kev} * \rho_{kev} + V_{MLI} * \rho_{MLI} \approx V_{STC} * \rho_{kev}}$$

Where:

- V_{kev}= Volume of Kevlar structure
- V_{MLI}= Volume of MLI structure

Approximation used is valid because the two density of Kevlar and MLI re very similar; also, the value is conservative as the density of Kevlar is higher than MLI.

Finally, area occupied by tanks can be estimated using the equation above:

$$\boldsymbol{A_{tank} = \pi * (R + t)^2}$$

It is important to point out that analysis has been conducted considering only one big tank. However, often several tanks are used instead of only one. This is made to optimize the volume and area distribution of tanks on both launcher and surface.
To take into account of separated tanks, it is necessary to repeat procedure above considering as V_i the ratio between the total volume of fuel and the number of tanks. What is more, volume, mass and area shall be multiplied by the number of tanks to obtain total values.

Appendix E: Link Budget

UPLINK

INPUTS	
Radius of Moon [km]	1737,1
Altitude [km]	700,58
Min. elevation angle (°)	15
Boltzmann Constant [dB]	-228,6
Data Rates [bps]	197770,81
Signal Frequency [Hz]	2680000000
Speed of light [m/s]	299792458
Wavelength [m]	0,11186286
Power Transmitted [W]	20
Transmitting antenna diameter [m]	2,5
Receiving antenna diameter [m]	0,5

TRANSMITTING	
TRANSMITTING ANTENNA	
Circumference [m]	7,853982
Gain [dB]	36,9215
Beamwidth (°)	3,134328
TRANSMITTING LOSS	
Cable Length [m]	100
Loss per meter [dB/m]	0,057
Other attenuation [dB]	0,3
Transmitting Loss [dB]	6
EIRP	
EIRP [dB]	43,9318

TRANSMITTING POINTING LOSS	
Pointing error (°)	1
Pointing Loss [dB]	-4,885986

SPACE LOSSES	
POINTING LOSS	
Pointing Loss [dB]	-5,08143
PATH LOSSES	
Slant range [km]	1318,71
Wavelength [m]	0,111863
Path Losses [dB]	-163,416
ATMOSPHERIC LOSSES	
Polarization mismatch [dB]	0
Gas Losses [dB]	0
Rain Losses [dB]	0
Ionosphere Losses [dB]	0
Atmospheric Losses [dB]	0

RECEIVING	
RECEIVING ANTENNA	
Gain [dB]	22,94209597
Beamwidth (°)	15,67164179
RECEIVING LOSS	
Cable Length [m]	2
Loss per meter [dB/m]	0,16
Other attenuation [dB]	0,2
Receiving Loss	0,887156012
SYSTEM NOISE TEMPERATURE	
Reference temperature [K]	290
T_ant [K]	290
LNA Figure of Merit	1,2
System Noise Temperature [K]	392,2647147
System Noise Temperature [dB]	25,93579244

TRANSMITTING POINTING LOSS	
Pointing error (°)	1
Pointing Loss [dB]	-0,19543946

LINK BUDGET	
Eb/No [dB]	48,07937
Eb/No required [dB]	13
Link margin [dB]	35,07937

DOWNLINK

INPUTS	
Radius of Moon [km]	1737,1
Altitude [km]	700,58
Min. elevation angle (°)	15
Boltzmann Constant [dB]	-228,6
Data Rates [bps]	197770,81
Signal Frequency [Hz]	2580000000
Speed of light [m/s]	299792458
Wavelength [m]	0,11619863
Power Transmitted [W]	10
Transmitting antenna diameter [m]	0,5
Receiving antenna diameter [m]	2,5

TRANSMITTING	
TRANSMITTING ANTENNA	
Circumference [m]	1,570796
Gain [dB]	22,61179
Beamwidth (°)	16,27907
TRANSMITTING LOSS	
Cable Length [m]	1
Loss per meter [dB/m]	0,057
Other attenuation [dB]	0,3
Transmitting Loss [dB]	0,357
EIRP	
EIRP [dB]	32,25479

TRANSMITTING POINTING LOSS	
Pointing error (°)	1
Pointing Loss [dB]	-0,181127

SPACE LOSSES	
POINTING LOSS	
Pointing Loss [dB]	-4,70929
PATH LOSSES	
Slant range [km]	1318,71
Wavelength [m]	0,116199
Path Losses [dB]	-163,085
ATMOSPHERIC LOSSES	
Polarization mismatch [dB]	0
Gas Losses [dB]	0
Rain Losses [dB]	0
Ionosphere Losses [dB]	0
Atmospheric Losses [dB]	0

