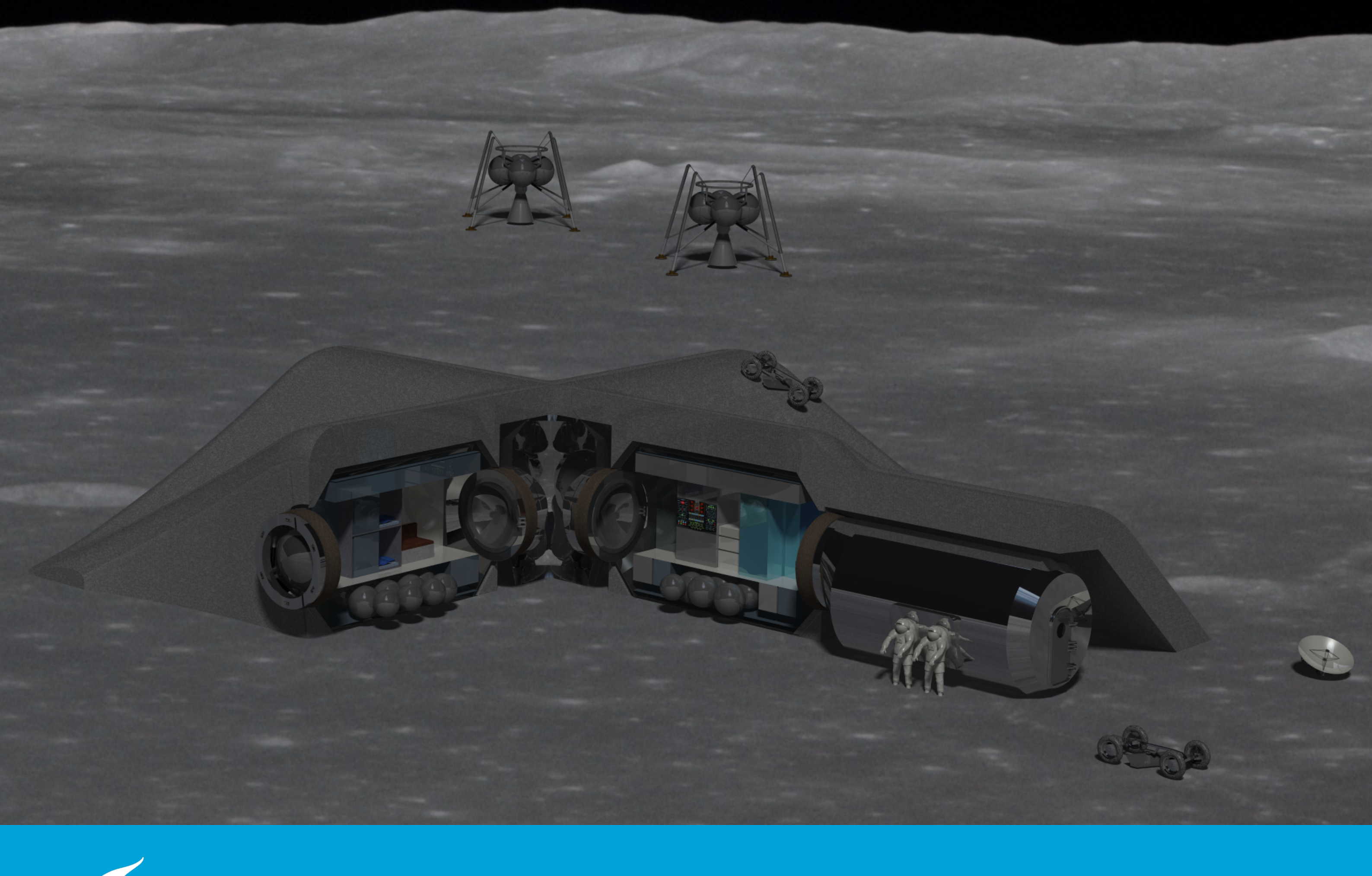
LUNARGOS

The first ever habitat on the Moon

DSE Group 26

Final Report
AE3200 - Design Synthesis Exercise



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AE3200 - Design Synthesis Exercise
Final Report

April - July 2018 (2017-2018 Q4)

by

DSE Group 26

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Cover image partly retrieved from [1]

Preface

July 20th, 1969: astronauts Neil Armstrong and Buzz Aldrin set foot on the Moon for the first time in the history of mankind. Forty nine years later, a team of ten students at the Delft University of Technology begins a project to design the first ever habitat on the Moon: LUNARGOS. Situated in the Peloponnese, Greece, Argos is one of the most antique cities in Europe. Being continuously inhabited its entire history, the team was inspired by its name and merged it with Luna, the ancient Roman divine personification of the Moon, to create the pioneering habitat name of LUNARGOS.

The Lunar base shall be designed to accommodate a total of four astronauts for a period of one year, with the possibility of future expansions and a minimum operational lifetime of ten years. The pioneering technologies, the alternating eclipses on the Moon, and the harsh environment drastically increase the technical challenges and the complexity of design choices. However, the team is target focused, highly motivated, and determined to succeed.

This is the conclusive report in fulfilment of the requirements for the course AE3200 - Design Synthesis Exercise, in the final year of the Bachelor in Aerospace Engineering at the TU Delft. This document is preceded by three preliminary ones, precisely a project plan to outline the basis of the assignment, a baseline report to initiate the design phase of the mission, and a Midterm Report to assess the identified possibilities and select a final concept. In fact, the chosen design idea is extensively investigated and analysed in this report. An in-depth study has been performed in terms of all stages of the mission: from orbit transfer, to landing operations, through both interior and exterior habitat layouts, as well as each subsystems design, and end-of-life sustainability approach.

We are indebted to Prof.dr.ir. S. van der Zwaag for mentoring the team with his extensive technical feedback and instructions throughout this eleven weeks process. Moreover, we would like to thank D. de Tavernier MSc. and Y. Xiao MSc. for the help and support they have provided since the beginning of this project. Furthermore, we are grateful to the PM/SE team for their insights and guidelines, especially to Dr.ir. W. Verhagen, Dr.ir. E. Mooij, and S. Singh. Finally, we would like to thank experts Dr. A. Menicucci for the explanations on radiation in space and Ir. B. Blank for his insights on the structure of the Moon surface, Dr. S.J. Hulshoff and Dr. R.P. Dwight for their support in computational modelling and Dr. Venkatesha Prasad and Sujay Narayana Msc. for their support in designing the Command and Data Handling system. Last but not least, each member of the team is extremely thankful to their family, friends and loved ones for their continuous affection throughout this whole experience.

DSE Group 26
Delft, June 27, 2018

Executive Summary

Our project's target is to design a Lunar Habitat hosting a crew of 4 astronauts for at least one year at a time, with a total mission lifetime of 10 years. The primary challenge faced is providing a reliable and safe environment in which astronauts wear minimal protective gear, in an environment that has a strongly fluctuating temperature, continuous meteorite impacts, high radiation levels, and a lack of atmosphere. This has to be done while complying with the international legislation regarding Moon exploration, using launchers that are commercially available no later than 2018, having adequate backup facilities to allow the safe return of the astronauts to Earth under all conditions and staying below an all-inclusive cost 500000 $€/kg$ of material delivered to the Moon (including research, development, manufacturing, transport, deployment, usage, and disposal). The project aspires to contribute to European Space Agency's (ESA) research on extraterrestrial habitation possibilities of humankind, improving the human species chances of long-term survival.

During the market analysis, the team identified that we lie at a crucial moment in the market where no similar product has ever been constructed, yet the opportunity to design and build this project would provide significant value to multiple stakeholders. Currently, the moon is seen as a target for five of six biggest space agencies on earth, as it is seen as a logical step towards colonising Mars. Furthermore, a significant rise in the number of space corporations has been occurring in the last decades, from less than 10 in the early 2000s to more than 60 in 2016. Most of them looking into space tourism, asteroid mining and Mars colonisation, for all of which a Lunar colony is of prime interest. However, this project does have weaknesses and threats, such as the high price tag which our primary client cannot afford and public opinion that does not prioritise space exploration. These can be mitigated by generating value for multiple stakeholders that do not rely on public opinion, such as space corporations and focusing on producing strong marketing campaigns. Based on the opportunities derived from the market analysis and the stakeholder requirements, the following concepts where generated:

1. Concept 1 - Reusable Lander Base: using the landing vehicle as the habitat itself, by interconnecting various landers together;
2. Concept 2 - Inflatable Dome: a habitat consisting of an inflatable dome, stemming out of a rigid cylinder;
3. Concept 3 - Rigid Cylinder: transporting rigid pre-fabricated cylinders from Earth, placing them horizontally and interconnecting them;
4. Concept 4 - Build-in Design: constructing an underground dome, by either using a crater/cave or digging a hole, inflating a structure in it and then covering it with regolith;
5. Concept 5 - Honeycomb Design: constructing a series of domes out of hexagonal tiles.

Concept four and five were discarded after a first qualitative trade-off as installation was found to be unfeasible. After further development, a second quantitative trade-off was performed based upon safety, mass, sustainability, TRL and launch cost. From this trade-off, concept three, the rigid cylinder, was found to be the best design, as it had the highest score as well as being the safest design. A sensitivity analysis was performed which confirmed the outcome.

A site 10 km north-east of the Apollo 11 landing site was selected as the building location of the habitat, and the geological, soil, radiation, thermal and meteoroid characteristics of this area were explored in detail.

After multiple iterations, the final habitat design consists of four horizontally placed cylinders, connected through a central node, an airlock, multiple rovers and an escape vehicle. The cylinder design is constrained by the size of the payload fairing of the chosen launch vehicle, specifically SpaceX's Falcon 9. To optimise the space use, the ends of the cylinders are conically shaped to fit in the conical shape of the fairing, the payload adapter is redesigned, the landing struts are folded, and the fuel is stored in the walls of the cylinder. The last feature is of special interest as this saves significant volume, serves as a sandwich structure to resist buckling loads during launch, and can be filled with water during operation to provide radiation protection.

The modules are interconnected using a specially designed mechanism to overcome misalignments during construction. The mechanism consists of three rings, two on the connecting module and one on the module to be connected to. The two rings on the connecting module are themselves interconnected by 6

actuators providing 6 degrees of freedom movement and a flexible material providing an airtight seal. In the centre of the base there is a node providing four connecting points. The final layout of the base can be seen in figure 1.

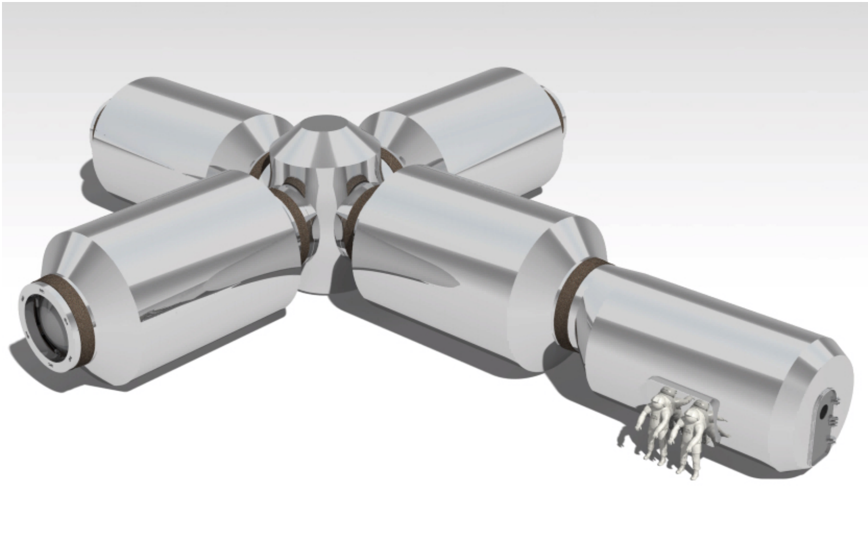


Figure 1: External layout of the cylinders, central node, and airlock

Given the Lunar environment has high radioactivity levels, meteorite impacts with 21.3 kJ of energy and temperature fluctuations from 100 K to 400 K, protective systems needed to be designed. It was concluded that one of the most efficient solutions is covering the base with a layer of 1.2 m of lunar regolith in combination with the aforementioned 4 cm thick water layer, in which radiation protection was the constraining factor. This configuration allows for meteoroid probability of no penetration significantly higher than the required of 99.8%, based on the Grun model and the Fish summer equation. Furthermore, the radiation dose inside the habitat will be lower than 78.3 mGy/yr and a stable temperature on the habitat walls of 247.50 $K \pm 0.5$ K will be reached. The radiation levels will be measured using piezoelectric film sensor panels, which will be mounted outside of the shell modules, similar to the ISS. As a first-order structural integrity estimation, multiple load cases on the cylinders were considered, during launch and operation. Furthermore, after multiple materials were investigated, it was determined that the cylinders would be made out of Aluminium 7055-T7751, resulting in the values for the layers as shown in table 1.

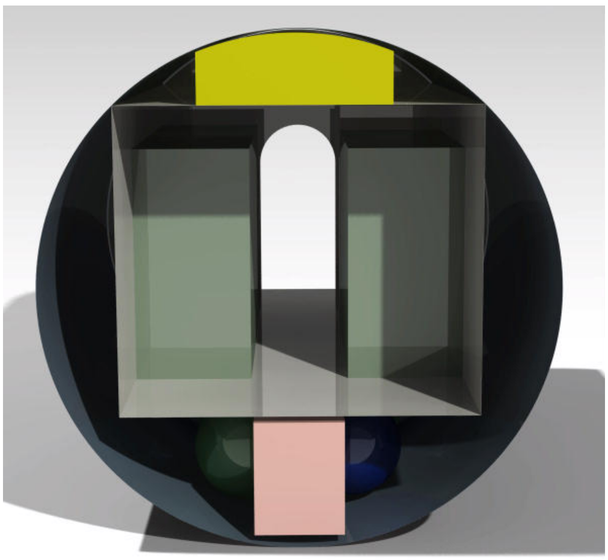
Table 1: Final dimensions of the structural layers, from outside in

Layer	**Thickness** $[mm]$	**Length** $[m]$	**Radius** $[m]$	**Weight** $[kg]$
Regolith	1200	7.7	-	-
Aluminium 7055-T7751	1.60	7.7	2.3	510
Water	40	5.9	$\approx$ 2.25	-
Aluminium 7055-T7751	2.74	7.7	2.25	830

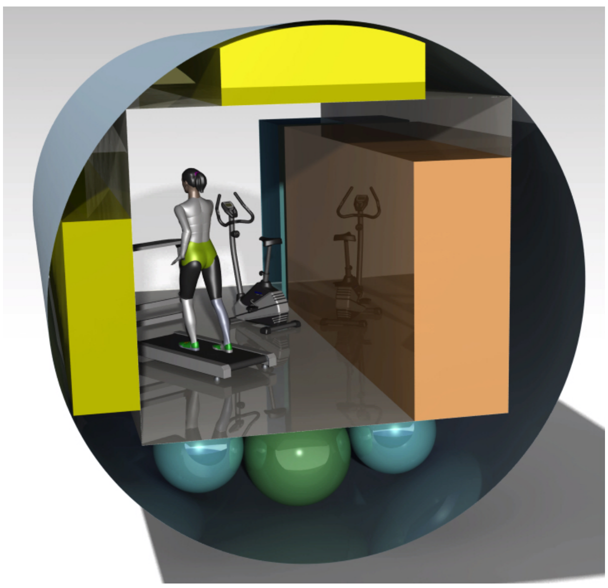
Communication system consists of two parabolic high gain antennas and an omnidirectional low gain antenna. HA6-1 provides the primary communication and has a diameter of 1.5 meters, and HA6-2 provides the BOL communication and serves as a redundant system. Data rates up to 564 Mbps can be reached, having a round trip time of 2.71 seconds. Concerning life support, 4200 kg of food will be brought from earth while experiments on growing food will take place using the Advanced Plant habitat. All waste left at the end of the mission will be sent back to earth using the return vehicle. To recycle water, the Alternative Water Processor will be used as it can also process hygiene and laundry water with a total recovery rate of 90%. The alternative water processor uses four bio-reactors which utilise bacterial metabolic processes to remove carbon and nitrogen from water by means of nitrification and denitrification. Afterwards, the residual water goes through a process of forwarded and reversed osmosis to assure removal of large molecules. The internal air pressure is chosen to be 101.4 kPa, while having a 21% oxygen and 79% nitrogen content with a 50% humidity. The atmospheric management system will regulate the air composition, based on the inputs received from the environmental monitoring system. It does so by utilising Astine beds for CO_2 removal, Sabatier reactors to reprocess the removed CO_2 and fixed alkaline electrolysis to generate the required oxygen. The inner thermal control system uses two active systems of water and ammonia trough heat pipes, as well as a passive system of regolith insulation. The water pipes are connected to the water layer in the habitat shell to ensure better climate control. Noise control will be provided by placing acoustic barriers on both the inside and outside of the structural wall, while simultaneously investigating the possibility of active noise control. Noise will be kept below NC-50 in every module, while the living cabin will serve as a quiet room having a maximum noise level of NC-40. More medical equipment than is present on the ISS will be taken, as evacuation is less feasible for this mission. However, future development is needed as knowledge on the subject is limited. To reduce muscle atrophy, the treadmill with Vibration Isolation and Stabilisation System, the ISS Advanced Resistive Exercise Device and the Cycle Ergometer with Vibration Isolation and Stabilisation as are used on the ISS are installed in the lunar habitat. The Command and Data handling system was designed to be completely redundant and distributed, with a reliability of 99.99 %. It is designed using a combination of commercial off-the-shelf components and radiation hardened components as it needs to continue functioning in critical situations where radiation protection breaches are possible.

Due to high eclipse time and high power requirement a conventional power system is not possible. After a thorough exploration of feasible power systems, the Kilopower project from NASA has been chosen. The use of this system lays the path to closer cooperation with NASA, supplying them with a chance to test their system close to earth, before it may be applied to Mars missions or deep space orbiters. KRUSTY (Kilowatt Reactor Utilizing Stirling Technology) is a nuclear fission reactor using a small and compact fast neutron reactor. The heat from the core is transported using sodium heat pipes to 4 Stirling generators. These generators are hooked up to a titanium radiator, to get rid of the waste heat and to supply a cold side for the thermodynamic cycle. Its design is inherently safe by making sure the dynamics between the core and converters are an underdamped, stable system. During launch, a boron carbide rod is inserted in the core. This neutron absorber is only pulled out after full installation of the system. The reliability of the system, assuming 3 out of 8 generators still supplies enough power to sustain life in emergency situations, is over 99.9999%. Contrary to solar panels, this system can survive without any significant maintenance, having no issues with dust or micrometeoroids.

As previously mentioned, the base is primarily made up of 4 cylinders. The first module, seen in figure 10.4a, contains the ECLSS system, bathroom, laundry facilities and medical bay. The second module, seen in figure 10.4b, contains the main working area, basic research equipment, exercise equipment and backup medical equipment. The third module, seen in figure 10.5a, is primarily designed as living quarters, containing sleeping capsules, personal storage space, a sofa and entertainment system. Finally, figure 10.5b shows the safety and command module, which serves as a shelter for the astronauts during emergencies. It is completely redundant, containing EVA suits, a toilet, a down-scaled ECLSS, emergency provisions, backup power, the main CDH system, medical and maintenance equipment, and secondary sleeping quarters. It is directly connected to the airlock, through which the astronauts can reach the escape vehicle.

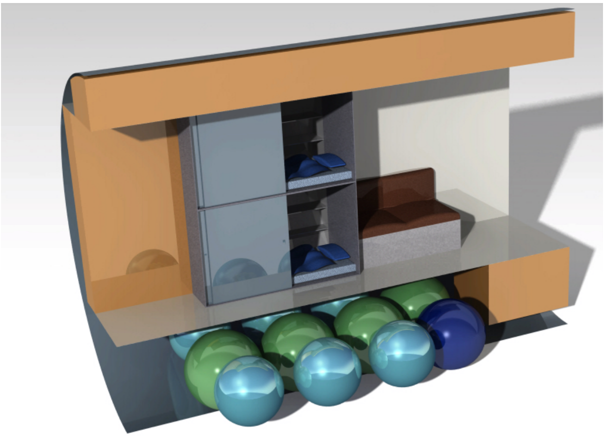


(a) Module 1, contains the ECLSS, med-bay, bathroom & laundry and a variety of storage space

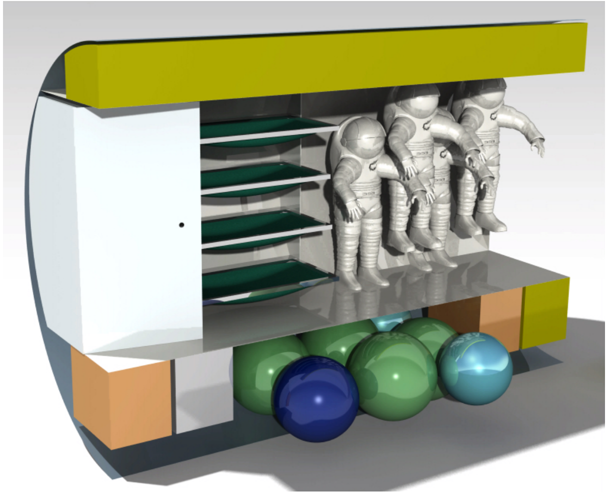


(b) Module 2, contains the exercise equipment and a generous amount of storage space

Figure 2: Renders of modules 1 and 2



(a) Module 3, the astronaut living compartment



(b) Module 4, the ECM. Also contains the galley, central server and other essential systems for standalone operation

Figure 3: Renders of modules 3 and 4

The process of getting the hardware and astronauts to the moon and installing the base has also been thoroughly investigated. Based on an astrodynamics model using third body perturbations, it was decided to use a Hohmann transfer to bring payloads to the moon. The cylindrical modules will be launched to Low Earth Orbit by Falcon 9 rockets and rendezvous with a transfer vehicle launched by a Falcon Heavy rocket. From there, the transfer vehicle brings the payload to Low Lunar Orbit, where the two systems are disconnected, and the payload starts the landing process. The installation of the different payloads on the moon is performed by small rovers dubbed "ants", which collect elements and place them in their designated location. The area needed to transport the modules is already sintered by the regolith rovers before the other launches arrive. Finally, regolith rovers cover the completely installed habitat with Lunar regolith and sinter the top layer. Our calculations show that eighteen launches and nine Lunar landings over a 19-week period are required to bring all habitat components and assembly machines to the Moon.