RECEIVING	
RECEIVING ANTENNA	
Gain [dB]	36,59119429
Beamwidth (°)	3,255813953
RECEIVING LOSS	
Cable Length [m]	100
Loss per meter [dB/m]	0,16
Other attenuation [dB]	0,2
Receiving Loss	0,023988329
SYSTEM NOISE TEMPERATURE	
Reference temperature [K]	290
T_ant [K]	290
LNA Figure of Merit	1,2
System Noise Temperature [K]	14507,05454
System Noise Temperature [dB]	41,61579244

TRANSMITTING POINTING LOSS	
Pointing error (°)	1
Pointing Loss [dB]	-4,52816327

LINK BUDGET	
Eb/No [dB]	35,0739
Eb/No required [dB]	13
Link margin [dB]	22,0739

Fig. E.1-1 Link budget of Moon-Moon communication

UPLINK

INPUTS

Radius of Moon [km]	1737,1
Altitude [km]	700,58
Min. elevation angle (°)	15
Boltzmann Constant [dB]	-228,6
Data Rates [bps]	33510000
Singnal Frequency [Hz]	8000000000
Speed of light [m/s]	299792458
Wavelength [m]	0,037474057
Power Transmitted [W]	500
Transmitting antenna diameter [m]	5
Receiving antenna diameter [m]	15

TRANSMITTING

TRANSMITTING ANTENNA	
Circunference [m]	15,70796
Gain [dB]	52,4412
Beamwidth (°)	0,525
TRANSMITTING LOSS	
Cable Length [m]	100
Loss per meter [dB/m]	0,057
Other attenuation [dB]	0,3
Transmitting Loss [dB]	6
EIRP	
EIRP [dB]	73,4309
TRANSMITTING POINTING LOSS	
Pointing error (°)	0,1
Pointing Loss [dB]	-1,741497

SPACE LOSSES

POINTING LOSS	
Pointing Loss [dB]	-17,415
PATH LOSSES	
Slant range [km]	404070,1
Wavelength [m]	0,037474
Path Losses [dB]	-224,421
ATMOSPHERIC LOSSES	
Polarization mismatch [dB]	0,3
Gas Losses [dB]	1,1
Rain Losses [dB]	0,5
Ionosphere Losses [dB]	0,1
Atmospheric Losses [dB]	-2

RECEIVING

RECEIVING ANTENNA	
Gain [dB]	61,98362492
Beamwidth (°)	0,175
RECEIVING LOSS	
Cable Length [m]	2
Loss per meter [dB/m]	0,16
Other attenuation [dB]	0,2
Receiving Loss	0,887156012
SYSTEM NOISE TEMPERATURE	
Reference temperature [K]	290
T_ant [K]	200
LNA Figure of Merit	1,2
System Noise Temperature [K]	302,2647147
System Noise Temperature [dB]	24,80387452
RECEIVING POINTING LOSS	
Pointing error (°)	0,1
Pointing Loss [dB]	-15,6734694

LINK BUDGET

Eb/No [dB]	20,12301
Eb/No required [dB]	13
Link margin [dB]	7,123006

DOWNLINK

INPUTS

Radius of Moon [km]	1737,1
Altitude [km]	700,58
Min. elevation angle (°)	15
Boltzmann Constant [dB]	-228,6
Data Rates [bps]	3396888,889
Singnal Frequency [Hz]	8100000000
Speed of light [m/s]	299792458
Wavelength [m]	0,037011415
Power Transmitted [W]	1000
Transmitting antenna diameter [m]	15
Receiving antenna diameter [m]	5

TRANSMITTING

TRANSMITTING ANTENNA	
Circunference [m]	47,12389
Gain [dB]	62,09153
Beamwidth (°)	0,17284
TRANSMITTING LOSS	
Cable Length [m]	5
Loss per meter [dB/m]	0,057
Other attenuation [dB]	0,7
Transmitting Loss [dB]	0,985
EIRP	
EIRP [dB]	91,10653
TRANSMITTING POINTING LOSS	
Pointing error (°)	0,1
Pointing Loss [dB]	-16,06776

SPACE LOSSES

POINTING LOSS	
Pointing Loss [dB]	-17,8531
PATH LOSSES	
Slant range [km]	404070,1
Wavelength [m]	0,037011
Path Losses [dB]	-224,421
ATMOSPHERIC LOSSES	
Polarization mismatch [dB]	0,3
Gas Losses [dB]	1,1
Rain Losses [dB]	0,5
Ionosphere Losses [dB]	0,1
Atmospheric Losses [dB]	-2

RECEIVING

RECEIVING ANTENNA	
Gain [dB]	52,54910046
Beamwidth (°)	0,518518519
RECEIVING LOSS	
Cable Length [m]	100
Loss per meter [dB/m]	0,16
Other attenuation [dB]	0,2
Receiving Loss	0,023988329
SYSTEM NOISE TEMPERATURE	
Reference temperature [K]	290
T_ant [K]	290
LNA Figure of Merit	1,2
System Noise Temperature [K]	14507,05454
System Noise Temperature [dB]	41,61579244
RECEIVING POINTING LOSS	
Pointing error (°)	0,1
Pointing Loss [dB]	-1,78530612

LINK BUDGET

Eb/No [dB]	21,05502
Eb/No required [dB]	13
Link margin [dB]	8,055024

Fig. E.1-2 Link budget of Moon-Earth communication

Appendix F: Atmosphere Budget

In this appendix it will be described the whole process implemented to estimate both the initial mass of gasses (oxygen and nitrogen) to be provide into habitat and quantify the losses. Based on 'losses budget' it will be possible size ISRU system, for oxygen production and nitrogen to be resupplied during cargo missions.
Whole analysis has been carried out building an Excel spreadsheet, which will be also explained in this appendix.

F.1 Assumptions

Assumptions made to carry out this study are:

- *Habitat pressurized volume:* As described in chapter 4, pressurized volume assumed for habitat is 440 m^3; it comprises both the habitable volume and system volume. It is important to point out that this value depends on the number of crew member according to a proportional law.

- *Airlock pressurized volume:* Also for airlock it has been found the total pressurized volume in chapter 4. In particular considering one equipment lock (34.34 m^3) and two crew locks (9.82 m^3), the total volume of airlock is: 54.98 m^3.
  It is important to point out that only atmosphere of crew lock will be lost during the EVAs, this is an important information that will be used following to estimate air losses.

- *Air consumed by humans:* According to studied, humans in nominal condition consume 0.835 kg per day of oxygen.

In addition to these assumptions made for volumes, other assumptions have been made based on ECLS system requirements [51, 52]:

- Total pressure: 102.7-97.9 kPa
- Oxygen partial pressure: 23.1-19.3 kPa
- Nitrogen partial pressure: 83.2-80 kPa
- Temperature:302.6-291.5 K
- Maximum air leakage allowed: 2.04 kg/day

In addition to these assumptions, other constant values about gasses are reported as molar masses and gas constant.

AIR COMPOSITION		
CONSTANTS AND ASSUMPTIONS		
Overall duration	[days]	180
Crew size	CM	4
Habitable volume	[m^3]	400
System volume	[m^3]	40
Total pressurized volume	[m^3]	440
Airlock volume	[m^3]	54,98
Gas constant	[J/(K*mol)]	8,314
Molar mass of O2	[kg/mol]	0,032
Molar mass of N2	[kg/mol]	0,028
Oxygen consumed	[kg/(CM*day]	0,835

Table F.1-1 Assumption section of Excel spreadsheet

REQUIREMENTS		
Minimum total pressure required	[Pa]	97900
Maximum total pressure required	[Pa]	102700
Minimum N2 partial pressure required	[Pa]	80000
Maximum N2 partial pressure required	[Pa]	83200
Minimum O2 partial pressure required	[Pa]	19500
Maximum O2 partial pressure	[Pa]	23100
Minimum temperature required	[K]	291,5
Maximum temperature required	[K]	302,6
Maximum leakage rate	[kg/s]	2,36111E-05

Table F.1-2 Requirements section of Excel spreadsheet

Last assumptions made concerns EVAs. It has been assumed that a maximum of 10 cycles of EVAs are performed per week. Considering 8 hours of activity for each astronaut for each EVA, It can be estimated 80 hours of EVA per week per astronaut [34].