As this project has never been done before, a high number of risks are identified throughout the duration of the mission. The project requires a high level of reliability engineering to deal with all these risks. Safety and mitigation procedures are in place for all activities that pose a higher risk. Especially the logistics and operations of this project are critical cases that need extensive design work. Given that this topic will be developed into further detail during future development, the current project risk level is considered within acceptable bounds. The final estimates of the relevant technical resource budgets are listed in table 2. These include the necessary contingencies on the calculated values.

Table 2: Mission technical budgets

Mass $[kg]$	**Cost** $[M€]$	**Pressurised volume** $[m^3]$	**Habitable volume** $[m^3]$	**Power generated** $[kW]$
87309	29802	423	226	20

To make this project a reality, lots still has to be done during the future development of this project. As a first, the rovers are at a TRL of 1. This means the complete design of this part of the project will be done during future development. Also, a KRUSTY information campaign will be held to change the public opinion towards the power plant. This shows that during the next phases, social sustainability is of more importance than environmental sustainability. How the future development will take place is already planned, with the first phases concentrated on the design itself and further phases more concentrated on the testing and usage of the design.

To finalise this design, the requirements stated in the beginning of the project are checked for compliance with the actual design. In conclusion, five of the 137 requirements were not met, and seven of the 137 requirements were not yet met, as the design was not developed far enough to meet the specific requirement. For all of the five requirements that are not entirely met, either a good reason is given for not meeting the requirement, or the requirement was changed as the requirement itself was not correct.

List of Acronyms

ACRONYM	FULL NAME
AC	Alternating Current
ACLS	Advanced Closed Loop System
APH	Advanced Plant Habitat
ARV	Astronauts Rover Vehicle
ATCS	Active Thermal Control System
AWP	Alternative Water Processor
BFR	Big Falcon Rocket
BHN	Brinell Hardness
BIRD	Battery□operated Independent Radiation Detector
BLE	Ballistic Limit Equation
BMD	Benchmark Dose
CAD	Computer Aided Design
CCA	CO_2 Concentration Subsystem
CDH	Command & Data Handling
CPAD	Crew Personal Active Dosimeter
CRA	CO_2 Reprocessing Subsystem
COTS	Commercial Off-The-Shelf
DSE	Design Synthesis Exercise
EAFTC	Environmentally Adaptive Fault Tolerant Computing
ECLSS	Environmental Control and Life Support System
EDS	Electrodynamic Dust Shields
EOL	End-of-Life
EPS	Electrical Power System
ESA	European Space Agency
EV	Extra-Vehicular
EVA	Extra-Vehicular Activity
FAA	Federal Aviation Authority
FBD	Functional Breakdown Diagram
FFD	Functional Flow Diagram
FH	Falcon Heavy
FMEA	Failure Mode & Effect Analysis
FNS	Fast Neutron Spectrometer
GCR	Galactic Cosmic Radiation
GEO	Geostationary Earth Orbit
GNP	Gross National Product
GTO	Geostationary Transfer Orbit
GUI	Graphical User Interface
HERA	Hybrid Electronic Radiation Assessor
HGA	High-Gain Antenna
IBDM	International Berthing & Docking Mechanism
ICT	Information & Communication Technology
ISRU	In-Situ Resource Utilisation
ISS	International Space Station
IV	Intra-Vehicular
JSC	Johnson Space Centre
KRUSTY	Kilowatt Reactor Utilising Stirling Technology
LED	Light-Emitting Diode
LEO	Low Earth Orbit
LGA	Low Gain Antenna
LLO	Low Lunar Orbit
LTO	Lunar Transfer Orbit
LRV	Lunar Roving Vehicle
LVS	Lander Vision System
MEM	Meteoroid Engineering Model
MLI	Multi-Layered Insulation
MMOD	Micro-Meteoroid and Orbital Debris
MMS	Method of Manufactured Solutions
MNS	Mission Need Statement
MPT	Miniaturised Particle Telescope
MULASSIS	Multi-Layered Shielding Simulation Software
NASA	National Aeronautics & Space Administration
NPP	Nuclear Power Plant
NDL	Navigation Doppler Lidar
OGA	Oxygen Generation Subsystem
PTCS	Passive Thermal Control System
PDE	Partial Differential Equation
PNP	Probability of No Penetration
PVDF	Polyvinylidene Fluoride
POS	Project Objective Statement
PR	Public Relations
PTCS	Passive Thermal Control System
QJA	Quest Joint Airlock
RAD	Radiation Assessment Detector
RAM	Radiation Area Monitor
RAMS	Reliability, Availability, Maintainability & Safety
R&D	Research & Development
RR	Regolith Rover
RPN	Risk Priority Number
RTG	Radioisotope Thermoelectric Generator
S/C	Spacecraft
SCM	Safety & Command Module
SCMLS	Safety & Command Module Life Support
SLS	Space Launch System
SNR	Signal-to-Noise Ratio
SOI	Sphere of influence
SWOT	Strengths, Weaknesses, Opportunities & Threats
TLI	Trans Lunar Injection
TRL	Technology Readiness Level
TRN	Terrain Relative Navigation
TV	Transfer Vehicle
US	United States
UV	UltraViolet
V&V	Verification & Validation
WRS	Water Recovery System

List of Symbols

ROMAN SYMBOL	FULL NAME	UNIT
A_1	Area of heat-radiating object	m^2
A_2	Area of eluminated side of an object	m^2
A_h	Footprint area of habitat	m^2
A_n	Footprint area of nuclear fission plant	m^2
c_p	Specific heat capacity	J/kgK
c_1	Constant for the micrometeoroid flux	$-$
c_2	Constant for the micrometeoroid flux	$-$
c_3	Constant for the micrometeoroid flux	$-$
c_4	Constant for the micrometeoroid flux	$-$
c_5	Constant for the micrometeoroid flux	$-$
C_t	Speed of sound in the target	km/s
D_M	Distance between Earth and Moon	m
D_r	Relative density of regolith	$-$
d_p	Projectile diameter	m
d_z	Depth of loaded regolith surface	m
E_s	Source of energy	J
e_r	Average void ratio of regolith	$-$
$F(E_s)$	Energy-based Lunar flux	$impacts/m^2/s$
$F(m, r_0)$	Mass-based Lunar flux	$impacts/m^2/s$
F_S	Solar flux	W/m^2
$F_{E,IR}$	Infrared flux of the Earth	W/m^2
G_L	Specific gravity of Lunar soil	$-$
g_0	Gravity on Earth	m/s^2
I_{sp}	Efficiency of a rocket engine	s
k	Constant for isotopic flux	$-$
K	Damage parameter for titanium alloy	$-$
K_{al}	Material constant aluminium	$-$
k_t	Thermal conductivity	$W/(mK)$
M_0	Initial mass	kg
M_1	Final mass	kg
m	Mass	kg
m_{tank}	Tank mass	kg
$N(m, r_0)$	Spatial density distribution	$-$
n_r	Average porosity of regolith	%
p	Number of nodes in the mesh in y-direction	$-$
P	Gas pressure	Pa
P_c	Critical gas pressure	Pa
P_E	Radiated power of the evaluated body	W
P_reg	Pressure of regolith	Pa
P_S	Radiated power of the Sun	W
r	Number of nodes in the mesh in x-direction	$-$
r_0	Radius of distance from the Sun	AU
r_E	Radius of the Earth	m
R	Universal gas constant	J/kgK
R_{Atmos}	Reliability of atmospheric regulation system	$-$
R_{CDH}	Reliability of command and data handling system	$-$
R_{Comms}	Reliability of communication system	$-$
$R_{Installation}$	Reliability of installation	$-$
$R_{Launcher}$	Reliability of launcher	$-$
R_{Lander}	Reliability of lander	$-$
R_{MMP}	Reliability of micrometeoroid protection	$-$
$R_{Operation}$	Reliability of operation	$-$
R_{Power}	Reliability of power system	$-$

$R_{Radiation}$	Reliability of radiation protection	$-$
R_{Rover}	Reliability of rover	$-$
R_{SCM}	Reliability of SCM	$-$
$R_{Structures}$	Reliability of structure	$-$
R_{System}	Reliability of system	$-$
R_{Tank}	Radius tank	m
$R_{Thermal}$	Reliability of thermal system	$-$
R_{TTM}	Reliability of transfer to Moon	$-$
R_{TV}	Reliability of transfer vehicle	$-$
t	Time	s
t_{tank}	Tank wall thickness	mm
T	Gas temperature	K
T_c	Critical gas temperature	K
T_E	Temperature of the evaluated body	K
T_S	Temperature of the Sun	K
$T_surface$	Temperature of the surface	K
u	Temperature	K
x	Position in the mesh in x-direction	m
y	Position in the mesh in y-direction	m
z_r	Depth of Lunar regolith	m

GREEK SYMBOL	**FULL NAME**	**UNIT**
α	Thermal diffusivity	m^2/s
$\alpha_a b$	Absorptance of a gray body	$-$
ϵ	Emittance of a gray body	$-$
γ_1	Exponent for the micrometeoroid flux	$-$
γ_2	Exponent for the micrometeoroid flux	$-$
γ_3	Exponent for the micrometeoroid flux	$-$
γ_4	Exponent for the micrometeoroid flux	$-$
γ_5	Exponent for the micrometeoroid flux	$-$
λ_p	Aspect ratio projectile	$-$
$\tilde{v}_m$	Average velocity of a meteoroid	m/s
ρ	Density	kg/m^3
ρ_{al}	Density of aluminum	kg/m^3
ρ_p	Average density of meteoroid	kg/m^3
ρ_r	Average density of regolith	kg/m^3
ρ_s	Density of dry sand	kg/m^3
ρ_{Ti}	Density of Titanium	kg/m^3
σ	Stefan-Boltzmann Constant	$Js^{-1}m^{-2}K^{-4}$
σ_{Ti}	Yield stress Titanium	MPa

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1 Mission Introduction

The purpose of this chapter is to introduce the mission and outline its main objectives. In fact, the mission need and project statements are provided in section 1.1, as well as an investigation into the primary user requirements that constrain the project in section 1.2 and the structure with which this report is organised.

1.1. Project Objectives

It is of utmost importance for the team and future readers that the 'mission need statement' (MNS) and the 'project objective statement' (POS) are defined and mentioned explicitly. With this properly achieved, the driving factors will stay clear throughout the entire duration of the project.

- **Mission need statement**: *"Analyse, design and build a safe, living and working environment for sustained manned Lunar activities in order to improve the chances of long-term human survival"*;
- **Project objective statement**: *"Design a semi-permanent living and working habitat for four astronauts on the surface of the Moon within eleven weeks with a team of ten students to contribute to European Space Agency's (ESA) research on extraterrestrial living possibilities of humankind"*.

1.2. User Requirements

Designing a product that will accomplish the POS is the preeminent goal of the team. To do so, an appropriate analysis of the users involved is needed, leading to a list of user requirements. It is vital to clearly identify what these are, as they drive the mission design and shall be considered at every stage of the project. They are found in the list of user requirements in appendix B.

1.3. Project Structure

Having defined the POS, a clear and concise structure for this project can be established. The design process will be as follows:

- Investigate the current market of the product, its opportunities and the available resources;
- Identify all functions that the product shall accomplish and in which order;
- Generate possible concepts for the product;
- Analyse the generated design ideas utilising qualitative and quantitative trade-offs;
- Verify the results via a sensitivity analysis;
- Investigate the final environment of the product and its challenges;
- Clearly define the design of all subsystems of the product;
- Integrate the system, verify, and validate the outcome;
- Develop a logistics and operational plan for the product;
- Assess the risks, reliability, availability, maintainability, and safety of the product;
- Perform an analysis of the future development;
- Draw appropriate conclusions on the process.

This report will follow this pattern, although the detailed version of the process up to the final concept selection can be found in the Baseline Report [2] and the Midterm Report [3]. It is worth mentioning that iterating is part of any design process, and it is done at any stage of the assignment, whenever necessary.

2
Market Analysis

The first step of designing a new product or service is to investigate the potential market characteristics. Hence, an introduction to the market and a possible market gap are identified in section 2.1. Secondly, a number of similar missions to the one being designed in this project are investigated in section 2.2. Thirdly, the possible values and opportunities that this pioneering mission could bring to the market are discussed in section 2.3. Finally, the available funding possibilities are described in section 2.4. To combine all the information regarding the market, an analysis of strengths, weaknesses, opportunities and threats of the product has been performed in section 2.5.

2.1. Market Gap

The space sector has had a huge development over the last decade, with the number of space agencies increasing enormously in the past ten to fifteen years. In fact, the emerging space companies established per year rose from less than ten in the early 2000s to almost sixty in 2016 [4]. This is a clear proof of a global increase in public interest in space activities and explorations. However, besides the space-giants that dominate the business, most of these companies are still in a research and development (R&D) phase and the required technologies are still being improved.

With this being said, a conclusion can be drawn on the fact that this field is still very much under progress. In particular, the exploration of the Moon is currently more frequently investigated by space agencies. Although we know a lot about this celestial body, there is a clear and unique market gap to be filled. There is simply no similar product to the one being investigated in this project. There have been several enterprises and other universities that have investigated similar possibilities of building a Lunar habitat. However, since this has never been realised, the market gap remains open.

2.2. Similar Missions

Even though there is no existing Lunar habitat and this is a pioneering project, we do have some similar missions which shall be taken into account throughout the design process: in particular, the Apollo program and the International Space Station (ISS). The investigation into these is crucial as it provides an overview into what has already been done, what challenges were faced, along with providing information in what may be replicable or improved upon for the Lunar habitat mission.

2.2.1. Apollo Program

The Apollo program is renowned and was a product of political tensions between the United States (US) and Russia. America's National Aeronautics and Space Administration (NASA) had access to a substantial budget: nearly 140 billion in current-day dollars according to Stine [5] (118 billion in 2008 dollars, adjusted for inflation). This might seem over budgeted, however, there was no hands-on experience and practical information from and comparable missions.

The Apollo program is the first and only mission that successfully sent a group of astronauts to the Moon and back in 1969. Despite the fact that most technologies used in the Apollo program are rather outdated, the technology developed for the landing and ascent of manned vehicles on another celestial body is still very valuable. Furthermore, an incredible amount of studies into the landing site characteristics have been performed in the last decades, which are used in this report where applicable.

2.2.2. International Space Station

The ISS is an international collaboration between NASA, Roscosmos, ESA, Japan and Canada. It is the largest single structure humanity has ever put in space, housing astronauts continuously since November 2000. The budget that was made available for the construction of the ISS is different for each party: NASA is by far the largest contributor having spent well over $100 billion, with a total cost approaching $160 billion [6]. The business model that drives contributions for the ISS is fairly straightforward: astronaut- and research time is allocated to the agencies according to expenditures, which is a strategy that could be copied for the Lunar Habitat project.

2.3. Added Value

A gap in the market has been identified. The reason why it should be filled shall be analysed too. What is the value added to the market with the implementation of this product? This question will be addressed firstly by looking into what similar missions brought and then by predicting future market opportunities of a Lunar habitat.

Both the ISS and the Apollo programs largely contributed to an economic growth. In fact, it has been proven that technological advances have a substantial effect on the prosperity of the market. Historically speaking, a good dose of social development on a global scale is attributed to either scientific or technological innovations and to the abilities derived from their expansion, as described by Benaroya [7] in his book ”Building Habitats on the Moon”.

Neil Armstrong was right: ”a giant leap for mankind”. The Apollo program was much more than just a small step, but an incredible opportunities-generator on a practical level. The popular shoemaker AVIA Inc. adopted the foam materials used in the Lunar boots to improve shock absorption and provide superior stability and motion control during physical activities [8]. The advances in structural materials were also enormous; a particular fabric has been created based on the Apollo programs, with which the roof of Houston’s Reliant Stadium is constructed. This permanent structure fabric is stronger than steel and incredibly light, maintains the natural grass of the field and decreases lighting, cooling and maintenance needs, as well as moisture and deterioration [8]. The remote and harsh environment of both the Moon and the ISS triggers the necessity of advanced mechanisms with regards to communications, safety measures, health assurance and a lot more. On an energy-related note, it is interesting to understand that solar panels, which are widely used nowadays, are derived from the Apollo programs. In fact, these were an implementation from NASA to counteract the extreme amounts of energy needed to power such a mission. The ISS brought into the game immense advances related to medicine and scientific researches, such as the first automatic heart monitors, essentially a smaller version of the defibrillators currently used in any hospital. These are just a few of the developments that these missions introduced in the market and discussing all of them would be outside the scope of this project.

Benaroya [7] believes that ”proponents of a manned and permanent return to the Moon have expressed the belief that over time, such activity can propel the Earth economy to a higher level of activity due to federal and industrial expenditures in high-tech research and development”. Also the added values that would follow from the creation of a Lunar habitat are more than extensive: generate demand of products, increase Earth’s economy, further develop technological advancements, set the base for future expansions to Mars and beyond, investigate extraterrestrial materials for various applications, survive as humans in the radioactive, low, and micro-gravity space environment, improve the chance of long-term survival, provide a base for nuclear weapons to destroy near-Earth asteroids, contribute to research in geology, astronomy and other sciences and an infinite more [7]. The technology developed for space industries is thus often used for the development in other sectors. Possibilities could be:

- Research on long duration micro-g space missions, as this mission can be used for the research of health care and sport medicine [7];
- The mission can give insights in implementing health care at very remote locations or when staying in underdeveloped communities [7];
- The mission shall provide insight into advanced processes for water and waste management [7];
- The findings of the mission could aid in building in extreme conditions, i.e. possibility in Antarctica;
- The mission’s findings could supply knowledge in designing self-sustainable living communities;
- The mission could possibly give insight into designing houses which can be assembled much quicker.

The options are not limited to the ones listed above. Developments in space industry are going fast, and new markets are expected to continuously emerge in the near future. These are all consequences that are likely to emerge with a Lunar habitat, and reasons why this product would be hugely beneficial to the market.

2.4. Funding Opportunities

Big projects require large funding, and large budgetary needs are part of this particular assignment. Hence, potential investors shall be analysed and evaluated. Furthermore, a brief investigation into nuclear technologies is conducted to determine if this would have an influence on the stakeholder decisions to contribute to this project or not.

2.4.1. Stakeholder Analysis

From space agencies to private companies, including research institutions and universities, there are many parties that could be considered for funding this mission. However, since the estimated costs of this project are so high, no such enterprise alone would be able to finance the entirety of the mission. Hence, just like for the ISS, a collaborations between entities is more likely. What follows in table 2.1 is a breakdown of the principal public and private space companies, their budgets and their willingness to collaborate with other space agencies based on past experiences. The main objective of this stakeholder analysis is to find opportunities to obtain funding from multiple sources. As research into Lunar habitation is being conducted by a whole range of parties, universities and national research institutes alike, it would be vital to identify institutes that may be willing to either support the project budget-wise or conduct outsourced research. As universities will not support the project budget-wise, they could only be contacted for research.

Table 2.1: Public and private space entities, their budgets and collaboration possibilities

Space entity	**Budget** [*$ billion*]	**Collaboration possibilities**
		Space agencies - public sector
ESA	6.9	ESA is interested in the possibilities of a Lunar basis on the Moon and has collaborated with other parties before [9]
NASA	20.7	NASA is the first space agency that landed on the Moon and already has a plan for a permanent Lunar basis in 2040 [10]
ROSCOSMOS	2.46	ROSCOSMOS has a small budget since the fall of USSR and has no known plans to go to the Moon [11]
JAXA	2.03	JAXA is working with ISRO on a mission to the Moon and is prone to collaborate. It wants to have a working Lunar basis in 2025 [12]
CNSA	1.3	CNSA is planning on unmanned missions to the Moon, but is reluctant to collaborate with other parties [13]
ISRO	1.4	ISRO is working with JAXA on a mission to the Moon and is prone to collaborate [14]
		Space companies - private sector
Airbus Space and Defence	332	Airbus supplies and develops technology needed for the mission and has collaborated with ESA before
Boeing Space and Defence	819	Boeing works with NASA on designing human Lunar habitation
Lockheed Martin	1000	Lockheed Martin works with NASA to design a Lunar human habitation
SpaceX	-	SpaceX is developing a human colony on Mars, collaborating with NASA
Blue Origin	-	Blue Origin is interested in having people living and working in space, although is not actively collaborating with any other space entity

2.4.2. Nuclear Technologies

Nuclear technology has the capability to provide solutions to many existing problems in spacecraft design. However, it also carries a stigma because of the various incidents that have happened over the last 50 years.

According to Downey *et al.* [15], the modern public is risk averse, especially concerning nuclear technology, and requires compelling reasons for its use to not rise up in protest. Given that the carte-blanche for the development of nuclear technology has been withdrawn, any new nuclear technology should be handled with "political" care in order to garner popular support.

Launius [16] does find a trend of decreased public protest against space nuclear power, which he attributes to the success of safety programs, efforts to increase public understanding of the technology and the rarity of use. He does state that advancing the future of nuclear space power will not happen without difficulties, especially if they involve more "aggressive" efforts. Realisation of these systems will only be possible with significant societal input.

It is interesting, however, to note that most potential investors mentioned in subsection 2.4.1 have some form of nuclear technology development program themselves. NASA is even working their own new reactors, and most space agencies use RTG power systems for small deep space satellites.

Reflecting on these studies, it becomes apparent that selection of nuclear power should be done with care and only in cases of absolute necessity, unless a thorough campaign to increase safety and public awareness is performed. Even though potential partners might not be scared away by nuclear technology, without popular support they might be turned away due to potential loss of face.

2.5. SWOT Analysis

This section contains an overview of the strengths, weaknesses, opportunities and threats (SWOT) of the current and future markets. The main positive points that have been identified are the recent development of the commercial space sector and the growing interest in Lunar explorations by space agencies. On the other hand, the main disadvantages lie in the high costs of the mission, combined with the little opportunities to generate profits and the extensively challenging technologies. The complete SWOT analysis is shown in figure 2.1.

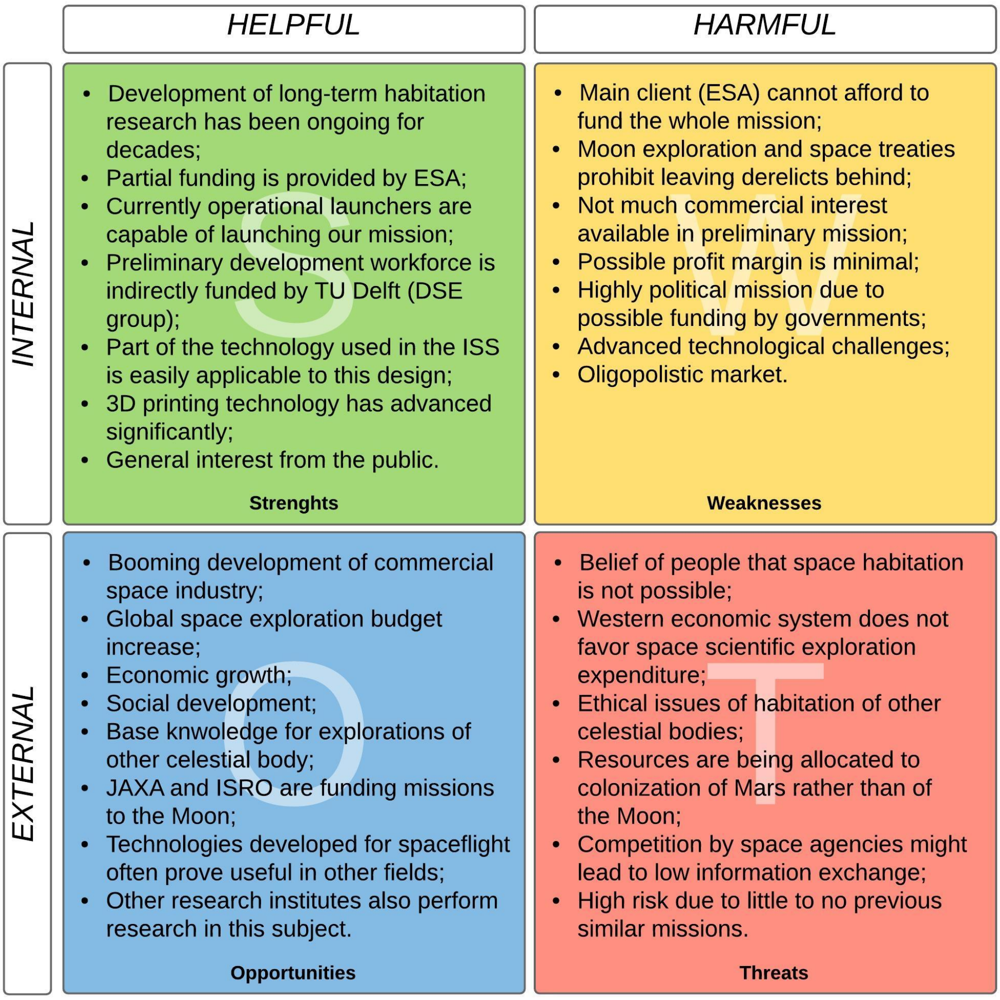


Figure 2.1: Market SWOT analysis

2.6. Concluding Remarks

Given the financial success of previous missions such as the Apollo or ISS program, and the tremendous opportunities a Lunar Habitation mission brings, both in the technological and societal sense, the proposed mission has the potential to open up whole new markets. Space agencies such as NASA and ESA, and commercial parties such as SpaceX and Boeing make-up but a few of the potentially interesting partners for the project. All in all, a Lunar habitation mission can be even more successful than the ISS from a market perspective.

3
Functional Analysis

In order to obtain a complete overview of all the functions that shall be performed by the system, the team performed a detailed functional analysis. This consists of two parts: the functional flow structure described in section 3.1 and the functional breakdown structure in section 3.2.

3.1. Functional Flow Structure

As a first step for creating the functional flow diagram (FFD), the high-level functions of the entire mission are distinguished. The team identified six major stages, which will form the first level of the structure:

FF 1: Design complete product;

FF 2: Produce product;

FF 3: Transfer product to the Moon;

FF 4: Install product;

FF 5: Operate product;

FF 6: Ensure EOL sustainability of product.

The purpose of the FFD is to identify the sub-tasks from each main stage of the project. The first and second levels of the FFD are presented in figure A.1. For easiness of identification purposes, every block of the diagram is labelled with the prefix ”FF”. Furthermore, for clarification's sake it is worth noting that the term ”product” refers to the whole mission, from the conceptual design phase of the habitat to its end-of-life (EOL). If a task needs reviewing, a feedback loop is created. In the diagram, this is indicated with a capital $\bar{G}$, whereas a simple G indicates a ”go” action.

For this report, it has been established that the overall FFD shall be developed only up to the second level. The more detailed sub-levels will be provided in the respective chapter sections. The diagrams for each subsystem will be given in chapter 8, whereas the flows regarding the operations and logistics in chapter 11.

3.2. Functional Breakdown Structure

In parallel to the FFD, the functional breakdown diagram (FBD) has been developed. Its purpose is to identify all system functions, as passive functions might not appear on a flow diagram. Four milestones found in the FFD are selected based on the product having to ”do” something:

FB 3: Transfer product to the Moon;

FB 4: Install product;

FB 5: Operate product;

FB 6: Ensure EOL sustainability of product.

These will represent the first level of the product's functions. FB 3, 4, and 6 have been developed to second and third levels in figure A.2, whereas FB 5 is broken down into figure A.3 and A.4 because of it being intensely functional. Because the functions performed by the product may be system- or human-centred, this distinction has been identified. These diagrams have been used to extensively tackle the design process of the whole mission. Again, for easiness of identification purposes, every function of the diagram is labelled with the prefix ”FB”.

FB 3 to 6 happen in chronological order, hence they shall be analysed and investigated accordingly. However, just like in any design process, a process of iteration is applied. For example, the design of the communication subsystem affects the power budget, and this in turn affects all subsystems of the habitat, including the communications subsystem in its turn. Hence, several functions are interlinked and shall be constantly reviewed as the design process moves forward.

4
Concept Selection

Having thoroughly defined all functions that together form the mission, it is time to generate possible design concepts and perform a trade-off to identify their strengths and weaknesses. This chapter treats this. The structure follows from the general approach of the concepts selection process as described in section 4.1.

4.1. General Approach

To select a final concept, the following process is determined. First, at an early stage of the project, five Strawman concepts were generated. After a first, purely conceptual qualitative trade-off based on a preliminary estimation in terms of orders of magnitude, two of these were disregarded because they were considered unfeasible. The three remaining concepts were then analysed further. At this point, a second trade-off could be performed, where a final concept was chosen. This converging flow is graphically outlined in figure 4.1.

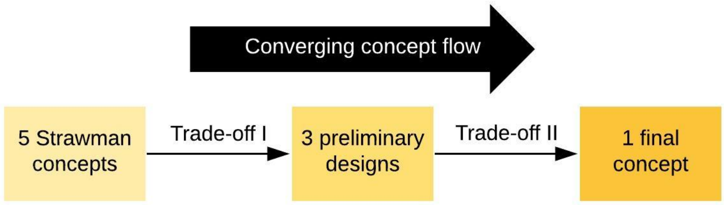


Figure 4.1: Work flow converging to the optimal concept

4.2. Strawman Concepts

To begin with, the general characteristics of the five Strawman concepts were defined. Further information on the designs can be found in the Baseline Report [2].

Concept 1 - Reusable Lander Base: using the landing vehicle as the habitat itself, by interconnecting various landers together;

Concept 2 - Inflatable Dome: a habitat consisting of an inflatable dome, stemming out of a rigid cylinder;

Concept 3 - Rigid Cylinder: transporting rigid pre-fabricated cylinders from Earth, placing them horizontally and interconnecting them;

Concept 4 - Build-in Design: constructing an underground dome by means of inflating a structure, either using a crater/cave or digging a hole;

Concept 5 - Honeycomb Design: constructing a series of domes out of hexagonal tiles.

4.3. Trade-off I

The first trade-off conceptually assessed these concepts based on seven criteria: safety, mass, reliability, transportability, expandability, volumetric efficiency and ease of installation. Differently from how trade-offs are usually performed, no weights were given to any of these criteria, as the main purpose is to simply disregard unfeasible options. Furthermore, there is no sufficient knowledge on which factors are the most critical ones for this mission. Hence, the design ideas were qualitatively scored as 'outstanding' (O), 'acceptable' (A), 'marginal' (M) or 'unfeasible' (U) for each criterion. The scoring was assigned following a set of guidelines that was identified in the Midterm Report [3]. For example, with regards to safety, two of the guidelines were "Time to reach the emergency room is less than 15 seconds" and "Multiple exits are present". Analysing each concept based on the provided guidelines resulted in the scoring shown in table 4.1.

Table 4.1: Trade-off I matrix

		Criterion						
		Safety	**Mass**	**Reliab.**	**Transp.**	**Expand.**	**Vol. eff.**	**Inst.**
Weight		-	-	-	-	-	-	-
Concept	**1**	A	A	A	A	A	O	O
	2	M	A	M	A	O	M	O
	3	A	A	A	A	O	O	A
	4	O	M	U	M	M	M	U
	5	M	M	U	M	M	A	M

Note: Reliab.: reliability, Transp.: transportability, Expand.: expandability, Vol. eff.: volumetric efficiency, Inst.: installation

For the purpose of this trade-off, any criterion that was considered as 'unfeasible' would imply the discarding of that design. This lead to the elimination of concepts 4 and 5. With regards to the Build-in Design, the installation was simply unfeasible as digging deeper than $1m$ on the Moon would require heavy machinery to dig in the enormous amounts of Lunar regolith and its underlying rock to facilitate the construction of such a base. For the Honeycomb Design, very complex installation equipment would be required, complicating even more the feasibility of an already complex base and drastically reducing its reliability. However, the good points of the disregarded concepts shall still be considered for possible merging into one optimal final design. For example, concept 4 could lead to the idea of having an underground safety room in case of emergency situations. These implementation shall be considered in chapter 6, where the final layout of the Lunar habitat will be discussed and finalised.

4.4. Preliminary Designs

Following the second trade-off, a more detailed investigation on the remaining concepts was undertaken. In this design phase, the team decided to focus on the following aspects: external layout of the Lunar habitat, mass budget, number and cost of launches, technology readiness level (TRL) and installation plan for each concept. Firstly, the external layouts were determined. These are schematically displayed in figure 4.2, 4.3, and 4.4 for concepts 1, 2, and 3 respectively.

With regards to the mass estimations, a first-order structural model was developed. This aimed at determining the general wall thickness of a cylindrical structure under pressure, thermal stress and launch loads. The critical load case for this mission was determined to be the axial loading during launch, since the crippling buckling stresses were the constraining factor for thickness of the cylinder walls. The masses coming from the structural model were validated by means of a parametric estimation based on the volume-mass relation. As the subsystems such as life support and communication were assumed to be the same for all concepts, the structural mass was considered to be the greatest driver in mass budget that a concept could have over another.

Sizing of the preliminary designs was done according to the requirements regarding minimum living volume and area, which meant that also a preliminary internal layout had to be designed. Hence, the packing, transportation to the Moon and installation plans could be set up, from which in turn the number and cost of launches required was calculated. Lastly, the various TRLs were estimated, as this would give an indication on the level of reliability per concept. A summary of these characteristics for each of the three remaining designs is given in table 4.2.

Table 4.2: Summary of the characteristics of concepts 1, 2, and 3 in the preliminary design phase

Characteristic	**Unit**	**Concept** 1	2	3
Total system mass	kg	39000	34000	39000
Technology readiness level	-	5	3	5
Total number of launches	-	10	5	10
Number of Falcon 9 launches	-	4	-	4
Number of Falcon Heavy launches	-	6	5	6

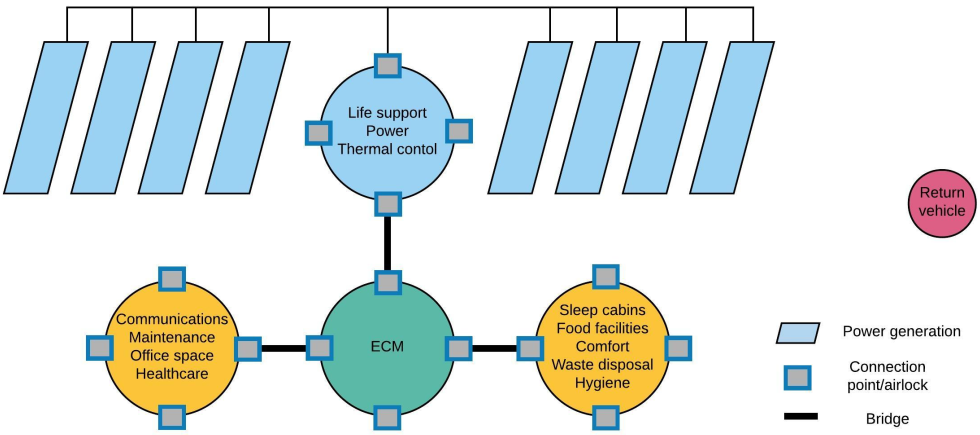


Figure 4.2: Preliminary external layout of concept 1 - Reusable Lander Base

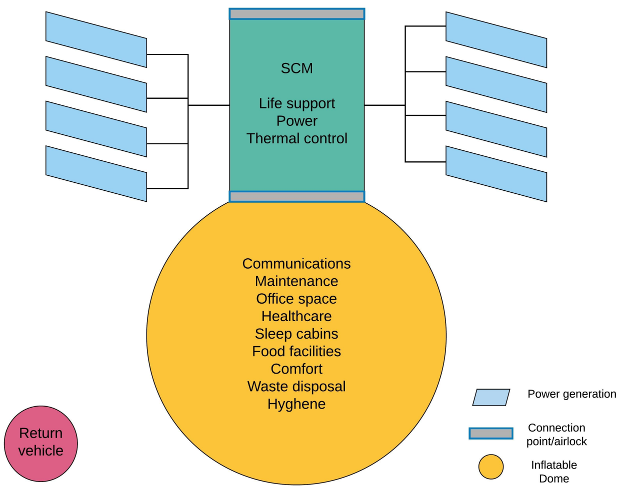


Figure 4.3: Preliminary external layout of concept 2 - Inflatable Dome

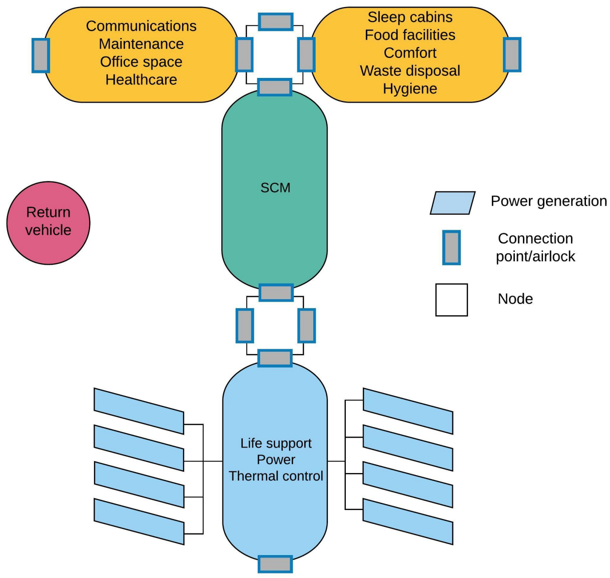


Figure 4.4: Preliminary external layout of concept 3 - Rigid Cylinder

4.5. Trade-off II

With the design basis established and the concepts worked out further to identify relevant differences, the second trade-off could be made. This is a modified version of the first one, where the criteria have been improved to better cover the important aspects of the designs and to analyse them on a more in-depth level. These were chosen to be safety, total system mass, sustainability, TRL, and launch cost.

The weights of the criteria were distributed based on the outcomes of both a risk analysis and a reliability, availability, maintainability, and safety (RAMS) study for the mission. Safety remains the driving design factor with highest priority together with mass. They are also the only ones directly related to one of the possibly constraining user requirements. Hence, their weights are chosen to be 30/100. Sustainability is a very broad term, but in this case it consists of environmental and social sustainability, transportability, expandability, volumetric efficiency, and ease of installation, with a total weight of 20/100. Subsequently, TRL is directly related to reliability of a system and is considered to be ranked third in terms of priority, with a weight of 15/100. Lastly, the total launch cost for each concept is considered an important parameter. This is because it was determined that different launchers shall be used for different concepts, altering the total cost of the mission. However, this is directly related to both mass and TRL, for which preliminary estimations have been already performed, so it was given a weight of 5/100.

Differently from trade-off I which followed a purely qualitative approach, here the concepts were quantitatively scored on a scale from 1 to 5, based on several guidelines established specifically for each criterion. Then, a weighted average out of 100 was calculated, from which a final concept could be chosen. The scoring of trade-off II can be found in table 4.3.

Table 4.3: Trade-off II matrix

		Criterion					Total
		Safety	Mass	Sustainabil.	TRL	L.c.	
Weight		**30**	**30**	**20**	**15**	**5**	**100**
Conc.	**1**	3	4	3.7	5	3	74.8
	2	3.4	5	4	3	5	80.4
	3	4.1	4	4.4	5	3	**84.2**

Note: Sustainabil.: sustainability, TRL: Technology Readiness Level, L.c.: launch cost, Conc.: concept

Trade-off II resulted in the conclusion that concepts 2 and 3 were the best ones. The Inflatable Dome design, with a total score of 80.4, was determined to be the lightest concept, whereas the Rigid Cylinder design the safest one and with a total score of 84.2. Hence, concept 3 was ultimately chosen. Same as for trade-off I, the elimination of one concept does not imply the complete exclusion of all its design ideas. In fact, the good points of the disregarded concepts shall still be considered for possible merging into one optimal final design. For example, the idea of an inflatable structure could be adopted in concept 3, where a dome could be exploited from one of the rigid cylinders. Again, any possible implementation shall be considered in chapter 6, where the final external layout of the Lunar habitat will be discussed and finalised.

4.6. Sensitivity Analysis

With the purpose of validating the trade-off, a sensitivity analysis was conducted. This confirmed the insensitivity of the trade-off to various variation: firstly, the scores of the concepts were adjusted by $\pm$ 20%, which still resulted in concept 3 being the one with the highest score for 26 out of 30 times; secondly, the weights of the criteria were altered by $\pm$10%, resulting in concept 3 being the one with the highest score in all the different scenarios considered.

4.7. Concluding Remarks

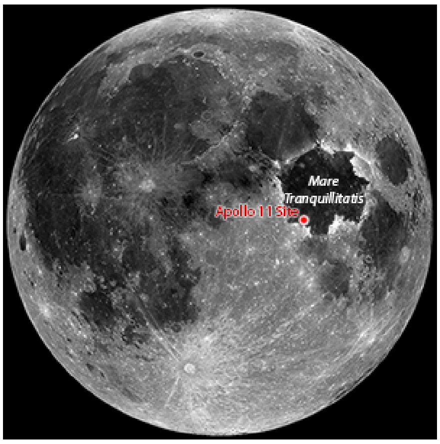
The five initial Strawman concepts have been evaluated by means of a first qualitative trade-off, where safety, mass, reliability, transportability, expandability, volumetric efficiency and installation were the criteria to be considered. The outcome of this trade-off lead to the exclusion of two designs, after which the remaining three have been developed to a preliminary level of detail. A quantitative trade-off could now be performed by evaluating safety, mass, sustainability, TRL and launch cost of the concepts. The outcome of trade-off II resulted in the third concept (Rigid Cylinder) being the one with the better performance based on the chosen criteria. Concept three scored best on Safety, Sustainability and TRL. A sensitivity analysis has been performed for checking the validity of the results. Since the outcomes are validated, concept 3 will be brought forward to the next design phase. The approach from now on is as follows: firstly, the habitat location will be investigated in detail in terms of its topography, composition and general environment. This will set a good basis for designing a final external layout and perform a structural analysis, after which the subsystem design phase can start.

5
Habitat Location

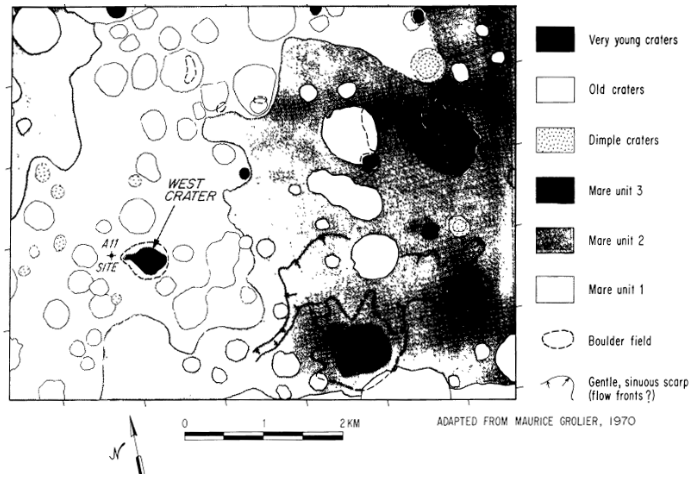
Having determined the landing location of the mission to be that of Apollo 11, as discussed in the Midterm Report [3], this chapter deals with the specific characteristics of this site. Firstly, the local topography and geology of the Mare Tranquillitatis area of the Moon will be presented in section 5.1, followed by the properties and possible applications of the Lunar regolith in the area as discussed in section 5.2. Next, the ionizing, non-ionizing and thermal Lunar radiation environments are discussed in section 5.3, followed by section 5.4 on the Lunar meteoroid and micrometeoroid characteristics.

5.1. Lunar Topography

Captured by the Lunar Reconnaissance Orbiter, figure 5.1a shows the near side of the Moon, including the Mare Tranquillitatis area and the landing site of Apollo 11. The area around this site has been thoroughly mapped in figure 5.1b, where a relatively flat area east of the original location is identified. Studies by Beaty and Albee [17] show that the local area can be divided into three main types of Mare soils based on differences in albedo and crater density. Furthermore, partially based on observations by Grolier [18], these studies state that the different compositions of the ground are potentially caused by separate volcanic flows. Finally, the gentle sinuous scarps shown in figure 5.1b present possible flow fronts.