F.2 Habitat set up Atmosphere budget

Set up atmosphere budget is carried out considering the worst condition, lowest temperature and highest pressure.
To estimate gas masses it has been used the Dalton law:

$$P_i * V = RN_iT \ [F.2-1]$$

Introducing the molar mass:

$$P_i * \frac{V}{m_i} = R * \frac{I}{M_i} * T \ [F.2-2]$$

Expressing mass in function of other parameters:

$$m_i = \frac{P_i * M_i * V}{R * T} \quad [F.2-3]$$

Where:

- m=Mass of i-th component [kg]
- P_i= Partial pressure of i-th component [Pa]
- V=Total volume (either of habitat or airlock) [m^3]
- R= Gas constant=8.314 [J/(K*mol)]
- M_i= Molar mass of i-th component [kg/mol]
- i= Gas component, oxygen or nitrogen

Applying equation F.2-3 to both habitat and airlock on both oxygen and nitrogen, it can be obtained:

INITIAL MASS OF GASES REQUIRED		
MASS OF GASES REQUIRED - HABITAT		
Mass of O2 only nominal condition	[kg]	134,2042
Mass of N2 only nominal condition	[kg]	483,3675
MASS OF GASES REQUIRED - AIRLOCK		
Mass of O2 only nominal condition	[kg]	16,76942
Mass of N2 only nominal condition	[kg]	60,39896
MASS OF GASES REQUIRED - TOTAL		
Mass of O2 only nominal condition	[kg]	150,9736
Mass of N2 only nominal condition	[kg]	543,7664

Table F.2-1 Initial mass of gases estimation (Excel spreadsheet section)

F.3 Atmosphere Losses

Estimation of atmosphere losses starts considering the causes of losses. As inside the habitat oxygen and nitrogen are used for different purpose, the cause of losses are not the same for both gases.
Oxygen losses are caused by:

- *Inefficiency of air revitalization system:* Air Conditioning System (ACS) can have an efficiency lower than 100% that means part of oxygen is lost during the revitalization process. As precise data about efficiency of ACS are not available in bibliography, it has been assumed an efficiency of 90% [4]. That means only 10% of oxygen process per day is lost. Considering that ACS process every day the among of oxygen consumed by astronauts multiplied for the number of crew members, we can estimate losses due to ACS as follow.

$$L_{OX_{ACS}} = 10\% * 0.835 \left[\frac{kg}{CM * day}\right] \quad [F.3-1]$$

- *Losses due to EVAs:* They are caused by the loss of oxygen during EVAs activities. It has been estimated that 0.15 kg of oxygen are lost for each hour of EVA [REFERENCE]. Considering a total of 80 hours of EVAs per week we have:

$$L_{OX_{EVA}} = 0.15 * \frac{80}{7} * \rho_{OX} \left[\frac{kg}{day}\right] \quad [F.3-2]$$

- *Losses due to airlock*: Every time EVA is performed, part of airlock atmosphere is lost. It has been estimated that 10% of airlock gas volume is lost for each cycle EVA (10 cycles per week). It is important to point out only crew lock volume shall be considered as equipment lock volume is not lost during EVAs. In addition, the value takes into account loss of air and not only oxygen, it is so necessary to take into account oxygen represents roughly 21% of total atmosphere lost.

$$L_{OX_{AIRLOCK}} = 10\% * V_{CREW\ LOCK} * 21\% * \frac{10}{7} * \rho_{OX} \left[\frac{kg}{day}\right] \ [F.3-3]$$

- *Losses due to leakage:* The last cause of oxygen lost is the normal leakage from habitat through the walls. As declared by requirements, the maximum value allowed of air leakage is 2.04 kg/day. Also in this case, it is necessary to take into account oxygen represents only the 21% of total air leaked.

$$L_{OX_{LEAKAGE}} = 21\% * 2.04 \left[\frac{kg}{day}\right] \; [F.3-4]$$

Equations above have been implemented into Excel to determinate the total oxygen losses every day.

OXYGEN		
Losses due to ACS	[kg/(day*CM)]	0,0835
Losses due to EVAs	[kg/day]	1,714
Losses due to Airlock leakage	[kg/day]	0,421
Leakage	[kg/day]	0,4284
Total oxygen loss per day	[kg/day]	2,8974
Total oxygen loss per mission	[kg/mission]	521,532
Total oxygen loss per year	[kg/year]	1057,551

Table F.3-1 Oxygen loses Excel section and results

As nitrogen is not used during the EVAs and it is not processed by revitalization ACS system, these losses are not present; the only loses of nitrogen are:

- Loses due to leakage: Similarly, to oxygen, also nitrogen is lost during opening of airlock. Equation is the same with the only difference that air is composed by 79% of nitrogen.

$$L_{N2_{AIRLOCK}} = 10\% * V_{CREW\ LOCK} * 79\% * \frac{10}{7} * \rho_{N2} \left[\frac{kg}{day}\right] \; [F.3-5]$$

- Leakage: Also nitrogen is lost due to leakage from habitat. Equation used is similar to F.3-4 with the difference concerning the nitrogen composition of air.

$$L_{N2_{LEAKAGE}} = 79\% * 2.04 \left[\frac{kg}{day}\right] \; [F.3-4]$$

Equations above have been implemented into Excel to determinate the total oxygen losses every day.