(a) Near side of the Moon and Mare Tranquillitatis [19]



(b) Local topography of the Apollo 11 landing site [17]

Figure 5.1: Global and local topography of the Apollo 11 site

The local flow regions presented in figure 5.1 allow the identification of three different options for the final habitat installation site. However, samples collected by the Apollo 11 mission were only taken from the Mare unit 1 area, meaning that there is no conclusive evidence that units 2 or 3 have preferable properties. These units can be identified by means of the legend of figure 5.1b. Combined with the fact that the actual Lunar soil of unit 1 has been thoroughly analysed, the location for the habitat installation is set to the flat area situated around 10 km northeast of the Apollo 11 landing site.

5.2. Regolith Properties

As stated in section 5.1, the Mare soil of the Apollo 11 landing site is most likely originated from volcanic flows, making basalts the primary mineral component. Several studies, among which that of Korotev and Gillis [20], have been performed based on the findings of the Apollo missions, confirming the mineral percentages to be 66% crystalline Mare basalt, 5% orange volcanic glass, 20% material of the feldspathic highland, 8% KREEP-bearing impact-melt breccia and 1% meteoritic material. Within these, the size range of particles > 1 mm is dominated by lithic clasts, whereas the size range of those < 1 mm is generally dominated by mineral or glass particles [21]. Analysis of bulk properties of Lunar regolith has provided the values that can be found in table 5.1.

Table 5.1: Material properties of bulk regolith [22]

Depth range	Average density	Average porosity	Void ratio	Relative density
z_r	ρ_r	n_r	e_r	D_r
$[cm]$	$[g/cm^3]$	$[-]$	$[-]$	$[\%]$
0 - 30	1.50	0.52	1.08	74
30 - 60	1.74	0.44	0.79	92

Average bulk density of regolith (ρ_r) changes with respect to depth due to the packing of its particles. Both porosity (n_r) and average void ratio (e_r) are direct or derivative functions of this density value, as shown in equation 5.1 and 5.2, respectively. The formula for the relative density (D_r) is presented in equation 5.3. Furthermore, ρ_w represents the density of water, whereas G_L the specific gravity of the Lunar soil with a value of 3.1. Also, e_{max} represents the maximum void ratio, and e_{min} the minimum void ratio. All equations are obtained from the NASA guidelines by Justh *et al.* [23].

$$n_r = 1 - \left[\frac{\rho_r}{\rho_w G_L}\right] \quad (5.1) \qquad e_r = \left|\frac{n_r}{n_r - 1}\right| \quad (5.2) \qquad D_r = \left[\frac{e_{max} - e_r}{e_{max} - e_{min}}\right] \quad (5.3)$$

5.2.1. Formability of Lunar Regolith

Using equation 5.1 to 5.3 of section 5.2, it is possible to obtain an estimated indication for the formability of the Lunar regolith. In particular, it is critical to quantify the change in surface height due to placement of a multiple ton mass. A relation for the density of Lunar regolith (ρ_r) is found in Carrier *et al.* [22] and presented in equation 5.4, where z_r indicates the depth of the regolith in cm and ρ_r approaches a theoretical maximum of 1.92 g/cm^3.

$$\rho_r = 1.92 \cdot \frac{z_r + 12.2}{z_r + 18} \quad (5.4)$$

Given that a 30 cm layer of regolith already experiences a 0.20 g/cm^3 increase in density, the following assumptions are made:

1. The density of the regolith approaches the theoretical maximum when loaded by a multiple ton mass;
2. The surface pressure caused by the habitat is constant across the contact area;
3. Only the first meter of regolith is considerably compressed.

Following these assumptions, one finds the expression for the depth of the loaded surface of regolith (d_z) as per equation 5.5.

$$d_z = 1 - z_r \cdot \frac{\rho_{min}}{\rho_{max}} \quad (5.5)$$

Plugging in the theoretical maximum from equation 5.4 and minimum density from table 5.1, one finds a depth of 21.8 cm for the loaded scenario. Of course, taking the minimum density for the entire first meter of regolith is a rather conservative assumption. For example, taking the average of the two presented densities leads to a depth of 15.6 cm. To accurately determine the actual depth, a more refined model is highly needed. However, a depth change of 10 cm is likely expected.

Finally, a property of interest, resulting from the previous analysis, is the maximum angle at which regolith can be compacted, which Ruess *et al.* [24] states to be 40°.

5.2.2. Possibility of Sintering Regolith

As discussed in subsection 5.2.1, placing a heavy object on pure regolith may lead to significant sinkage. It is possible to pre-compress the building site by means of a Lunar compactor, but a valuable alternative has been identified in the form of a microwave-treatment, with the goal of sintering the regolith.

Studies into the possible application of this process have been performed by Taylor and Meek [25], concluding that Lunar regolith is the best material for in-situ resource utilization (ISRU) purposes on the Moon. Regolith has unique properties that make it an ideal candidate for microwaving, such as rapid heating rates of $1000°C/min$ to high temperatures ($2000°C$), enhanced reaction rates with faster kinetics and lower sintering temperatures.

Given that actual Lunar regolith is not readily available, simulant materials are currently the only option for testing purposes. According to Taylor *et al.* [26], the following principles should be adhered to when performing tests on these materials:

1. A simulant material is chosen for a specific end goal, based on its properties and required tests to be performed;
2. New simulant materials shall be developed only when existing ones are proven insufficient or unacceptable.

With different tests being readily available by carefully selecting the right simulants, the feasibility of using regolith for ISRU purposes is fairly high. Possible uses of regolith as installation material include:

- Creating a solid protection layer by means of a regolith-covered module;
- Paving roads on the Lunar surface;
- Building entire modules or shells from regolith.

Within the scope of this report, the first two options will be explored in chapter 8 and 11, respectively.

5.3. Radiation Environment

Radiation poses one of the largest threats to the health of the astronauts living in the habitat, so having adequate protection against the multiple types of radiation present on the Moon is a critical part of the design. The different types of radiation encountered during a Lunar mission are presented in table 5.2.

Table 5.2: Radiation types in a space environment [27]

Type	Form	Source	SOI	Frequency	Intensity
Ionizing	Solar particles	Solar events	Outside habitat	Infrequent	High
	Trapped particles	Planetary surface	Localized outside habitat	Constant	Moderate
	GCR	Outside Solar system	Ubiquitous	Constant	Low
Non-ionizing	Technology sources	Human equipment	Inside habitat	Constant	Low
	Solar UV	Solar corona	Outside habitat	Constant	Relatively high

5.3.1. Ionizing Radiation

During a long-term mission, both primary and secondary exposure to ionizing radiation can have profoundly detrimental effects on crew health. As outlined in table 5.2, there are multiple sources for ionizing radiation, whose characteristics are presented in table 5.3. The Solar particles category has been split up into two types, Solar wind and Solar flare particles.

Table 5.3: Characteristics of ionizing radiation on the Moon [28]

Type	Flux $[impacts/cm^2/s]$	Energy $[eV/nucleon]$	Composition
GCR	3	$10^8 - 10^{20}$	Protons, alpha particles, electrons
Solar wind	10^8	$< 4 \cdot 10^3$	Protons, alpha particles, electrons
Solar flare	Up to 1000	10^6 - 10^9	Protons, alpha particles, heavy ions

One of the dangers of the high-energy galactic cosmic radiation (GCR) is the production of secondary radiation, also known as Bremsstrahlung. Aghara *et al.* [29] present an analysis of regolith materials and the produced secondary radiation, and find that > 1 GeV photons are not completely attenuated by a 50 cm layer of Lunar regolith. They also find that photon and neutron production, and thus the contribution of Bremsstrahlung, increases drastically for these high energy photons. A multi-layered approach to the design of radiation shielding is desirable based on their preliminary results.

5.3.2. Non-ionizing Radiation

In terms on non-ionizing radiation, the variance of the ultraviolet (UV) radiation caused by solar cycles is critical for the analysis of the photoelectron sheath. Andrew and Mihály [30] found that the increased photo-emission for solar maxima and flare conditions causes higher charge build-up in grains present in the sheath. Design work on electronics should take into account the maximum charge conditions.

5.3.3. Thermal Radiation

Since the Moon is tidely locked with Earth, a Lunar day differs from one on the Earth: one full day and night cycle takes approximately 27 days. Since there is no atmosphere, the Moon's surface gets heated up by thermal radiation during day time and cools down by emitting heat during night time. This cycle translates in a specific thermal flux on the surface which can heat up to 400 K during day time and cools down to 100 K during night time. This thermal flux is presented in figure 5.2.

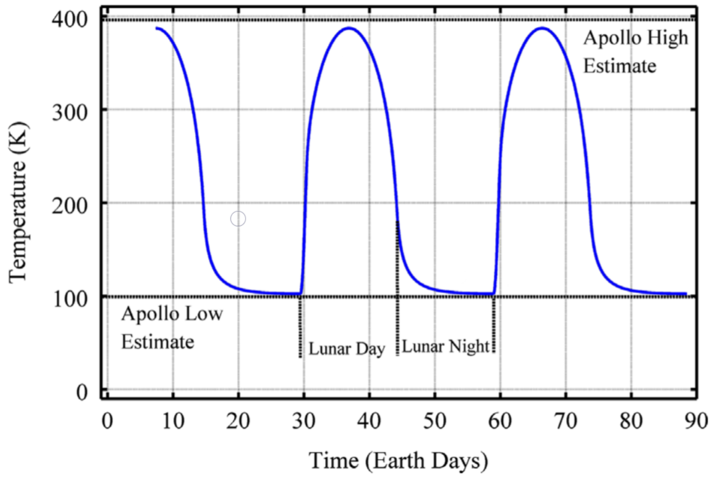


Figure 5.2: Temperature flux on the surface of the Moon [31]

5.4. Meteoroids Investigation

Meteoroid impacts have the capability to cause a catastrophic mission failure. Hence, knowing in advance what kind of impact events can be expected is vital to the success of the mission. To facilitate this, four models of meteoroid flux were consulted, presented in table 5.4 for comparison purposes.

Table 5.4: Meteoroid flux model comparison [32]

Meteoroid model	Year of release [−]	Applicable mass domain [g]	Regime from Sun [AU]
Grün *et al.* [33]	1985	$10^{-18} - 100$	≈1
Divine [34]	1993	$10^{-18} - 1$	0.1 – 20
Staubach *et al.* [35]	1996	$10^{-18} - 1$	0.1 – 20
Dikarev *et al.* [36]	2003	$10^{-18} - 1$	0.1 – 10

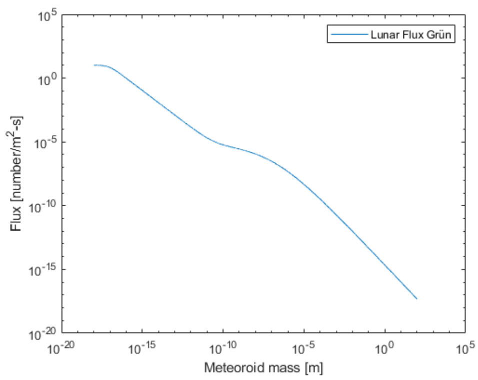
For 1 AU, all models are tuned to the older and updated one by Grün *et al.* [33]. However, they could differ considerably in their directional distribution, velocities and assumed sources. Another limitation of these models is that they only cover smaller meteoroids. For that reason, a different study on large meteoroids by Oberst *et al.* [37] was used to cover the ”higher mass” end-of-impact flux spectrum.

5.4.1. Micrometeoroid Flux at the Lunar Surface

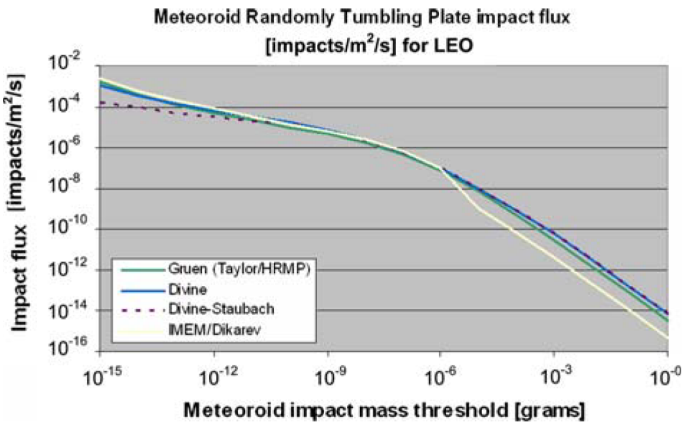
Since only the Grün *et al.* [33] model directly states an analytic relation, it was decided to use this for Lunar flux values. Taking into account measurements by *in-situ* experiments, zodiacal light observations, and oblique angle hyper-velocity impact studies, the so-called Lunar flux (F) for different masses (m) can be obtained as seen in equation 5.6 from Grün *et al.* [33] .

$$F(m, r_0) = (c_1 m^{\gamma_1} + c_2 m^{\gamma_2} + c_3)^{\gamma_3} + (c_4 m^{\gamma_4} + c_5)^{\gamma_5} \qquad (r_0 = 1AU) \tag{5.6}$$

In the above analytic form of the Grün *et al.* [33] model, $c_1 = 4 \cdot 10^{29}$, $c_2 = 1.5 \cdot 10^{44}$, $c_3 = 1.1 \cdot 10^{-2}$, $c_4 = 2.2 \cdot 10^3$, and $c_5 = 15$. The exponents are given by $\gamma_1 = 1.85$, $\gamma_2 = 3.7$, $\gamma_3 = -0.52$, $\gamma_4 = 0.306$, and $\gamma_5 = -4.38$. Plotting this analytic model, figure 5.3a is obtained and verified by means of the remaining three models.



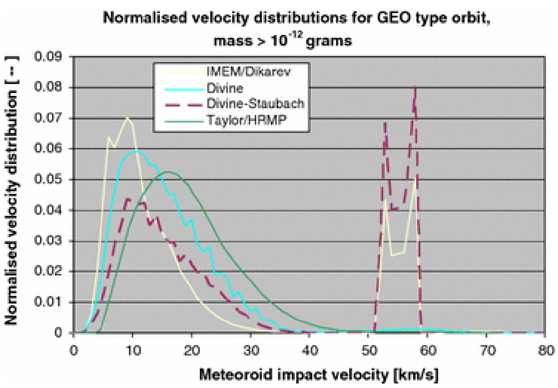
(a) Plot of Lunar flux at 1 AU vs mass spectrum, based on the Grün *et al.* [33] model



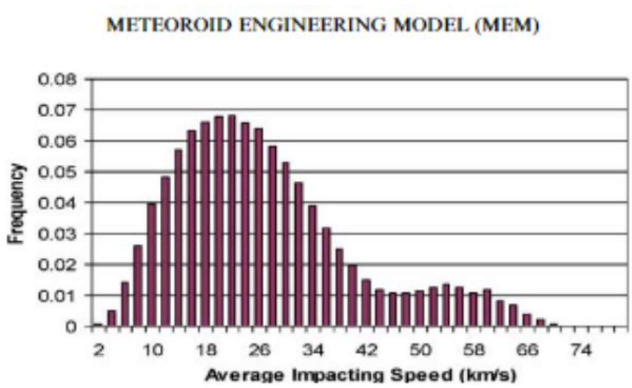
(b) Plot of impact flux for different flux models [32]

Figure 5.3: Comparison of Lunar flux with impact flux at LEO

As stated by Grün *et al.* [33], Lunar flux tends to be two to three orders of magnitude larger than interplanetary flux. Hence, if figure 5.3a and 5.3b are consulted, one finds an impact flux of approximately 10^{-3} $impacts/m^2/s$ and a Lunar flux of approximately 10^{-1} $impacts/m^2/s$ for objects with a mass of 10^{-15} g. These two values differ by the two orders of magnitude mentioned above, verifying the applied version of the model. The Grün *et al.* [33] model only uses an average velocity of 20 km/s, a meteoroid density of 2.5 g/cm^3 and a distance of 1 AU for the meteoroid model. Newer models make use of meteoroid densities ranging between 0.5 g/cm^3 for particle masses of 0.01 g, 1 g/cm^3 for meteoroids between 0.01 g and 10^{-6} g and 2 g/cm^3 for particle masses of 10^{-6} g and less, according to Anderson and Smith [38]. They also contain a more detailed description of the meteoroid velocity ranges. This is one of the causes of the small difference in total flux generated between the different models [32]. A presentation for the velocity distribution of each one in GEO is presented in figure 5.4a. However, the largest difference between the newer models and that of Grün *et al.* [33] is the inclusion of orbital debris, which is non-existent in the Moon's orbit. Since only the depiction of flux for masses higher than 10^{-12} g is presented in figure 5.4a, one must consider whether this second peak decreases with higher masses. Therefore, the newer McNamara *et al.* [39] model was used to find a more accurate velocity distribution at 1 AU for higher masses and thus more comparable to the Moon's meteoroidal environment. This new distribution can be found in figure 5.4b.



(a) Velocity distributions for GEO and mass > 10^{-12} g [32]



(b) Velocity-frequency range for a mass of $10^{-6} - 10$ g [39]

Figure 5.4: Comparison of meteoroid impact velocities for different models and masses

The original Grün *et al.* [33] model formula before modifications can be seen in equation 5.7, where $\bar{v}_m$ is the average velocity of a meteoroid, k is a constant equal to 4 in the case of an isotopic flux and $N(m, r_0)$

is the spatial density distribution. A distance of 1 AU and an average velocity of 20 km/s were used as previously stated. From the MEM model of McNamara *et al.* [39], it was found that the average velocity is 23.9 km/s. This would result in a 19.5% increase in flux compared to the original Lunar model. To verify if this statement is valid, the plot of Grün's Lunar flux equation 5.6 was plotted in the same figure as the original one of equation 5.7. The results are shown in figure 5.5, where it can be seen that the differences between the distributions are relatively small and their distributions are similar. Therefore, even though negligible compared to the scale, the increase is valid.

$$F(m, r_0) = \left(\frac{\bar{v}}{k} \cdot N(m, r_0)\right) \tag{5.7}$$

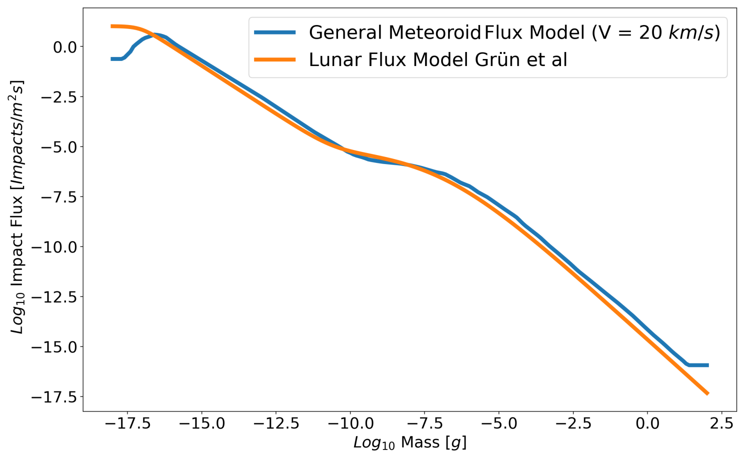


Figure 5.5: General meteorite flux model compared to Lunar flux model of Grün et al. [33]

An important factor to consider is that an isotropic directionality assumption was applied for the Grün model. However, in reality, the real number of meteoroid impacts on the Earth's facing side of the Moon will be lower compared to the far one. This is due to Earth's gravity field causing a change in the trajectories of meteoroids. This difference hasn't been modelled with certainty and would require highly advanced determination software. Therefore, equal distribution of the flux shall be assumed for this assignment. Secondly, for meteoroids less than 10^{-6} g, the mass is uncertain in a range from 0.2 to 5 times the estimated value, implying that the total flux is uncertain in range from 0.33 to 3 times at a given mass. For meteoroids above this size, the flux is well defined. However, the associated mass is even more uncertain. This implies an effective uncertainty in the flux of a factor between 0.1 and 10, at a set mass [38].

5.4.2. ”Large” Meteoroid Flux at the Lunar Surface

The model presented by Oberst *et al.* [37] classifies large meteoroids as having a mass of > 100 g, following the range investigated in subsection 5.4.1 and covering a broad spectrum of possible impacts. This model takes into account gravitational focusing and acceleration of approaching objects and is based on seismic data of impacts and flash detection. Equation 5.8 provides the flux per year (F) for the entire Lunar surface for meteoroid objects with a range in source energy (E_s) from $1.44 \cdot 10^3$ to $2.99 \cdot 10^7$ J.

$$F(E_s) = 8.56 - 0.72 \cdot \log(E_s) \tag{5.8}$$

The maximum flux is produced by the minimum energy. Plugging this in, one finds approximately 3 impacts per year over the entire surface. This frequency is low enough to consider large meteoroids as mission ending events. Hence, a detection method for meteoroids classified as ”large” is required, but no additional design work is needed in terms of meteoroid protection.

5.5. Concluding Remarks

A site 10 km north-east of the Apollo 11 landing site was selected as the building location of the habitat: hence, the geological, radiation, and meteoroid characteristics of this area were explored to an adequate level of detail for the current stage of the mission. Furthermore, the versatility of using regolith for multiple purposes was briefly explored and will be further elaborated upon in the next chapters.

6
Habitat Design

This chapter contains the sizing and layout regarding the external layout of the base itself, as well as all different elements that form the habitat. The sizing constraints are considered in section 6.1, whereas the designs of the components in section 6.2. The final overview of the design is provided in section 6.3.

6.1. Sizing Constraints

The main constraint imposed on the design is the size of the fairing of the launching vehicle. At an earlier stage of this project, it has been assumed that either the Falcon Heavy or Falcon 9 rockets would have been used to transport all elements to the Moon. An iteration of this procedure is performed in subsection 11.2.2. Since they both have the same fairing restrictions, as detailed in figure 6.1, this is used to design the cylindrical modules of the habitat. The maximum diameter allowed for a cylinder is 4.6 m, whereas the maximum height is 6.7 m. However, the tip of the fairing becomes thinner in the last 4.3 m, meaning that the modules cannot have a purely cylindrical shape. Indeed, all elements of the habitat to be designed shall comply to these restraints. The bottom conical space has been added, by designing a payload adapter with a wider diameter. This allows our habitat tubes to be directly attached and gives an additional 0.7 m in height.

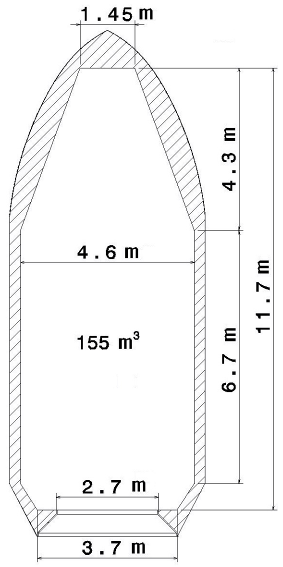


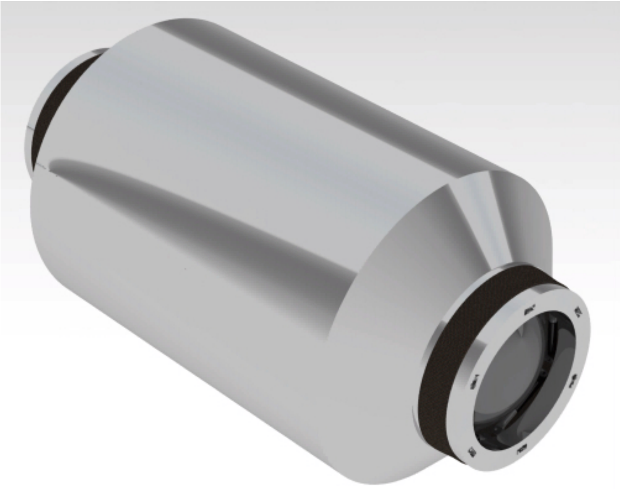
Figure 6.1: Sizing constraints of the fairing of the Falcon rockets

6.2. Components Design

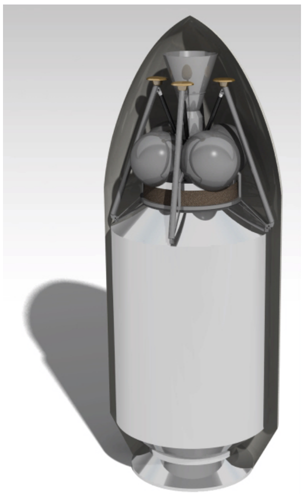
The habitat consists of six separate components, divided into three types: the habitation modules, a central node, and the airlock. Four habitation cylinders are connected to the node, while one of them is also attached to the airlock on its outward-facing side. This is more precisely the Safety & Command Module (SCM), which is a fully independent system. The three different types of components are discussed below.

6.2.1. Habitation Modules

The main habitable volume and area is provided inside the four habitation modules. In fact, these contain the bulk of the subsystems as it is later described in section 10.4. A render of the external design of one habitation module is shown in figure 6.2a, showing one docking port on each side for connection with the node. The end-parts of the cylinders have a narrower part in order to ease the installation between the docking ports of the node; this also fulfils the thinning of the fairing of the Falcon rockets, as shown in figure 6.2b. The final dimensions of the cylindrical modules are 7.5 m in length and 4.6 m in diameter, which leads to a total pressurised volume of 88 m^3 and 49 m^3 of that habitable.



(a) Render of the external design of one module



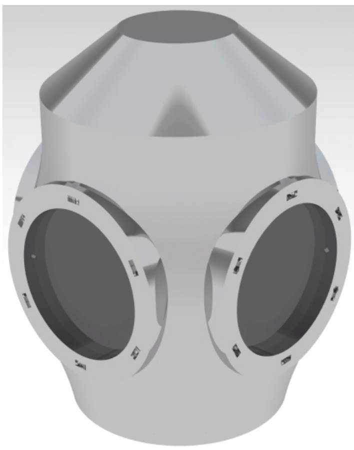
(b) Module as it fits in fairing of Falcon rockets

Figure 6.2: Habitation module design

6.2.2. Central Node

The four habitation modules are linked together via a central node, which has been inspired by the connection module of the ISS. The mechanism is better described in subsection 6.2.4. However, a preliminary render is provided in figure 6.3a. The node is a shorter cylinder compared to the ones previously described: it is 4.6 m and it stands upright, with four docking ports spaced evenly around its diameter. The pressurised volume sums up to being 60 m^3, against an habitable of 30 m^3. A render of the element can be visualised in figure 6.3b. All modules have the possibility of being isolated from the rest of the habitat by means of sliding doors that will eject from the top compartment of the node. However, it has to be outlined that the SCM houses all systems necessary for survival in case of failure of one of the other modules: it in fact is the only component, besides the airlock, equipped with spacesuits for extra-vehicular activities (EVAs).

(a) Render of the docking port of all habitat components



(b) Render of the central node

Figure 6.3: Central node and docking port element

6.2.3. Airlock

As already stated, the airlock is linked to one side of the SCM. The main reasoning behind this is to allow astronauts to escape the compromised habitat from the safety module in case of system failure. During normal operation, the airlock will be used by the astronauts to perform EVAs. This is why it is provided with four so-called "suitports" [40], two on each side as seen in figure 6.4.

Over the past years, Ross *et al.* [41] have been developing Z-2 space suits for NASA, specifically designed to matching with the suitport concept. These suits are developed using 3D body scanning technology to ensure a more precise fitting of the custom. Interfaces needed for the life support backpack are being designed: currently, a prototype of the Z-2 spacesuit is being tested while the design and development of the Z-3, its older brother, is still in the earliest stages. The main entrance, the coning of a reinforced pressure door, will be located on the flat head of the cylinder, whereas the suitports will be located on the sides of the airlock. While the airlock can be depressurised and re-pressurised if necessary, this will not be needed for most EVAs.

A system for cleansing Lunar regolith will be implemented into the airlock and consists of two main parts: a charger and a ventilation. Lunar regolith is often statically charged, which makes it attracted to objects that are not. To negate this effect, the system will induce a small static charge in the airlock, so that the regolith is not attracted to the items inside anymore. The venting system simultaneously blows through the airlock, thrusting the regolith particles into filters. Once the integrated computer detects that the decontamination is complete, the SCM door is opened. To conclude, the airlock's dimensions are 5.5 m in length and 3.3 m in diameter, totalling 11 m^3 of pressurised volume. The airlock does not contribute to the habitable volume as it is only used when strictly required, due to safety concerns.

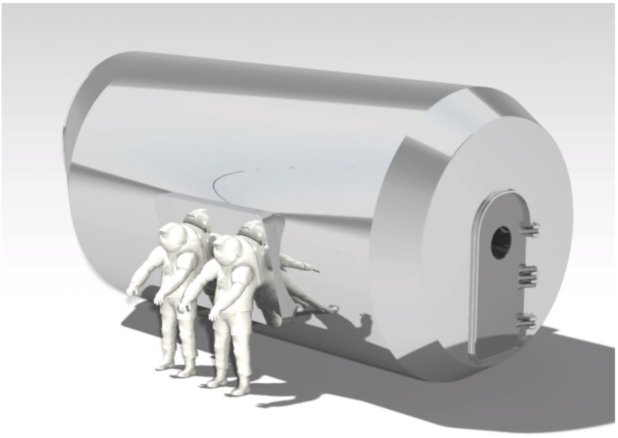
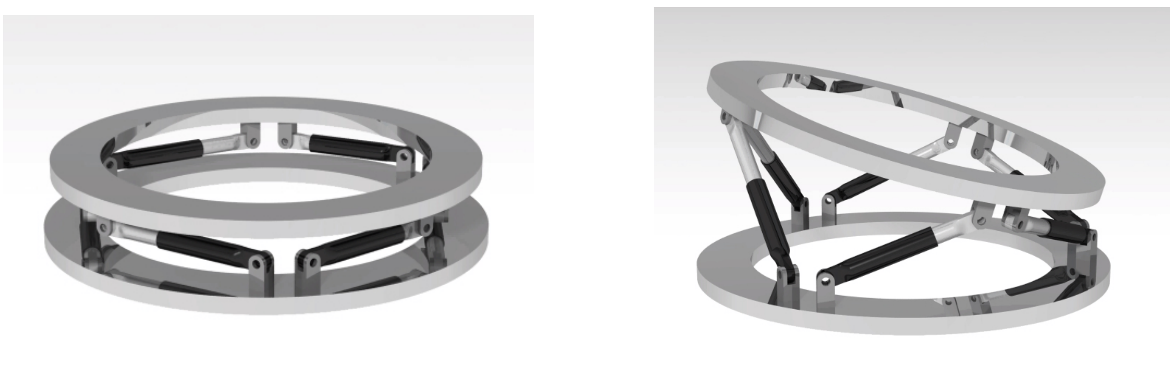


Figure 6.4: Render of the airlock component

6.2.4. Docking Ports

The docking port shown in figure 6.3a and used to connect modules, node, and airlock together is discussed in this section. After a thorough investigation of space berthing and docking mechanisms, it was determined that none of the current flight-tested designs are suitable for Lunar use. This is due to the presence of gravity, which has a negative impact on the actual degrees of freedom. Despite the technical challenges to be faced, this suggests the need of a new design that can adjust to new tolerances. This has been developed and can be seen in figure 6.5.



(a) Folded docking mechanism

(b) Extended docking mechanism

Figure 6.5: Docking port mechanism for connecting the components of the habitat

The docking port provides a six degrees of freedom system that allows the connecting components not to be perfectly aligned. The core of the design consists in two rings: one inside and one outside the module. The inner one is able to close airtight, completely sealing off the module. The outer ring, on the other hand, is connected to six actuators: three needed to provide movement and three for redundancy. The two rings are connected to via a flexible material that provides an airtight seal between them. This component is essentially the same as that considered for the Inflatable Dome design of concept 2. The outer ring contains hooks that latch on to an identical ring on the module that it is connecting to. Once an airtight seal is achieved by using O-rings, the doors mounted on the inner ring may be opened.

6.3. Habitat External Configuration

By combining the designs of the modules, the central node, and the airlock, the external layout of the entire habitat can be represented in figure 6.6. Furthermore, the geometrical properties of each component have been grouped into table 6.1 for a first-order estimation of the volume of the habitat. The outcome of this computation, together with the preliminary layouts of each component, is now brought onto the next step of the design, precisely the structural analysis of chapter 7.

Table 6.1: Geometrical properties of the modules

Component	Length $[m]$	Width $[m]$	Height $[m]$	Volume Pressurised $[m^3]$	Volume Habitable $[m^3]$	Quantity $[-]$
Module	7.5	4.6	4.6	88	49	4
Central node	4.6	4.6	4.6	60	30	1
Airlock	5.5	3.3	3.3	11	0	1
Total	-	-	-	423	226	6

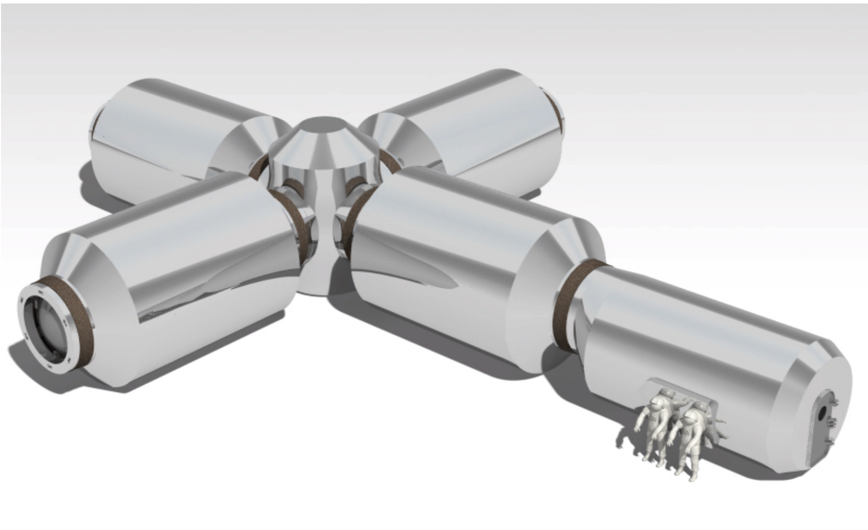


Figure 6.6: External layout of the cylinders, central node, and airlock

6.4. Concluding Remarks

The habitat consists of four horizontally placed cylinders, connected by means of a central node. One of the modules, precisely the SCM, is attached to an airlock that will be used to perform EVAs during operations of the habitat. The first-order estimation of the volume suggests that 423 m^3 and 226 m^3 will be taken for the pressurised and habitable areas, respectively. The designed elements will now be assessed under a structural point of view.

7
Structural Design

This chapter covers all structural design methods and models. It start with a brief description of the sensitivity analysis procedure in section 7.1, followed by a study of both a meteoroid and radiation protection in section 7.2 and 7.3, respectively. Consequently, the thermal design of the habitat is handled in section 7.4 and finally the load bearing structural design is presented in section 7.5.

7.1. Sensitivity Analysis

To determine to what extent the input variables of the model influence the output variables, a sensitivity analysis is conducted. The chosen method is the one described by Sobol [42], which decomposes the variance of the outputs due to uncertain inputs into their respective contributions. Before this method can be applied, two precursory steps have to be performed. All models should be reduced to the so-called 'black box', meaning that they represent an unknown function that produces a set of outputs according to the given inputs. The input and output variables for the models need to be defined. Note that the set of input variables encompasses not only the variables that define the design space, but also variables that were previously estimated or determined.

The second step is that a nominal state has to be defined, which refers to the set of expected inputs and corresponding outputs. Given the maturity of the design, the nominal state will be defined already for most models. The bounds set for the sampling of the Sobol analysis will be ±20% from the nominal value. The analysis will then produce two main values and their confidence intervals for every input variable. The first one is the so-called first-order sensitivity index, which is a measure for how much the specific input variable alone influences the output. All of these indices shall add up to one, signifying the influence of the corresponding input parameter as a percentage. The second value is the total sensitivity index: a measure for mow much the specific input variable and its interactions with other ones influence the output. The sensitivity analysis will be conducted on the models used for this design and on complex or extremely interacting sets of equations, but not on simple, single ones, as the relation and thus the sensitivity of input and output is evident from the equation itself.

7.2. Meteoroid Protection Analysis

The meteoroid protection system design, together with its layout, is presented in this section, and the used relevant models are elaborated upon. For this section, the following assumptions have been made:

- A_n and A_h are the top-view footprint areas;
- Lunar ejecta damage is negligible to direct meteoroid damage;
- The speed of meteoroids impacting the Lunar surface does not exceed 70 km/s;
- The craters due to impact have an hemispherical shape;
- The shape of a meteoroid is a perfect sphere;
- Required regolith thickness is directly related to test material via a factorisation of their densities;
- Impact of meteoroids occur perpendicular to the surface (worst case scenario);
- Analytical and empirical formula's used are valid for speeds higher than 15 km/s;
- Seismic activities due to impact are negligible;
- The radiator of the power source is a titanium monolithic shield.

7.2.1. Meteoroid Protection Analysis - Design Parameters

The Lunar (micro-)meteoroid environment was analysed in section 5.4. Before designing a system that protects the astronauts inside the habitat, a limit on the flux and probability must be set. This value is also known as the Probability of No Penetration (PNP). To define a realistic PNP, Christiansen *et al.* [43] was consulted, where a PNP of 0.98 to 0.998 per critical element over 10 years is defined for the case of the ISS. Since both systems house humans, and have a lifetime of 10 years, this value can be used as guideline for this project. PNP is related to flux according to equation 7.1 and 7.2 [43].

$$N = \sum_{i=1}^{n} N_i = \sum_{i=1}^{n} (FAt)_i \tag{7.1}$$

$$PNP = e^{-N} \tag{7.2}$$

Here, N is the expected number of impacts causing damage exceeding the failure criteria in each region, consisting of the sum of each impact (N_i) over all regions. This is equal to the product of F $[number/m^2 - year]$, the cumulative flux that exceed the failure limits, A $[m^2]$, the exposed area, and t $[year]$, the duration of time exposed to the flux. The input values used in this section are shown in table 7.1.

Table 7.1: Input parameters and corresponding values

Parameter	Value	Unit
ρ_p	1	g/cm^3
ρ_r	1.6	g/cm^3
ρ_s	1.6	g/cm^3
ρ_{al}	2.7	g/cm^3
ρ_{Ti}	4.73	g/cm^3
C_t	4.26	km/s
t	10	$years$
θ	0	°

Parameter	Value	Unit
K_{al}	0.57	–
λ_p	1	–
PNP	0.98 - 0.998	–
A_h	941	m^2
A_n	40	m^2
K	3	-
t_{ti}	0.3	cm
BHN	257	-

7.2.2. Meteoroid Protection Analysis - Impact Model

The design of protection schemes for spacecraft in a meteoroid environment is based on ballistic limit equations (BLEs). These generally consist of either design equations, used to size the shielding elements for particular threat particle sizes under certain impact conditions, and performance equations, used to define the ballistic limit particle size of a particular shield as a function of impact conditions. The most straightforward method of deriving BLEs is to run a series of hypervelocity impact experiments and then correlate the collected damage data to the target and impact parameters. In order to be applicable to the micro-meteorite shielding design, BLEs must span the impact velocity ranges of on-orbit micrometeoroid impacts (3 - 70 km/s). Since these velocities are beyond the capabilities of laboratory hypervelocity launchers, BLEs should be obtained from a combination of laboratory experiments, analytical models and numerical simulations. Since numerical solutions are often performed in the form of hydrocode simulation using advanced software outside the scope of this project, it was decided to use an analytical approach.

Habitat Model

With the aim of designing the penetration model, Hayashida and Robinson [44] was consulted. It compares five existing empirical single plate BLE models, in their accuracy and effectiveness: Fish-Summers, Schmidt-Holsapple, two equations developed for the Apollo project Rockwell and Johnson Space Center (JSC) and the JSC Cour-Palais. Their effectiveness and results are based on velocities of up to 8 km/s and predictions were made for velocities up to 15 km/s with masses of approximately 3 g. Though these limits are far exceeded by the meteoroids on the Moon, the equations developed from the experiments are the only ones available. In addition, the materials used for the projectiles, as well as the targets, in most experiments were not the same materials considered for the habitat. The most conservative equation of the five seems to be the Fish-Summers equation 7.4, hence it was used to verify the results of Savvateev *et al.* [45]. Savvateev *et al.* [45] proposed equation 7.3, which is a formula derived from high-speed penetration tests into sand using bullets of different materials, sizes and speeds up to 3 km/s. Because sand is fairly similar to regolith in structure, the results can be verified by comparing the two. Steinberg and Bulleit [46] was also consulted and seemed to have a very fitting formula for calculating penetration debt in regolith, however this formula seemed to be based on wrong assumptions. The results of this last equation were therefore deemed unfit.

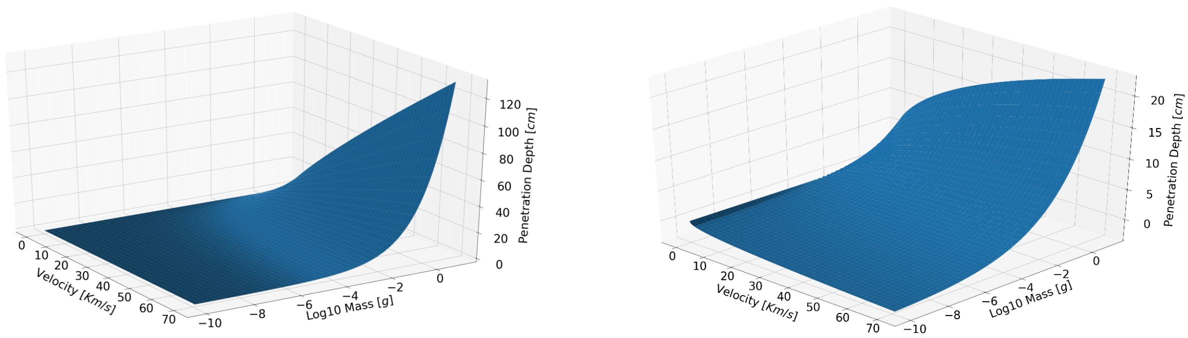
$$t_r = 0.1 \cdot \rho_p \left(\sqrt{26.2} \cdot (d_p) - \frac{1}{\lambda_p} + 17.5 \cdot l_p - 5.3 \right) \bar{v}^{0.4} \frac{\rho_s}{\rho_r} \tag{7.3}$$

$$t_r = K_{al} m_p^{0.352} v^{0.875} \rho_p^{1/6} \frac{\rho_{al}}{\rho_r} \tag{7.4}$$

Where:

- ρ_p = density projectile $[g/cm^3]$;
- ρ_r = density regolith $[g/cm^3]$;
- ρ_s = density sand $[g/cm^3]$;
- ρ_{al} = density aluminium $[g/cm^3]$;
- t_r = thickness regolith $[cm]$;
- t_s = thickness sand $[cm]$;
- d_p = diameter projectile $[cm]$;
- λ_p = aspect ratio projectile $[-]$;
- m_p = projectiles mass $[g]$;
- l_p = length projectile $[cm]$;
- v = projectiles velocity $[km/s]$;
- K_{al} = material constant aluminium $[-]$.

As previously stated, it is assumed that the projectile is a perfect sphere. Using this and the density of meteoroids, equation 7.3 is used to relate mass and velocity to the required regolith thickness. 3D plots are created using equation 7.3 and 7.4 to show their mass - thickness - velocity relation. They are presented in figure 7.1a and 7.1b.



(a) Fish-Summers thickness relation

(b) High-speed sand test thickness relation [45]

Figure 7.1: Comparison of required regolith thickness models

To verify if the 15 kJ mentioned in requirement SYS-OP-SFT-M-1 from the DSE Group 26 [3] is a valid requirement, one assumes the highest possible projectile velocity of 70 km/s to arrive at a mass m of $6.122 \cdot 10^{-3}$ g from the kinetic energy equation 7.5.

$$E_k = \frac{1}{2} m v^2 \tag{7.5}$$

Where, E_k is the kinetic energy in J, m is the projectile's mass in kg, and v the projectile's velocity in m/s. Using equation 5.6, including the 19.5% increase as stated in subsection 5.4.1, this mass relates to a flux of $6.9471 \cdot 10^{-05}$ $[number/m^2 - year]$. Taking into account a habitat top view area $A_h = 941m^2$, results in a PNP of 0.5201. This value is a lot lower than the minimum allowed value. Doing similar calculations for an average velocity of 23.9 km/s, results in a flux of $4.1668 \cdot 10^{-06}$ and a PNP of 0.9616. This is a logical outcome as a lower velocity relates to a higher meteoroid mass to protect for, for which the flux is lower. Figure 7.2 visualises the process of going from PNP to energy and back.