NITROGEN		
Leakage	[kg/day]	1,6116
Losses due to Airlock leakage	[kg/day]	1,58
Total nitrogen loss per day	[kg/day]	3,1916
Total nitrogen loss per mission	[kg/mission]	574,488
Total nitrogen loss per year	[kg/year]	1164,934

Table F.3-2 Nitrogen loses Excel section and results

Appendix G: Water budget estimation

In this appendix it will be explained the process utilized to estimate the water consumption using data available in bibliography. After that it will be estimated the wet waste budget, also in this case using data available. After that, estimated the total wet waste to be processed, it will be estimated the water loss, which should be resupplied during cargo missions.
The whole process has been carried out building an Excel spreadsheet, details of it will be also explained in this appendix.

G.1 Water budget

In this study it has been assumed that water is mainly utilized for the following macro needs:

- *Drinking and Food:* This is the water required both to process food and to drink. According past studies and ISS experience, it has been assumed a drinking water consumption of 2.8 litres (equal to kilograms) per crew member per day. Concerning food processing, it has been assumed a total of 0.5 litres per day. [34].

$$M_{WATER_{D\&F}} = 2.8 * CM + 0.5 \ \left[\frac{kg}{day}\right] \ [G.1-1]$$

- *Hygiene:* It comprises all activities related to self-person washing: shower, face/hand wash, physiological needs (urine and feces) and teeth washing. It has been assumed a total water need of 7.65 kg per day per crew member. [53]

$$M_{WAT \quad HYG} = 7.65 * CM \ \left[\frac{kg}{day}\right] \ [G.1-2]$$

- *Machine washing:* It includes the water consumption both to wash crockery and to wash clothes. Unluckily, at the moment there is not any real project of washing machines for lunar application and so values are very variable and not accurate. In this works it has been assumed a total water consumption of 71.64 kg per day [53].

$$M_{WATER_{WM}} = 71.64 \ \left[\frac{kg}{day}\right] \ [G.1-3]$$

- *Medical:* It includes water needed for all medical needs. In this work it has been assumed that medical water con be expressed by the following equation, where part of water needed depends on number of crew member and part is fixed [53].

$$M_{WATER_{MED}} = 0.5 * CM + 7 \ \left[\frac{kg}{day}\right] \ [G.1-4]$$

Equations above have been implemented into the Excel spreadsheet. Results take into account an addition 5% of margin.

WATER BUDGET		
DRINKING WATER & FOOD WATER		
Water required to drink	[kg/(day*CM)]	2,8
Water required for rehydratation	[kg/day]	0,5
Total drinking and food water per day	[kg/day]	11,7
HYGIENE WATER		
Oral water	[kg/(day*CM)]	0,36
Urinal Flush Water	[kg/(day*CM)]	0,49
Hand and Face wash water	[kg/(day*CM)]	4,08
Shower	[kg/(day*CM)]	2,72
Total Hygiene water per day	[kg/day]	30,6
MACHINE WASH WATER		
Dish Washer water	[kg/(day*CM)]	5,44
Clothing Washer Water	[kg/(day*CM)]	12,47
Total Machine Washer water per day	[kg/day]	71,64
MEDICAL WATER		
Medical water	[kg/(day*CM)]	0,5
Total Medical water per day	[kg/day]	7
Total Water required per day	[kg/day]	126,987
Total Water required per mission	[kg/mission]	22857,66
Total Water required per year	[kg/year]	46350,255

Table G.1-1 Water budget implemented in Excel spreadsheet

G.2 Wet Waste budget

Wet waste budget takes into account all outputs from users which use water. Details about wet waste budget is shown in table below [53].

DIRTY WATER		
Urinal Water	[kg/(day*CM)]	1,5
Flush urinal water	[kg/(day*CM)]	0,5
Total dirty water	[kg/day]	8
WASTE WATER		
Oral Water	[kg/(day*CM)]	0,36
Hand and Face wash	[kg/(day*CM)]	4,08
Shower	[kg/(day*CM)]	2,72
Dish Washer water	[kg/(day*CM)]	5,44
Clothing Washer Water	[kg/(day*CM)]	12,47
Dirty water	[kg/day]	6,8
Total waste water	[kg/day]	112,434

Table G.2-1 Wet waste implemented in Excel spreadsheet

Also in this case an additional 5% of margin is considered.

It is important to recall the difference between dirty water and waste water. While dirty water comes from urinal flush, and need particular recycle process, waste water come from all other users and does not need particular recycling process.

G.3 Water Loss

Water loss is caused by recycling process, in particular:

- Dirty water: Dirty water is processed by UPA to get waste water. According data available in bibliography, UPA efficiency is 85% [54].
  It is important to say that this value come from experience of ISS where, due to microgravity environment urine is usually rich of minerals, this causes a decrease of efficiency; this means that future UPA utilized in partial gravity environment could have higher efficiency.
  Waste water produced from dirty water can be expressed as follow.

$$M_{WASTE\,WATER}\ (from\ dirty\ water) = 85\% * M_{DIRTY\,WATER}\ \left[\frac{kg}{day}\right]\ [G.2-1]$$

  And so, dirty water loss is:

$$M_{LOST_{DW}} = 15\% * M_{DIRTY\,WATER}\ \left[\frac{kg}{day}\right]\ [G.2-2]$$

- Waste water: Waste water is processed by Water Processing Assembly (WPA). There is not accurate data in bibliography, for this reason it has been assumed an efficiency of 90% in line with assumption made by ESA/MIT/SoM [4]. Water lost is:

$$M_{LOST_{WW}} = 10\% * M_{WASTE\,WATER}\ \left[\frac{kg}{day}\right]\ [G.3-3]$$

Equations above have been implemented in the Excel spreadsheet.

LOSS OF WATER		
Loss of Diry Water due to UPA	[kg/day]	1,2
Loss of water due to WPA	[kg/day]	11,2434
Total Water Loss per day	[kg/day]	12,4434
Total Water Loss per mission	[kg/mission]	2239,812
Total Water Loss per year	[kg/year]	817531,38

Table G.3-1 Water loss Excel implementation

Appendix H: Thermal Analysis of Habitat

The main purpose of this thermal analysis is to estimate the thermal power lost by walls and which must be generated by heaters to maintain constant temperature inside the habitat in conformance with ECLS system requirements. In addition, it will be estimate temperature of walls; this represents another necessary information to understand if wall temperatures meet the defined requirements.

H.1 Thermal model

The thermal analysis have been carried out using as reference A.Peresotti thermal model [5].
This thermal model considers three types of heat exchange:

- *Convection*: Due to the presence of atmosphere inside the habitat, convection can happen. Heat inside the habitat is transmitted from the Air to the internal layer of shell.

- *Conduction*: It is how the heat is transmitted through the shell layer.

- *Irradiation*: It is how heat is transmitted by external layer of shell to lunar environment. As there is not atmosphere on the Moon, it represents the only way to transmit heat from habitat to external environment.

Conductive heat transmission equation is:

$$Q_{conv} = A_{int} * H_{air} * \left(T_{in} - T_{w,in}\right) \ \ [H - 1.1]$$

Where:

- Q_{conv} = Thermal power transmitted per convection [W]
- A_{int}= Internal area of habitat [m_2]
- H_{air}= Air thermal convection coefficient [W/(m^2 * K)]
- T_{in}= Temperature inside the habitat [K]
- $T_{w,in}$=Temperature of shell internal layer [K]

Concerning conduction heat transmission equation:

$$Q_{cond} = A * \lambda * \frac{T_{w,in} - T_{w,out}}{L} \ [H - 1.2]$$

Equation H-1.2 represents the typical equation for conduction heat transmission; however, as we will consider a multi thin multilayer shell composed by different material and so different thermal conductivity values, it is more convenient to use the following equation, which use the concept of thermal resistance:

$$Q_{cond} = 2 * \pi * h * \frac{\left(T_{w,in} - T_{w,out}\right)}{R_{tot}} \ [H - 1.3]$$

Where:

- Q_{cond} = Thermal power transmitted per conduction [W]
- h= Height of habitat [m]
- R_{tot}=Thermal resistance of whole shell
- $T_{w,out}$= Temperature of shell external layer [K]
- $T_{w,in}$=Temperature of shell internal layer [K]

Considering geometric with cylindrical symmetry, it is possible to obtain the toral resistance with the following equation:

$$R_{tot} = \sum_{i=1}^{n} \frac{1}{\lambda_i} * \ln\left(\frac{r_{i+1}}{r_i}\right) \ [H - 1.3.1]$$

Where:

- λ_i=Thermal conductivity of i-th layer [W/(m*K)]
- r_i=Radius of i-th layer [m]