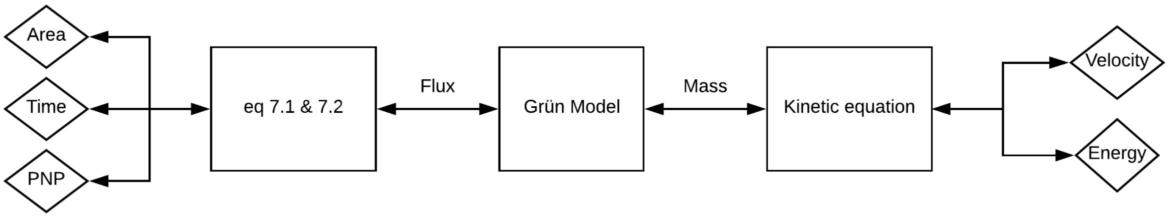


Figure 7.2: Calculation flow relation meteoroid values

In reality the PNP is higher than 0.5201, since a velocity of 70 km/s rarely occurs, as is shown in subsection 5.4.1. Additionally, the energy of projectiles moving at speeds higher than the tested 15 km/s will most likely convert more of their kinetic energy into thermal energy. Therefore these will most likely create relatively smaller penetration depths and cause more melting and vaporisation [47].

Because a PNP of 0.98 is the goal set for an inhabited base, the requirement shall have to be altered. A PNP of 0.98 relates to a flux of $2.147 \cdot 10^{-06}$ $[number/m^2 - year]$, a mass of 0.0868 g and a related energy of 21.3 kJ assuming a velocity of 70 km/s. The requirement has been satisfied but will be adjusted to the new energy level.

The required thickness of regolith can be computed using a mass of $8.68 \cdot 10^{-2}$ g travelling with a velocity of 70 km/s. Using these parameters, one finds a required regolith thickness of 16.75 cm and 4.33 cm by using equation 7.4 and 7.3, respectively. This was repeated for lower speed values and their corresponding masses requiring lower thicknesses when using equation 7.4 and higher thicknesses when using equation 7.3 but not thicker than the initial 16.75cm as could also be evaluated by inspecting figure 7.1a and 7.1b. These values may seem low, but are based on very low masses and their related chance of occurrence.

As stated in subsection 5.4.1, for masses higher than 10^{-6} the predicted fluxes have an uncertainty in the range between 0.1 and 10. Hence, the same calculation was performed using a lower boundary for PNP, namely 0.98, resulting in a flux of $2.147 \cdot 10^{-07}$ $number/m^2 - year$ (instead of $2.147 \cdot 10^{-06}$ $number/m^2 - year$), which corresponds to a mass of 0.493 g. This leads to a required thickness of 30.9 and 7.1 cm. Doing the same for a PNP of 0.998 and including the factor 10 uncertainty, resulted in a required thickness of 56.88 cm for the worst case scenario of an energy impact of 6.86 MJ. Because these values are very feasible, and the required Lunar regolith thickness is larger due to the radiation protection, no further investigation was done using double walled protection for the habitat itself.

7.2.3. Meteoroid Protection Analysis - Sensitivity Analysis

A sensitivity analysis was performed as described in section 7.1. All the input values described in this section were used and provided a list of first-order sensitivity indices per input value. PNP was clearly the most sensitive input parameter with a value of 0.93 out of 1. This seems logical as it is the limiting input value and related to the output through a negative natural logarithm. To verify the change of results due to a change of the PNP, a PNP of 0.8982, a 10% reduction of 0.998, was ran through the model. This results in a mass of $3.618 \cdot 10^{-13}$ (including factor 10 inconsistency) and a required thickness of 10.72 cm. Since PNP is related through a natural logarithm, increasing it with 10% wouldn't be reasonable as the value would attain a value larger than 1 which has no physical meaning. The model was found to be negligibly sensitive to other input parameters (< 0.034).

7.2.4. Meteoroid Protection Analysis - Verification and Validation

Verification

As mentioned in subsection 7.2.2, only analytic and empirical formulas were used. These formulas are well defined and verified formulas. Equation 5.6 has been verified by comparing the formula and its outcomes, in the form of a plot, to plots from Grün *et al.* [33], Staubach *et al.* [35], Oberst *et al.* [37], Dikarev *et al.* [36], Staubach *et al.* [35], and Drolshagen *et al.* [32]. Equation 7.1, 7.2 and 8.6 are all acquired from Christiansen *et al.* [43], which is an official article part of a NASA journal used for designing MMOD protection on the ISS and therefore assumed to be verified. Equation 7.4 is another empirical model which was acquired from Hayashida and Robinson [44]. In Hayashida and Robinson [44] a verification and comparison of each model was provided. Lindsey [48] provided some example calculations which were used to compare to the results of the Matlab model. Savvateev *et al.* [45] was then used to verify the order of magnitude of the results. It provided a whole new model, with thicknesses as results that are a factor 5 to 10 lower. Because this is an empirical dependence equation based on tests with speeds up to 4 km/s it was expected to be less accurate.

Validation

As stated in subsection 7.2.2, all models are based on hypervelocity tests with speeds up to 15 km/s [43], therefore they haven't been validated for speeds from 15 to 70 km/s. The only way of approaching reality in a more detailed manner is using hydrocodes. This requires the application of complicated software which was deemed too time consuming. All the assumptions made to model the impact environment have been stated in section 7.2. Lindsey [48] documents a required thickness - flux graph, with which the results were compared. No existing concepts exist to compare the results to.

7.3. Radiation Protection Analysis

As stated in section 5.3, there are various types of radiation present on the Moon, of which some are more harmful than others. The radiation levels vary significantly, since the Moon is in an orbit that partially exits the protected environment of Earth's magnetic field, as shown by Justh *et al.* [23]. Radiation has the potential to end astronaut careers or lives, and it is therefore of utmost importance to design a protection system that adequately limits their exposure to radiation.

7.3.1. Radiation Protection Analysis - Radiation Dose Limits

Requirement SYS-IO-SFT-R-1 states that the interior levels shall not exceed 165 mGy/yr. However, radiation doses measured in Grays can cause different equivalent doses in various biological objects. Townsend and Fry [49] present an overview of the findings of the National Council on Radiation Protection and Measurements (NCRP), which has adjusted the recommended dose to represent a 3% excess risk of cancer mortality. An overview of these doses can be found in table 7.2.

Table 7.2: Recommended career dose limits, adjusted from Townsend and Fry [49]

Limit	**Bone Marrow** $[Gy-Eq]$	**Eyes** $[Gy-Eq]$	**Skin** $[Gy-Eq]$
Career	1	4	6
Annual	0.5	2	3
30-day	0.25	1	1.5

Note that these dose limits have been set up for missions to LEO, whereas a mission to the moon is outside of the scope of such limits. As stated in section 5.3, an astronaut on the moon is exposed to "highly energetic particles" in the multiple GeV range, which have the potential to cause extensive damage. Therefore, the team has decided that a required radiation limit of approximately one-third of the annual "Bone Marrow" limit should provide adequate protection.

7.3.2. Radiation Protection Analysis - Radiation Model

With a strict requirement,a possibility that was looked into was the radiation blocking effectiveness of Lunar regolith together with water (which is known to perform well as radiation protective layer). As discussed in subsection 7.2.2, the thickness of the regolith layer necessary for meteoroid protection equals 569 mm. Beneath that layer of regolith will be the cylindrical module with two aluminium layers of approximately 2 mm thick.

Lastly, the empty fuel tanks in the walls of the modules can be filled up with water that will form a layer that is between 30 mm and 150 mm thick. To analyse the multi-layered structure described above, the radiation dose inside the habitat will be simulated using MULASSIS (MUlti-LAyered Shielding SImulation Software), which is a part of the SPENVIS software suite. SPENVIS is ESA's Space Environment Information System, used to model the space environment, including cosmic rays, natural radiation belts, Solar energetic particles, plasmas, gases and micro-particles [50]. A simulation was made for a one year Moon orbit, using the layer types as described above. Since the final lay-up was the result of an iterative process between meteoroid protection, structural integrity and thermal control, only the final outcome will be discussed in this section. The justification for the used layers can be found in section 7.2 and 7.5.

The radiation dose behind these layers is simulated in SPENVIS and checked for compliance with the requirement for the annually allowed dose. In the case of non-compliance, iterations are performed for the water and regolith layer thickness. The assumed composition of regolith was found in the "Lunar Sourcebook" by Carrier *et al.* [22], which was used to set up a chemical formula that could simulate the radiation doses absorbed by regolith. According to Chavy-Macdonald [51] the total radiation dose is dominated by isotopes of H and He ions. However a better approximation is found by simulating a group of H and He ions together with C, O, Fe, Si, Mg, Ca and Cr ions. The effect of other ions can be neglected, as they have a contribution of less than 1%. All assumptions made while using SPENVIS can be found below:

- The layers are spherical of shape;
- Every layer material is isotropic;
- No isotopes other than the most critical nine need to be analysed;
- The habitat is simulated as a satellite in the same orbit as the Moon;
- For contingency, the simulated dose error was always added to the result (worst case scenario).

Two major radiation regimes dominate the radiation environment on the Moon. Radiation during a Solar maximum is dominated by the low-penetrating but severe radiation caused by Solar flares, whereas the Solar minimum regime is dominated by GCR. GCR penetrates much deeper than the Solar flare radiation [52], and therefore poses the critical case for the radiation environment. To obtain a model of the radiation environment during a Solar minimum, the CREME96 model was used.

Radiation Protection Analysis - Results

The first iteration was done with the minimum amount of regolith needed for meteoroid protection and a layer thickness of only 3 cm for water. The radiation dose inside the habitat after a year was determined to be over 1 Gy. This is more than six times beyond the allowed radiation levels, and therefore an iterative process was started. The layers of regolith and water were gradually increased, and radiation levels were simulated. Since an increase in one layer can either increase or decrease radiation levels due to secondary particles, various combinations were taken into account trying to use the least amount of resources.

After multiple iterations, a protection of 1.2 m of regolith and 4 cm of water together with the aluminium layers was found to be a good solution. The total radiation dose inside the habitat for a year, with this protection lay-up, was found to be 78.3 mGy/yr. A smaller regolith layer of 1.1 m was also possible, however, since the radiation levels increase significantly (over 130 mGy/yr) in this situation, it was decided to go for the safer option of 1.2 m of regolith. Since the water layer is used for radiation protection, it needs to be checked whether the water goes through phase changes due to varying thermal conditions on the surface of the Moon. In section 7.4 this will be discussed in detail. Since the regolith layer is thicker on the sides of the habitat, water is not needed all around. At 1.3 m regolith thickness, an air layer of 5 cm is already sufficient (< 50 mGy/yr) to be below the required maximum radiation dose.

7.3.3. Radiation Protection Analysis - Verification & Validation

MULASSIS and SPENVIS are both completely verified and validated by ESA. Performing full model verification and validation is therefore a waste of resources. However, the team's use of the software is something that needs to be verified, and is accomplished by comparing our simulation results to similar studies on multi-layered shielding design for Lunar habitats.

Simonsen and Nealy [53] have studied the radiation dose in lunar soil due to both SPE and GCR, using a baryon transport code (BRYNTRN) combined with an a heavy-ion transport code, simulating the transport of high energy ions up to atomic number 28. They find an SPE equivalent dose of approximately 5 rem (50 $mGy-Eq$) for a regolith thickness of 150 g/cm^2, which translates to approximately 75 cm using a regolith density of 2 g/cm^3. In terms of GCR, they find an equivalent dose of 18 rem/yr (180 $mGy-Eq/yr$) at a regolith thickness of approximately 50 cm. Clearly, the SPENVIS simulation performed by the team assumes higher equivalent doses, but the total dosages are within the same order of magnitude.

Pham and El-Genk [54] present a study on dose estimates inside a Lunar shelter using regolith shielding. They find an effective dose of approximately 40 mSv (40 $Gy-Eq$) beneath a regolith shield of 200 g/cm^2 (about 1 m, using a regolith density of 2). Most of this dose is caused by secondary neutron and proton radiation, similar to the team's radiation analysis using SPENVIS. Once more, our own simulation shows higher dose results in the same order of magnitude.

Given the similarities in doses between both studies mentioned above and our own simulation, and the fact that the SPENVIS simulation shows a higher resultant dose, the teams deems the 1.2 m of regolith with 4 cm of water an adequate protection system.

7.4. Thermal Analysis

The thermal cycle of the Moon's surface has been presented in subsection 5.3.3. Due to the large difference in maximum and minimum temperature, good thermal insulation and control has to be implemented to provide for a comfortable internal temperature. In order to estimate the heat flux through the different layers of material, it was decided to make a MATLAB model of the thermal diffusion through the layers. The model had to give insight on the temperature flux beneath 120 cm of regolith which covers the cylinders, which was determined by the radiation requirements. Due to the low conductivity of Lunar regolith, it was predicted that the temperature would be a constant, low value [31]. If the model would give this result, it could be investigated if the empty fuel tanks could be re-used for passive or active thermal control. By means of iterating, together with the results of the iterative process of the radiation model, it was looked at the possibility of either filling up the space with water or air. If water was needed for radiation protection, the model had to check if the water would not freeze or boil. If air were to be used, it should be checked how the freezing constant temperature would influence the internal temperature of the habitat.

7.4.1. Thermal Analysis - Thermal Model

To simulate the diffusion of the heat and the possibly moving water, a combination of the linear diffusion and linear advection partial differential equation was established. In order to implement this equation, which is presented in equation 7.6a, into a model, the equation had to be discretised. Since the time scale is large (period of 27 days), an implicit method was preferred over an explicit one since an implicit method is unconditionally stable. Therefore, the Euler implicit discretisation scheme was used.

$$\frac{\partial u}{\partial t} + c \cdot \frac{\partial u}{\partial y} = \alpha \cdot \frac{\partial^2 u}{\partial x^2} + \alpha \cdot \frac{\partial^2 u}{\partial y^2} \tag{7.6a}$$

$$\alpha = \frac{k_t}{c_p \rho} \tag{7.6b}$$

In this equation $'u'$ represents the temperature, $'t'$ the time variable, $'x'$ the radial direction from outside pointing inside the habitat, and $'y'$ the circumferential direction around the habitat. As can be seen in the equations, the water flows around the habitat and the temperature fluctuation is in radial direction. The material property $'\alpha'$ is represented by equation 7.6b. In this equation k_t is the conductivity of the material, c_p is the specific heat and ρ is the material density. It should be noted that $'\alpha$ is therefore different for each material. In reality α changes slightly with respect to depth and temperature, however for simplicity it is assumed to be constant throughout the layer. Finally, $'c'$ is the velocity of the flow of the water around the habitat. Certain assumptions to simplify the model were made:

- For each layer, α is considered to be constant;
- The temperature fluctuation on the Moon's surface is assumed to be sinusoidal as stated in equation 7.7;
- The water velocity is constant everywhere;
- The wall of the habitat is flat;
- The internal temperature of the habitat will be constant at 20$^\circ C$.

$$\begin{aligned} T_{surface} &= \frac{T_{max} - T_{min}}{2} \cdot \sin\left(\frac{2\pi \cdot t}{period} - \frac{\pi}{2}\right) + \left(T_{max} - \frac{T_{max} - T_{min}}{2}\right) \\ T_{surface} &= \frac{400 - 95}{2} \cdot \sin\left(\frac{2\pi \cdot t}{2\pi} - \frac{\pi}{2}\right) + \left(400 - \frac{400 - 95}{2}\right) \end{aligned} \tag{7.7}$$

A discretised model needs a mesh on which it works. Since different materials are used, a variable mesh was created with p nodes in x-direction and r nodes in y-direction. The mesh is finer for thinner layers, and becomes finer near material transitions. This way the model is able to capture rapid changes with higher accuracy without compromising computation time. A close-up of the mesh distribution can be found in figure 7.3, where a layer of water instead of air was assumed.

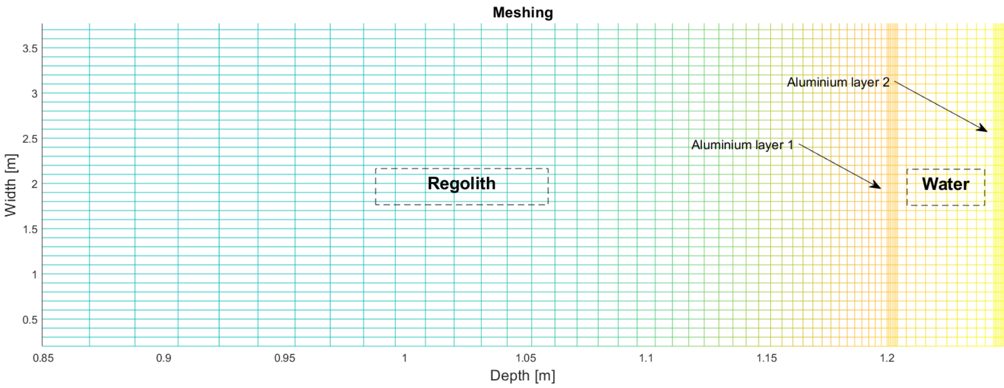


Figure 7.3: Close up of variable mesh of thermal model

In order to produce the implicit scheme, equation 7.8 was developed.

$$\begin{aligned}
&u_{i,j}^{n+1}\left[1-\Delta t\cdot\alpha\left(\frac{-(\Delta x_1+\Delta x_2)}{0.5(\Delta x_1^2\Delta x_2+\Delta x_1\Delta x_2^2)}-\frac{2}{\Delta y^2}\right)\right]+\\
&u_{i+1,j}^{n+1}\left[-\Delta t\cdot\alpha\frac{\Delta x_2}{0.5(\Delta x_1^2\Delta x_2+\Delta x_1\Delta x_2^2)}\right]+\\
&u_{i-1,j}^{n+1}\left[-\Delta t\cdot\alpha\frac{\Delta x_1}{0.5(\Delta x_1^2\Delta x_2+\Delta x_1\Delta x_2^2)}\right]+\\
&u_{i,j+1}^{n+1}\left[-\Delta t\cdot\alpha\frac{1}{\Delta y^2}+\Delta t\cdot c\frac{1}{2\Delta y}\right]+\\
&u_{i,j-1}^{n+1}\left[-\Delta t\cdot\alpha\frac{1}{\Delta y^2}-\Delta t\cdot c\frac{1}{2\Delta y}\right]=u_{i,j}^n
\end{aligned}\tag{7.8}$$

In the above equation, Δx_1 and Δx_2 are the distances from nodes i to $i+1$, and $i-1$ to i, respectively. The difference in distance originates from the variable mesh in x-direction. With equation 7.8 a matrix could be set up as can be seen in equation 7.9. In order to visualise the matrix all the arguments with respect to (i,j), $(i+1,j)$, $(i-1,j)$, $(i,j+1)$ and $(i,j-1)$ are grouped as a, b, c, d and e respectively.

$$\begin{bmatrix}1&0&0&0&0&0&0&\dots&0\\ \vdots&\ddots&\ddots&\ddots&\ddots&\ddots&\ddots&\ddots&\vdots\\ \dots&e&\dots&c&a&b&\dots&d&\dots\\ \vdots&\ddots&\ddots&\ddots&\ddots&\ddots&\ddots&\ddots&\vdots\\ 0&0&0&0&0&0&0&\dots&1\end{bmatrix}\cdot\begin{bmatrix}u_{1,1}^{n+1}\\ \vdots\\ u_{i,j-1}^{n+1}\\ \vdots\\ u_{i-1,j}^{n+1}\\ u_{i,j}^{n+1}\\ u_{i+1,j}^{n+1}\\ \vdots\\ u_{i,j+1}^{n+1}\\ \vdots\\ u_{i+p,j+r}^{n+1}\end{bmatrix}=\begin{bmatrix}u_{1,1}^{n}\\ \vdots\\ u_{i,j-1}^{n}\\ \vdots\\ u_{i-1,j}^{n}\\ u_{i,j}^{n}\\ u_{i+1,j}^{n}\\ \vdots\\ u_{i,j+1}^{n}\\ \vdots\\ u_{i+p,j+r}^{n}\end{bmatrix}\tag{7.9}$$

The length of the solution matrix is $p\cdot r$ since there are p nodes in x-direction and r nodes in y-direction. In order to solve the system for u^{n+1}, the matrix is inverted and pre-multiplied with the u^n matrix which is done each time interval. since a differential equation is used, boundary and initial conditions are needed, these are listed below:

- Initial: all the points on the grid start with a temperature of 247.5 K;
- Boundary 1: at the surface the temperature changes sinusoidal according to equation 7.7;
- Boundary 2: the left boundary has a constant temperature of 247.5 K, except where the water/air comes in where a constant temperature of 293.5 K is stated;
- Boundary 3: the right boundary has a constant temperature of 247.5 K except for where the water/air streams out where a Neumann condition is applied. The temperature at the boundary is equal to the temperature of the nodes next to it;

- Boundary 4: the boundary at the second aluminium layer is constant at 293.5 K since this is the desired inside temperature of the habitat.