Finally the equation for irradiation heat transmission is:

$$Q_{irr} = \varepsilon * \sigma * A_e * \left(T_{w,out} - T_e\right) \ [H - 1.4]$$

Where:

- Q_{irr} = Thermal power transmitted per irradiation [W]
- ε= Emissivity
- σ=Boltzman Constant [W/(m^2 * K^4)]
- A_e=External area of shell [m^2]
- $T_{w,out}$= Temperature of shell external layer [K]
- T_e= External temperature [K]

In this problem the unknown values are the two temperature of wall and the heat transmitted.
To solve the problem it is necessary to combine the three equation in a single equation which has only one unknown. Found that value, it is possible to estimate also the other two unknowns.
Let's combine equation H-1.1 and H-1.2 to write $T_{w,in}$ as only function of $T_{w,out}$:

$$(2 * \pi * ri * h) * H_{air} * \left(T_{in} - T_{w,in}\right) = 2 * \pi * h * \frac{\left(T_{w,in} - T_{w,out}\right)}{R_{tot}}$$

We can write:

$$T_{w,in} = \frac{r_i * H_{air} * T_{in} * R_{tot} + T_{w,out}}{r_i * H_{air} * R_{tot} + 1} \ [H - 1.5]$$

Then we can combine equations H-1.1 and H-1.4 where $T_{w.in}$ can be substituted using equation H-1.5.
We obtain:

$$(2 * \pi * r_i * h) * H_{air} * \left(T_{in} - \frac{r_i * H_{air} * T_{in} * R_{tot} + T_{w,out}}{r_i * H_{air} * R_{tot} + 1}\right) = \varepsilon * \sigma * (2 * \pi * r_e * h) * \left(T_{w,out}^4 - T_e^4\right) \ [H - 1.6]$$

All values in equation XX-1.6 are known except $T_{w,out}$. We can solve this equation non linear equation writing equation in form of *f($T_{w,out}$)=0* and using a numerical method or Matlab function.

$$r_i * H_{air} * \left(T_{in} - \frac{r_i * H_{air} * T_{in} * R_{tot} + T_{w,out}}{r_i * H_{air} * R_{tot} + 1}\right) - \varepsilon * \sigma * r_e * \left(T_{w,out}^4 - T_e^4\right) = 0 \ [H - 1.7]$$

Found $T_{w,out}$ we can obtain the heat loss using equation H-1.4 and then we can found also $T_{w,in}$ using equation H-1.3.

H.2 Assumptions

Assumptions made to perform this thermal analysis are:

- *Habitat Temperature*: As shown in requirements table (8.2-1) the required temperature of habitat is comprises between 18.5 °C and 29.6°C. To carry out the analysis it has been used an average value of 23 °C.

- *External temperature*: As reported by Benaroya [7] the estimate average temperature, supposed to remain constant, inside Lava Tube is roughly -15°C.

- *Habitat size and volume*: As described in chapter 4, the reference habitat is a cylindrical module with maximum heigh of 17 m and maximum internal diameter of 6.55 m. The total volume is 573.77 m^3.

- *Inflatable shell*: As described in chapter 4, habitat is separated from outside by an inflatable shell composed by several layers of different material. The total thickness of shell is 17.43 cm. Detail of thickness and thermal conductivity of material is reported in table below.

Layer	Conductivity [W/(m*K)]	Layer Thickness/Total Thickness [%]
Nomex	0.16 [REF]	0.57
Nylon	0.23 [REF]	0.583
Kevlar	0.3 [REF]	7.34
Foam	0.042 [REF]	87.2
MLI	0.14[REF]	0.287
Nextel	0.13 [REF]	4.02

Table H.2-1 Resume of shell layers features

- *Heat internal sources*: In this analysis it has been assumed negligible the heat generated by systems. Heat produced by human has been assumed to be 400 W per crew member [34].

- *Heat exchange direction:* As the major part of heat is transmitted through the lateral are, in this preliminary analysis it will be considered only that contribute.

- *Emissivity:* Thanks to use of MLI, it is possible to assume a low value of emissivity. In particular, using 20 MLI it is possible to assume a value of 0.1. [REFERENCE].