7.4.2. Thermal Analysis - Verification

Code Verification

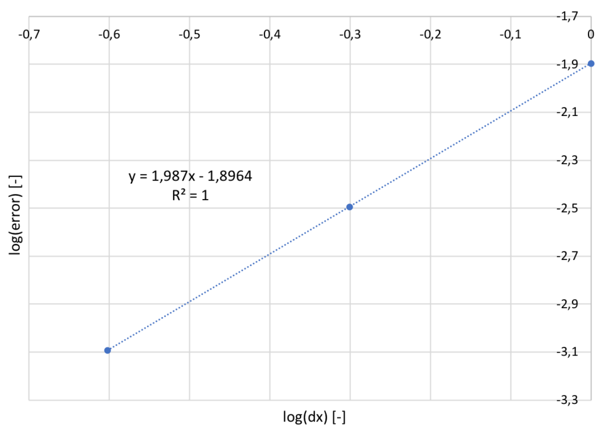
The code developed to produce these results needed to be verified. In order to verify the code, the method of manufactured solutions (MMS) was used. A solution for u was set up via equation 7.10, it is important to note that the function should be chosen in such way that the terms do not disappear after taking the derivatives in the PDE. This exact solution was put into the PDE and solved using the developed code. Since the exact solution u is known it could be compared to the numerical solution and an average error could be computed.

$$u = \sin x \cdot \cos y \cdot \sin t \tag{7.10}$$

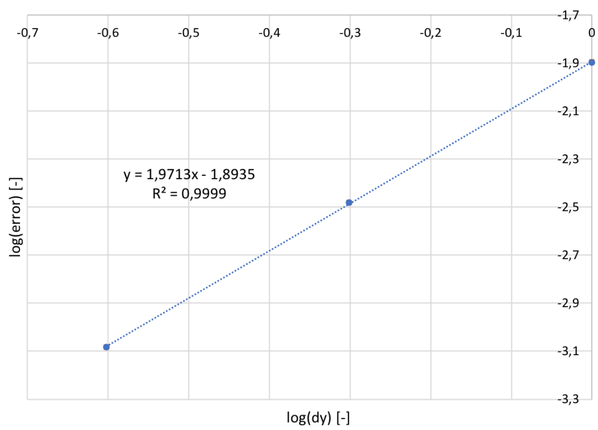
By using the MMS, an order-of-accuracy test could be performed. In this test the convergence rate of the model is checked. Since the model is first order accurate in time and second order accurate in mesh spacing, the model should converge with order 2 as derived in equation 7.11 and 7.12. The equation for $\frac{\partial^2 u}{\partial y^2}$ is equivalent to the one presented in equation 7.12. To check this convergence behaviour, dx was changed three times, with each iteration halving the spacing. Due to the orders of accuracy, the time-step had to be quartered each time dx got halved. The same procedure was taken for dy. The logs of the results were taken and plotted which are presented in figure 7.4a and 7.4b. As can be seen in the logplots the slopes are 1.987 and 1.971 for a change in dx and dy respectively, meaning the model converges with an order of approximately 2.

$$\begin{aligned} u_{i,j}^{n+1} &= u_{i,j}^{n+1} + \Delta t \frac{\partial u}{\partial t} + O(\Delta t^2) \\ \Rightarrow \frac{\partial u}{\partial t} &= \frac{u_{i,j}^{n+1} - u_{i,j}^{n}}{\Delta t} + O(\Delta t) \end{aligned} \tag{7.11}$$

$$\begin{aligned} u_{i+1,j}^{n+1} &= u_{i,j}^{n+1} + \Delta x \frac{\partial u}{\partial x} + \frac{\Delta x^2}{2}\frac{\partial^2 u}{\partial x^2} + \frac{\Delta x^3}{3!}\frac{\partial^3 u}{\partial x^3} + O(\Delta x^4) \\ u_{i-1,j}^{n+1} &= u_{i,j}^{n+1} - \Delta x \frac{\partial u}{\partial x} + \frac{\Delta x^2}{2}\frac{\partial^2 u}{\partial x^2} - \frac{\Delta x^3}{3!}\frac{\partial^3 u}{\partial x^3} + O(\Delta x^4) \\ \Rightarrow \frac{\partial^2 u}{\partial x^2} &= \frac{u_{i+1,j}^{n+1} - 2u_{i,j}^{n+1} + u_{i-1,j}^{n+1}}{\Delta x^2} + O(\Delta x^2) \\ \Rightarrow \frac{\partial u}{\partial y} &= \frac{u_{i+1}^{n+1} - u_{i-1}^{n+1}}{2\Delta y} + O(\Delta y^2) \end{aligned} \tag{7.12}$$



(a) Logplot of a change in mesh size in x-direction with a change in time-step



(b) Logplot of a change in mesh size in y-direction with a change in time-step

Figure 7.4: Logpolts of changes in mesh size with changes in time

7.4.3. Results and Conclusion

With the thermal model the impact of the temperature fluctuations will be investigated as well as the temperature of the water layer inside the habitat shell. When the temperatures inside the shell are either too high or too low, a velocity will be added to the water flow to make sure the water temperature does not fluctuate out of the required range. In this case, the inflow of water will have a temperature of 20°C. The water which flows out with an increased or decreased temperature will go to the radiator or heater depending on which one is necessary. The integration will be explained in section 10.2 which covers the design integration. After simulating time passes, the initial conditions will gradually have less influence on the solution. After enough time the solution will approach the steady state solution. Such result is presented in figure 7.5.

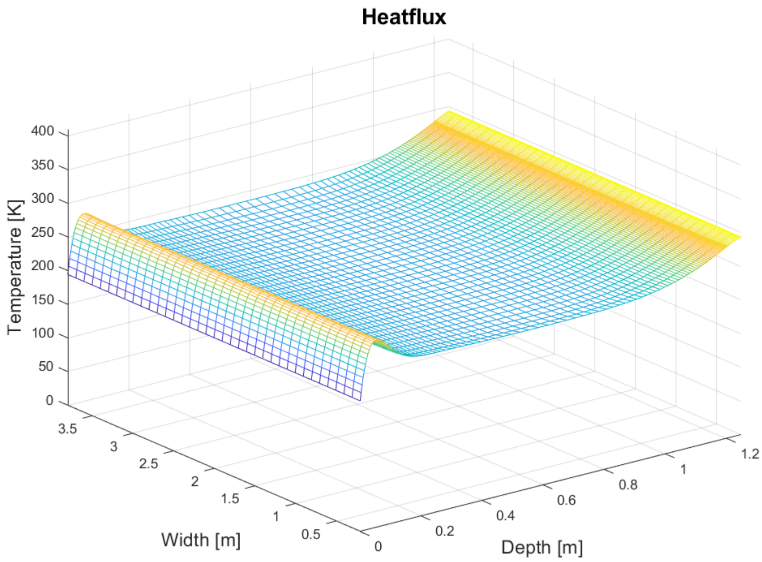


Figure 7.5: 2D model of the heat propagation

It was found that, for a depth from 20 cm onward, the temperature flux is constant. In this region the temperature was found to fluctuate around 247.50 $K \pm 0.5$ K. This result complies with the expectation that the temperature should remain constant at a certain depth due to the low conductivity of the regolith. If the water would be preheated at room temperature and injected in the walls the model predicts that the water would not freeze over. Instead it would cool down to a minimum temperature of 282.80 K, having an average water temperature of 287.68 K. Which means the water does not have to be pumped around to keep it in a liquid phase. However, since the water is cooled this could be used for the thermal subsystem when it needs cooled water. Therefore the water will slowly be pumped around the habitat wall with a velocity of 1 cm/s and then pumped into the radiator of the thermal subsystem for further cooling. One would expect that, contrary to the static problem, the water would be warmer when being pumped around. The model confirms this expectation, however the average temperature only goes up by 0.1 K to an average temperature of 287.80 K.

7.4.4. Thermal Analysis - Validation

For validation it is necessary to find out whether the used equations actually represent the reality. Since the diffusion and advection equations are used worldwide to solve thermal and advection problems, there is little to no doubt that these equations represent reality accurate enough to use. In order to be more certain, a simple experiment could be conducted in the future. This experiment would heat one side of a multi-layered structure, and multiple temperature sensors will measure the local temperature. This temperature will be compared to the values from the program.

7.5. Cylinder Design

There are several load cases during the lifetime of the habitat. However two cases are of importance for the conceptual design stage, the launch and the operation. The transfer, landing and deployment of the cylinders should also be analysed, however at this level of design not enough information is known to obtain relevant results. Furthermore, more elements of the habitat, such as the lander and transfer vehicle should also be analysed, however the analysis also suffers from the same problem. For this reason, the structural analysis shall focus on the main elements of the cylinders and only for the case of the launch and operation, wheres sufficient infromation is available.

7.5.1. Launch

During the launch stage the cylinders are subjected to an axial acceleration of 6 g, resulting in compressive forces on the structure. The free body diagram of the whole structure, the outer and inner cylinders can be seen in figure 7.6. The analysis of this load case is done with the following assumptions:

- The structure is clamped at the payload adapter and all loads are introduced there;
- The main load is carried by the inner cylinder as most systems are connected to it;
- The outer cylinder only carries its own weight;
- The fuel in the tank is compressed by the acceleration and creates a pressure on the cylinder walls;
- The material of the cylinder is uniform;
- The structure can be considered thin walled;
- The inner cylinder can be considered to be straight, its fillet is considered negligible;
- All calculations are done for Aluminium Alloy 6070.

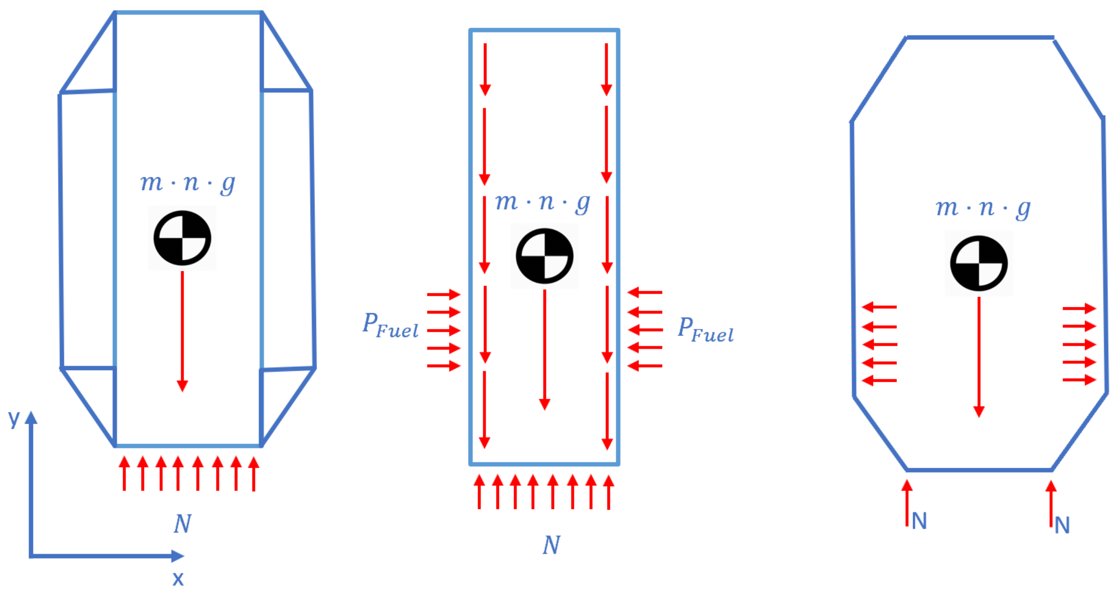


Figure 7.6: Free body diagrams of the main structural components during launch

Due to the simplicity of the loads the sum of moment and forces is omitted in this case, it must be note that the structure is considered static under an acceleration on the reference frame. Two important loads are used to size the cylinders, the axial load due to launch and the pressure the fuel causes on the walls due to the previous load.

Fuel Pressure

As mentioned previously, the cavity between the cylinders is filled with fuel for the landing, during the launch as 6 g acceleration is applied causing the mass of the fuel to create a pressure on side of the tank. It can be calculated using equation 7.13 to be 503.5 kPa.

$$P_{Fuel} = h \cdot n \cdot g \cdot \rho \tag{7.13}$$

Where:

- h is the height of the fuel tank, equal to 5.9 m;
- n is the number of g, equal to 6;
- g is earths gravitational constant, equal to 9.81 m/s^2;
- ρ is the density of the fuel, equal to 1450 kg/m^3.

In order to estimate the necessary wall thickness to resist this pressure, the method presented in equation 7.5.1 is used.

Pressure Difference

Due to this pressure difference there will be induced loads in radial, circumferential and axial direction. In order to calculate these loads the Lamé equations were applied. Starting with an equation for the equilibrium of forces and using derivations from Katna [55], the three sets of equation 7.14 could be computed.

$$\sigma_r = C_1 + \frac{C_2}{r^2} \qquad \sigma_c = C_1 - \frac{C_2}{r^2} \qquad \sigma_a = \frac{p_i r_i^2 - p_o r_o^2}{r_o^2 - r_i^2} \tag{7.14}$$

In which C_1 and C_2 are the constants of integration given in equation 7.15.

$$C_1 = \frac{p_i r_i^2 - p_o r_o^2}{r_o^2 - r_i^2} \qquad C_2 = \frac{r_i^2 r_o^2 (p_i - p_o)}{r_o^2 - r_i^2} \tag{7.15}$$

$$\sigma_{vm} = \sqrt{\frac{1}{2}\left[(\sigma_r - \sigma_c)^2 + (\sigma_c - \sigma_a)^2 + (\sigma_a - \sigma_r)^2\right] + 3\left[\tau_{rc}^2 + \tau_{ca}^2 + \tau_{ar}^2\right]} \tag{7.16}$$

These equations are obtained by using the following assumptions:

- For the equilibrium of forces, higher order terms are negligible;
- Body forces may be ignored;
- Uniformly distributed stress, whereas effect of end caps ignored;
- During launch the outer pressure is assumed 0 bar, to remain conservative;
- The cylinder is considered pressurised to 1 bar.

σ represents the stress, whereas r_o and r_i the outer and inner radius respectively, P_o and P_i are the outer and inner pressure and subscripts r, c and a denote 'radial', 'circumferential', and 'axial', respectively. Finally, these stresses are combined in equation 7.16 to compute the Von Mises stress.

Using the pressure difference of 5 bars together with the above equations, the required wall thickness in order for the material to not yield can be determined. These values are 4.19 mm for the outer cylinder and 2.7 mm for the inner cylinder (due to the lower radius and the 1 bar pressure inside). Since the structure includes stiffeners that connect the the inner and outer tank providing extra rigidity and relieving some of the load, the value can be considered very conservative, without the need to apply a safety factor. It should be noted that this value would only be required at the very bottom of the tank, however for simplicity sake, as a first-order estimate, it is applied to the whole structure.

Axial Load - Compression

During launch, the payload undergoes an axial load, and it needs to be checked whether the material's yield stress is not exceeded. The maximum axial load can be calculated by multiplying the weight of 22 $tons$ by the gravitation acceleration, and then dividing this by the cross-sectional area. Since the cross-sectional area is dependant on thickness, the minimum required thickness is found so that the cylinder does not yield. The minimum required thickness equals 0.23 mm, and is therefore not critical for the design.

Axial Load - Buckling

It is assumed that the inner cylinder caries the weight of the whole structure minus the weight of the outer cylinder which is self standing. There are two types of buckling analyses that can be made, column buckling and thin plate buckling, i.e. crippling. Due to the relatively low aspect ratio of the structure, which approximately measures 4.39 m in diameter and 7.7 m in height, the critical load will be in crippling considerations. However, both cases will be checked. Structures fail by column buckling when the whole structure reaches its critical load is reached and buckles outwards. The critical load can be calculated using equation 7.17 [56].

$$F = \frac{n\pi^2 EI}{L^2} \tag{7.17}$$

In which n is the factor accounting for end conditions, chosen as 0.25 [56], E is the young's modulus, I the moment of inertia and L the length of the column. As can be seen in figure 7.7, the required thickness is in the order of 10^{-6} m, and therefore can be disregarded by means of the thin-wall approximation.

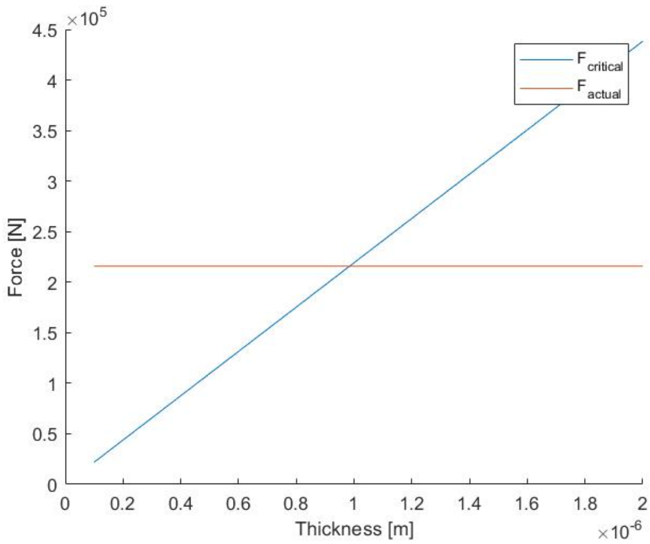


Figure 7.7: Column buckling load

The more likely form of buckling to be present is crippling. Crippling is the buckling of the 'sheet' the cylinder rather than the whole column itself. Since the habitat is a thin-walled structure, this has to be taken into account. The pertinent load is modelled as the weight of the whole structure placed on the top of the inner cylinder at launch. Since the whole structure can at maximum weigh 22.8 $tons$ and the outer cylinder weighs 1.26 $tons$, taking into account a wall thickness of 4.19 mm as per calculated in equation 7.5.1, the load carried by the inner cylinder is 21.54 $tons$. As mentioned before, the Falcon 9 produces an acceleration of 6 g This leads to a load of 1.26 MN. In order to determine the critical buckling load, it was assumed that the cylinder is a thin cylindrical plate. Using equation 7.18 [57], where t is the thickness, l is the length, μ is Poisson's ratio, k is the end fixity coefficient, and E is Young's modulus, it was determined that the necessary thickness to sustain the load is 3.04 mm. The end fixity coefficient used is for the case of one end clamped and all other sides free, as it is one of the most conservative cases that matches the cylinder being only clamped at the payload adaptor. The resulting diameter to thickness ratio of 1444 validates the thin walled assumption. In the case of the outer cylinder the crippling load is 1.32 MN, while the actual load is 0.074 MN, which indicates that the 4.19 mm are more than sufficient.

$$F_{cr} = \frac{k\pi^2 E(\frac{t}{l})^2}{12(1-\mu^2)b^2} \tag{7.18}$$

7.5.2. Dimensions for Launch

The dimensions necessary to sustain the loads during launch for the cylinders are shown in table 7.3.

Table 7.3: Cylinder wall thicknesses and sizing loads

Cylinder	Wall thickness	Sizing load
Inner	3.04	Crippling
Outer	4.19	Fuel pressure

7.5.3. Operational

The final load case which needs to be taken into account is the weight of Lunar regolith covering the habitat, together with and without a pressurized habitat. The critical case would be a non-pressurized situation with the loads induced by the regolith on top. If this case would not lead to failure, a pressurized habitat with regolith on top will not fail either, since they are only counteracting each other. In order to simplify the problem a conservative assumption was made, which states that the regolith is equally high everywhere. It is also assumed by equation 7.19 that the pressure loads due to the regolith is linearly varying with the depth.

$$P_{reg} = \rho g h \tag{7.19}$$

Above, ρ is the regolith density, g is the gravitational acceleration on the Moon, and h the depth of the regolith. To calculate the internal loads through the cylindrical shell of the habitat, a model was made which cut the outer cylinder in small parts of $d\theta$ as can be see in figure 7.8.

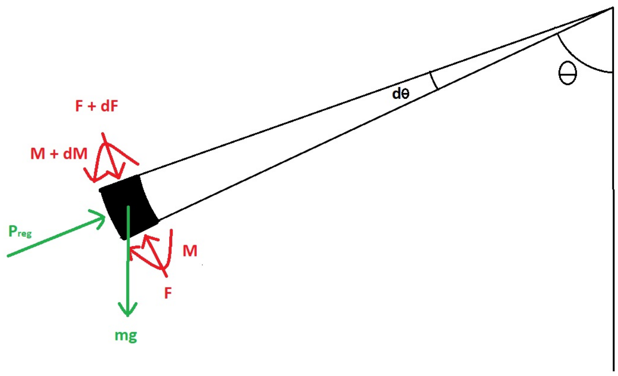


Figure 7.8: Element of habitat cylindrical wall

In the figure, the weightis calculated in weight per unit length, similar to the pressure load which is multiplied with the element length. The internal calculated forces are then divided by the thickness to obtain the stress through the material. In these calculations, an overestimate of 1 m regolith on top is used to be conservative. The largest thickness for the outer shell due to buckling is used, equalling 4.2 mm. Also, a thin-walled assumption is made since D/t > 1000, neglecting the shear through the thickness of the material. The results can be found in figure 7.9.

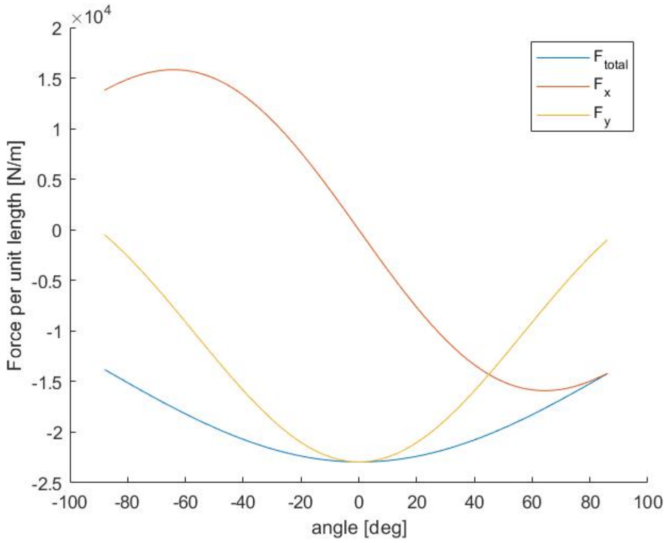


Figure 7.9: Forces per meter length through cylindrical wall

As can be seen in the figure above, the resultant force through the cylinder is compression. Dividing the results found by the thickness will yield the internal stresses. The maximum internal stress is found to be -5.46 MPa in compression. This result is not close to the yield stress of aluminium (276 MPa) and therefore this load case will not influence the design of the habitat. To make sure these results are right, a verification is performed. For the validation of this model, a similar load case of pipelines underground was used [58], and can be found in the next section.