H.3 Matlab implementation

The first part of script it is used to declare all assumptions and set constant.

```
%% DATA
%%Habitat data
h=17; %Height of habitat [m]
Vol=573.77; %Total volume of habitat [m^3]
ri=sqrt(Vol/(h*pi));  %Internal radius of habitat [m]
L=0.1743; %Thickness of wall [m]
re=ri+L;  %External radius of habitat [m]

%%Constants
H=10; %Coefficient of air heat exchange [W/(K*m^2)]
eps=0.1; %MLI emissivity
sigma=5.67*10^(-8); %Boltzmann Constant [W/(m^2*K^4)]
W_hum=300*4; %Average heat produced by humans [W]
Te=273-15;  %External temperature [K]
Ti=273+23; %Initial Internal temperature [K]

%%Data Wall layers
lambda_nomex=0.16; %Thermal conductivity of nomex [W/(m*K)]
L_nomex=0.0057*L;
lambda_nylon=0.23; %Thermal conductivity of nylon [W/(m*K)]
L_nylon=0.00583*L;
lambda_kevlar=0.3; %Thermal conductivity of kevlar [W/(m*K)]
L_kevlar=0.0734*L;
lambda_foam=0.042; %Thermal conductivity of foam [W/(m*K)]
L_foam=0.872*L;
lambda_MLI=0.14; %Thermal conductivity of MLI [W/(m*K)]
L_MLI=0.00287*L;
lambda_nextel=0.13; %Thermal conductivity of nextel [W/(m*K)]
L_nextel=0.04016*L;
```

Fig. H.3-1 First part of Matlab script

In picture it is possible to identify three declaration parts:

- *Habitat data:* Which allow to set all assumption made about habitat design. While all data are inserted as input, internal radius is estimated directly by the code, known the total volume, using the following equation, valid for cylindrical structure.

$$r_i = \sqrt{\frac{Volume}{\pi * h}}$$

- *Constants:* Here are declared various constant values. Temperatures are set depending on assumption made.

- *Data wall layer:* Here are declared thicknesses and conductivities for each layer of shell. Data are inserted depending on the required stratification.

After all data have been declared, script estimates the total thermal resistance of shell using equation H.1-3.1.

```
lambda_vec=[lambda_nomex, lambda_nylon lambda_kevlar, lambda_foam,lambda_MLI, lambda_nextel];
L_vec=[L_nomex, L_nylon, L_kevlar, L_foam, L_MLI, L_nextel];
n=length(lambda_vec);
```

```
%% Extimation of Total Thermal Conducibility of wall
r=zeros(1,(n+1));
r(1)=ri;
for i=1:n
    r(i+1)=r(i)+L_vec(i);
    R(i)=1/lambda_vec(i)*log(r(i+1)/r(i));
end
R_tot=sum(R);
```

```
%% Geometrical data
Si=2*pi*ri*h;  %Internal habitat heat exchange surface [m^2]
Se=2*pi*re*h;  %External habitat heat exchange surface [m^2]
S_avg=(Si+Se)/2; %Average habitat heat exchange surface [m/2]
```

Fig. H.3-2 Second part of Matlab script

Estimation is performed first creating two vectors (lambda_vec and L_vec) which contain all data about shell layers. Then, using the 'for' loop, which goes from 1 up to 'n=number of layers', it is possible to estimate resistance of each layer and then total resistance of shell.
In case of addition of new layer, it is only necessary to declare feature of new layer (thickness and conductivity) and modify the vectors adding the new value; script will autonomously consider the new layer.

The third part of code has the aim to estimate the surface area on internal part and external part of shell. Equation used is the classic lateral area of a cylinder.

```
%% Geometrical data
Si=2*pi*ri*h;  %Internal habitat heat exchange surface [m^2]
Se=2*pi*re*h;  %External habitat heat exchange surface [m^2]
```

Fig. H.3-3 Third part of Matlab script

Next step is to solve equation H-1.7 to estimate $T_{w,out}$. To do it has been used matlab function 'fsolve'.

```
%% Extimation of Tw_o

u=@(x) ri*H*Ti-ri*H*(ri*H*Ti+(x/R_tot))/(ri*H+(1/R_tot))-eps*sigma*re*x^4+eps*sigma*re*Te^4;
Tw_o=fzero(u,280);
```

Fig. H.3-4 Fourth part of Matlab script

Known the first temperature, it is possible to determinate the heat loss using equation H-1.4.
After that, it is also possible to estimate $T_{w,in}$ solving equation H-1.3.

```
%% Heat required and Tw_i
w_loss=eps*sigma*(Tw_o^4-Te^4); %Heat rejected from habitat to external environment [W/m^2]
W_loss= w_loss*Se; %Heat rejected from habitat to external environment [W]
W_req=(W_loss-W_hum)/1000; %Heat to be produced by heaters [kW]
Tw_i=(W_loss*R_tot)/(2*pi*h)+Tw_o;

%% Results
W_req
```

Fig. H.3-5 Final part of Matlab script

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