7.5.4. Structural Analysis - Verification and Validation

Verification

Since the load case during the operational phase is the only one for which a model was produced, this is the only case which needs to be verified and validated. For the remaining cases standard handbook equations were used. For this model, a verification system test was performed. For this test the loading was simplified to a uniform loading like a pressurised tank. With a uniform pressure, the resultant force magnitude should be similar for each different element, while the forces in x- and y-direction vary. Since this was the case when the model was used, it could be verified that the model is actually doing what is desired.

Validation

The method for pipelines underground lies on calculating the ovalisation of the pipe based on the load on the top, defined in equation 7.20. The inputs are moment of Inertia (I), Young's modulus (E), deflection lag factor ($D_l \approx 1.5$) , bending constant (k), pipe radius (R), pipe diameter (D) and pressure on pipe (P). In order to determine the soil modulus (E'), the plasticity and relative density need to be determined. Using regolith stimulants developed by NASA, it was determined that Lunar regolith exhibit limited to no plasticity, with a relative density (D_r) of 65 and a Proctor of 91% [59] [60]. From this information, it can be determined that regolith has a value of soil modulus of 400 lb/in^2 [61].

$$\frac{\Delta y}{D} = \frac{D_l k P}{\frac{EI}{R^3} + 0.061E'} \tag{7.20}$$

It was determined that the pipe ovalisation at the conditions is 0.00127. This value was used in equation 7.21 to determine that the internal stress is approximately -3.2 MPa. The value differs from the model by 40%, however it is in the same order of magnitude and significantly lower than the yield stress of aluminium. The difference is mainly due to the estimation of the E' value from the given tables, as the pipeline formula is highly sensitive to variations its magnitude.

$$\sigma = 4E\left(\frac{\Delta y}{D}\right)\left(\frac{t}{D}\right) \tag{7.21}$$

7.5.5. Other Materials

The requirements related to materials and structural integrity can be found in appendix B. The same

analysis is repeated using other materials, the results are shown table 7.4. It can be clearly seen that Aluminium 7055-T7751 results in the lowest structural weight. This material is also one of the newest alloys of aluminium developed, hence the price is also high. Since the cost of 1 extra ton of launch weight is in the millions, the price of the material is negligible.

Table 7.4: Structural results in case where other materials were used [62–67]

Metal alloy	Wall thickness Inner $[mm]$	Wall thickness Outer $[mm]$	Wall weight Inner $[tons]$	Wall weight Outer $[ton]$	Total weight $[ton]$
Aluminium 7075-T6	3.05	2.30	0.91	0.72	1.63
Aluminium 7055-T7751	2.74	1.60	0.83	0.51	1.34
Aluminium 2195-T8	2.11	2.03	0.99	0.68	1.67
Aluminium 7178-T6	3.05	2.15	0.92	0.68	1.59
Titanium Ti-6AI-4V	2.42	1.32	1.1	0.65	1.78

7.6. Final Dimensions

After iterating the design for structural, radiation and thermal purposes, the final dimensions converged to a final number, which are shown in table 7.5. It must be mentioned that this mass estimation is overly conservative as it assumes that the cylinders are not reinforced by stiffeners, and hence carry all the loads. If stiffeners were added, this mass would most likely drop.

Table 7.5: Final dimensions of the structural layers

Layer	Thickness $[mm]$	Length $[m]$	Radius $[m]$	Weight $[kg]$
Regolith	1200	7.7	-	-
Inner cylinder	2.74	7.7	2.25	830
Water	40	5.9	$\approx$ 2.25	-
Outer cylinder	1.60	7.7	2.3	510

7.7. Concluding Remarks

The structural design of the habitat constitutes of 4 parts, namely, the meteoroid protection, the radiation protection, thermal stability and structural integrity. The main objective of this section was to size the thicknesses of the aluminium, water and regolith layers around the cylinder for these cases. The regolith layer thickness is determined by radiation protection constraints, requiring a thickness of 1.2 m. It is over-designed for micrometeoroid protection which only requires 56.88 cm to provide a PNP of 99.8 %. Hence, the actual probability of penetration drops significantly below 0.0001%. Furthermore, the thermal analysis concludes that using the aforementioned regolith layer, only 4 cm of water is required to provide sufficient thermal stability. Finally, once all non-structural layers were determined, the aluminium layers that carry all the loads could be sized. It was determined that a thickness of 2.74 mm and 1.6 mm, for the inner and outer cylinder walls, respectively, are needed. A final iteration was conducted with the radiation model and thermal models using these new aluminium thicknesses, verifying that the design is viable. For all above cases, the layer was overestimated to provide an upper bound on the mass and volume of material required. The goal of this overestimation is to have detailed design activities decrease the total mass, instead of add to it.

8
Subsystem Design

In this chapter the final design, each of the subsystems and their related properties are provided. The subsystems evaluated are: meteoroid protection in section 8.1, radiation detection in section 8.2, environment control and life support in section 8.3, communications in section 8.4, command, data handling and computer system in section 8.5 and the power system in section 8.6.

8.1. Meteoroid Protection

Following from the analysis of section 7.2, active meteoroid protection subsystem is now defined.

8.1.1. Meteoroid Protection - Inputs and Requirements

The models and results defined in section 7.2 have been used to update the requirements. The requirements can be found in appendix B.

8.1.2. Meteoroid Protection - Functional Flow Diagram

Figure 8.1 shows the functional flow diagram of the active part of the meteoroid protection subsystem.

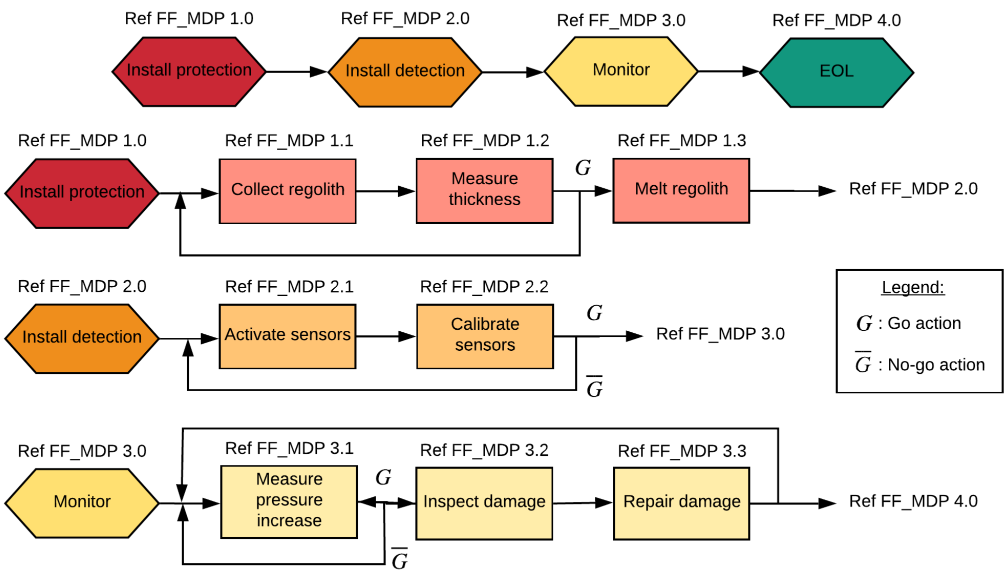


Figure 8.1: Functional flow of the meteoroid protection subsystem

8.1.3. Meteoroid Protection - Design Layout

The model from section 7.2 pointed out that no protection other than 56.88 cm of Lunar regolith was required to protect the habitat with a 0.998 PNP for a period of 10 years. In addition, section 7.3 pointed out that a thickness of 120 cm regolith is required to sufficiently shield the astronauts from radiation. Regarding meteoroid impacts the protective layer is overdesigned. Additionally, the outer habitat aluminium layer also provides some protection which has not even been accounted for because of the requirement SYS-OP-SFT-M-7 that no damage shall occur to it.

Nevertheless, it was decided to place sensors in the outer shell of the habitat. In case penetration or contact does occur, its location can be checked for damage and repaired if deemed necessary. Christiansen *et al.* [43] mentions piezoelectric film impact detection panels. The panels are compatible with MLI blankets and the typical operational temperature range is $-50°C$ to $150°C$, which is satisfied by section 7.4. A two-wire buss system connects the panels. These two wires provide power and data to all panels. The panels, physically closest to the SCM, are connected to the main computer system. This requires two electrical feedthroughs through the pressure barrier. The film used by NASA is usually manufactured in rolls of 60.96

cm wide, with an overall weight of 1.6 kg/m^2 including all the additional required layers. Only the top half of each cylinder shall have sensors, assuming the lower half won't be affected due to the dug-in aspect and the thicker layer of regolith. Half of the total surface area of one cylinder is 42.6 m^2. Using 12 strips over the full length of the cylinder and an adapted configuration, for the curved ends, using one strip per side results in a total of 14 strips per module and a total weight of 68.12 kg per module. One panel uses 10 mW, hence a total of 140 mW per module. For the airlock 45.6 kg and a power use of approximately 90 mW is needed. The node is estimated to require an additional 26.5 kg with a total power usage of 30 mW. The quality of the regolith layer will be monitored via EVA visual inspection rounds. If deemed necessary, the rover shall repair the damaged or affected area. Important to note is that the sensors shall have to be re-calibrated once the modules have been pressurised and the Lunar regolith has been placed over the habitat to include the permanently induced strains due to the pressure differences.

All techniques for locating meteoroids are based on locating the ionisation effect that occurs when it either enters the Earth's atmosphere and when it becomes a meteor, or when it impacts a celestial body such as the Moon. The most advanced technique used in active localisation is using forward scattering radio waves. However, this is not possible on the Moon since these systems are located on the Earth. But way more important is the fact that a lot of meteoroids striking the Moon, never emit light or leave an ionisation trail before they strike the Lunar surface. This only occurs if they pass through the Earth's atmosphere before heading to the Moon [68]. Because the habitat is situated on the Earth facing side of the Moon, some of the meteoroids impacting near the habitats site are assumed to have passed through the Earth's atmosphere. Therefore, the best way of observing them is acquiring data from Earth's observation stations in case a projectile is expected to impact the Lunar surface close to the habitat by calculating and predicting their trajectory. This would be similar to the detection of particles approaching the ISS but less accurate due to the distance to the Moon. Currently, the US Space Surveillance Network has the most advanced technology for locating meteors and therefore a collaboration is desired. Since they already provide information to ESA, this is considered to be a realistic solution. In case puncture or penetration does occur, the module will be closed off and the safety procedures will be effectuated as described in subsection 12.1.3.

8.1.4. Meteoroid Protection - Technical Budgets

Mass

The required mass for the sensors is 68.12 kg per module, 45.6 kg for the airlock and 25.6 kg for the node. This translates to a total added weight of 344 kg. Other mass contributions come from the installation and regolith rover. The budget of the rovers shall be covered in subsection 9.2.4. The mass of the required regolith was not included as it does not have to be transported to the Moon.

Volume

The thickness of the sensor panels is in the order of 0.08 mm. With a total estimated area of 215 m^2, this results in a volume of 0.0172 m^3, which is regarded as negligible with regards to the total volume.

Power

As stated before, the total estimated power is 260 mW in monitoring mode. In communication mode, the required power could increase to several watts [43].

Cost

A price indication of the piezoelectric film was found in Santamarina *et al.* [69]. It states a price of 10 $\$/ft^2$ for non-space use. This does not include the wiring, assembly and installation cost. Material cost for non-space use would equal $23142. It is estimated that the total cost would be in the order of $100000. This is only a fraction of the total cost and shall be included in the final project cost estimation in the form of a factor increase together with other small components. Using the services of the US Space Surveillance Network might also require funding. If collaboration could be effectuated, by for example providing seismic Lunar data, the costs might be heavily reduced or even sponsored.

8.1.5. Meteoroid Protection - RAMS

To analyse RAMS properties of subsystems, a Failure Mode Effect Analysis (FMEA) is used for every subsystem. In such an analysis, the functions presented in the functional flow diagram are analysed for cause and effect, and assigned a score for severity, occurrence and detection. The product of these scores leads to a so-called Risk Priority Number (RPN), a useful tool to prioritise mitigation strategies.

Severity is scored on a 1-4 scale: 1 being a negligible failure, i.e. minor annoyance, 2 being a marginal failure, i.e. something has to be repaired, 3 being a critical failure, i.e. early mission end or major astronaut injury, and 4 being a catastrophic failure, i.e. immediate mission end or astronaut death. Occurrence is scored on a 1-2 scale, with 1 representing electronic systems and 2 representing mechanical or biological systems, as these systems have lower reliability in general terms. Determination is scored on a 1-3 scale, with a 1 representing immediate detection, a 2 representing delayed detection and a 3 representing no detection possible. The FMEA of the meteoroid protection & detection system is presented in figure 8.2.

Function ID:	Function	Failure Mode	Potential Effect of failure	SEV	Potential causes of failure	OCC	Current Control	DET	RPN
FF_MDP1	Install Protection	No installation of protection	Installation schedule delay	2	Rover breaks down	2	Health monitoring of rovers	2	8
			No hardended protection available	2	Sintering unfeasible	1	Visual inspection	2	4
FF_MDP2	Install Detection	No method of detecting hits	Permanent distortion of data	2	Calibration error	1	Network detection	1	2
FF_MDP3	Monitor	Failed detection	Local decrease of detection accuracy	1	Broken sensor	1	Network detection	1	1
			Complete loss of detection	2	Cable rupture	1	Redundancy and pre-inspection	3	6

Figure 8.2: FMEA of the meteoroid protection & detection system

8.1.6. Meteoroid Protection - Risk & Mitigation

As is seen in figure 8.2, two situations exist for which the RPN's are relatively high and contingency strategies should be elaborated upon:

1. Rover breaks down: multiple rovers are present, one rover breaking down is an annoyance but not critical for mission continuation. Testing the rovers for similar amounts of time in a similar simulated environment on Earth could prevent such failures. Because their TRL is low, the suggested mitigation strategy is to perform an extensive testing campaign;

2. Cable rupture: there are only a few locations where the sensors penetrate the module's wall to be connected to the CDH, these locations could be critical points. By building in redundancy this could be prevented. The suggested mitigation strategies are: using solely space proven high TRL wiring and connecting to the CDH at multiple points.

8.1.7. Meteoroid Protection - Product Verification

Each requirement stated in the Midterm Report [3] has been verified and adjusted to the list displayed in subsection 8.1.1. SYS-OP-SFT-M-1 was verified in subsection 7.2.2 and was adjusted to the new value. SYS-OP-SFT-M-2 will be part of the ICT system and was concluded in subsection 8.1.3, shall be activated from Earth when the US Space Surveillance Network calculates a detected meteoroid has a high probability of impacting close to the habitat. Since the radiation protection subsystem requires a regolith layer thickness of 120cm, SYS-OP-SFT-M-3 has been adjusted. Reversing the Fish-Summer equation 7.4 and using an average velocity of 23.9 km/s a correlated penetration energy of 57 MJ was obtained. In case puncture does occur, the safety procedures will be effectuated immediately as described in subsection 12.1.3. SYS-OP-SFT-M-4 has been adjusted to the new energy penetration value. As described in subsection 7.2.2, the chance of such a meteoroid hitting is negligibly small. Might the US Space Surveillance Network detect a meteoroid that has a chance of impacting close to the habitat, the astronauts would be alerted hours before impact. SYS-OP-SFT-M-5 has been deleted, it was not a user requirement and an in-situ impact detector is only effective for modelling meteoroid impacts patterns, not predicting impacts directly. SYS-OP-SFT-M-6 will be described in subsection 12.1.3.

8.1.8. Meteoroid Protection - Product Validation

The sensors mentioned are well established, reliable sensors. They are currently used on the ISS [43]. To validate the results of the sensors in this set-up, they should be submitted to tests on Earth, which is easily possible due to their on-Earth assembly.

8.2. Radiation Detection

In this section, the radiation detection subsystem is defined. This follows from the analysis performed in section 7.3.

8.2.1. Radiation Detection - Inputs and Requirements

To start the design of the radiation detection system of the habitat, the requirements and constraints regarding radiation shall have to be satisfied. All given requirements on radiation are given in appendix B

8.2.2. Radiation Detection - Functional Flow Diagram

Figure 8.4 shows the functional flow diagram of the radiation detection subsystem. It shows the active functions of the system.

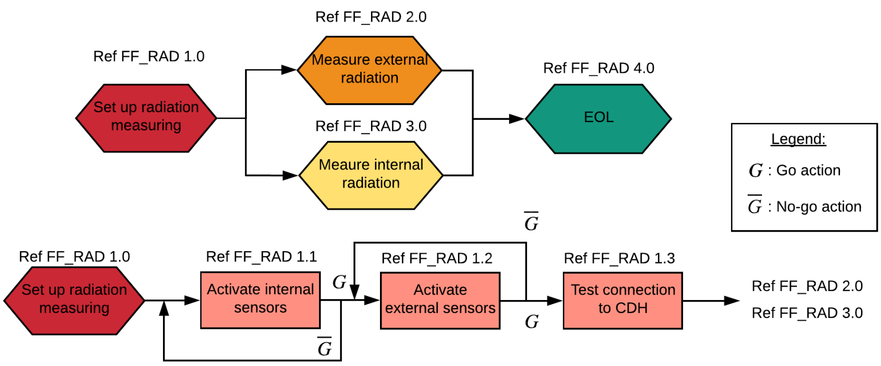


Figure 8.3: Functional flow of the radiation detection subsystem

In addition to the functional flow diagram, the process flow diagram was added to visualise the functioning of the system.

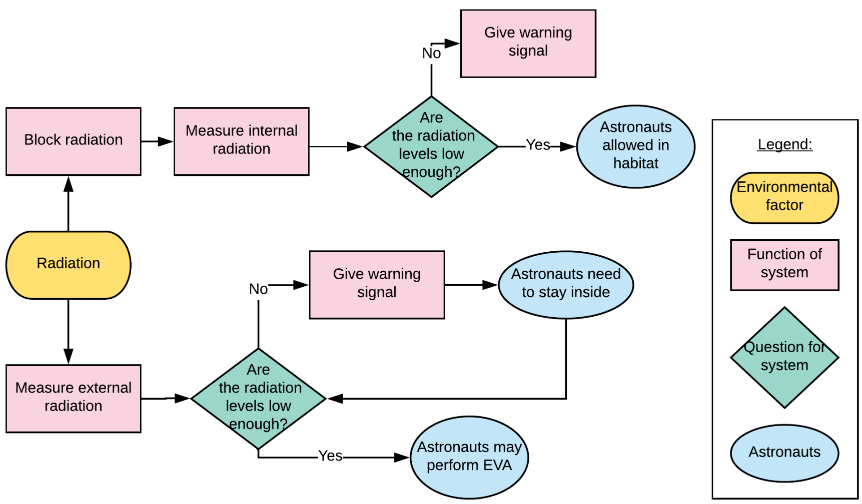


Figure 8.4: Functional flow of the radiation protection subsystem

8.2.3. Radiation Detection - Layout

Consulting Sharek *et al.* [70], Gaza *et al.* [71] and Semones and McLeod [72] and assuming the operational date of the habitat is not before 2025, the following combination of instruments are implemented:

- CPAD: Crew Personal Active Dosimeter;

- RAM: Radiation Area Monitor;
- HERA: Hybrid Electronic Radiation Assessor;
- MPT: Miniaturised Particle Telescope;
- ISS-RAD: ISS-Radiation Assessment Detector;
- FNS: Fast Neutron Spectrometer.

This configuration is going to be used on the EM-2 mission and was mentioned to be covering for a habitat mission in Semones and McLeod [72].

Table 8.1: Characteristics of radiation sensors

Instrument	Quantity	Mass $[kg]$	Power $[W]$	Location	Purpose
CPAD	8	240	$9.6 \cdot 10^{-04}$	IV + EV	- Active personal dosimeter
RAM	22	242	–	IV	- Passive dosimeter for area monitoring
HERA	1	1400	11.7	IV + EV	- Active radiation detector with full vehicle integration
MPT	1	155	< 3	IV + EV	- Miniaturised Particle Telescope spectrometer
ISS-RAD	1	10000	12	IV + EV	- Charged and neutral particle spectroscopy and dosimetry
FNS	1	4700	< 7.5	IV + EV	- Neutron spectrometer

It has to be mentioned that some of these systems are currently being demonstrated, tested or are in their last stages of testing and development. They are assumed to be fully developed at the end of 2021 [71]. The last four systems of the table are most likely to be converged into one system in the coming years [72].

8.2.4. Radiation Detection - Technical Budgets

Mass and Power

As stated in table 8.1, where the total power and mass budgets of each sensor is presented, a total power estimate was found to be 35 W. The total mass of all the sensors equals around 17 kg.

Cost

There is hardly any information available regarding development costs of the radiation detection configuration mentioned in the previous subsection. As a reference, a retrospective audit document, Sharek *et al.* [70], has been consulted. This document states the total cost to be $26.4 million for the radiation detection systems used on the ISS ARI program to date. Costs of the program increased as it experienced heavy delays. However, since the implementation of this program has to be more efficient but is also more complex, the stated costs are regarded to be a good estimate for the radiation detection system in the habitat.

8.2.5. Radiation Detection - RAMS

The FMEA for the radiation detection system is presented in figure 8.5. All the explanations for the scoring scheme can be found in subsection 8.1.5.

Function ID:	Function	Failure Mode	Potential Effect of failure	SEV	Potential causes of failure	OCC	Current Control	DET	RPN
FF_RAD1	Set up radiation measuring	Installation failure	Incomplete measurements	2	Missing sensors	1	Inspect launch inventory	3	6
			No radiation measurements	3	Sensors broken during transfer	1	Network detection	1	3
FF_RAD2	Measure external radiation	Failed/Wrong measurement	Exposure to high radiation events	3	Broken sensors installed	1	Network detection	1	3
				3	Calibration error	1	Network detection	1	3
FF_RAD3	Measure internal radiation	Failed/Wrong measurement	Chance of astronaut radiation sickness	2	Broken sensors installed	1	Network detection	1	2
				2	Calibration error	1	Network detection	1	2

Figure 8.5: FMEA of the radiation detection system

8.2.6. Radiation Detection - Risk & Contingency

As can be observed in figure 8.5, the sensor suite for radiation detection is a rather robust system when linked to a central network. One critical failure that has to be mitigated is the potential lack of sensors, which is now controlled by inspecting launch inventory. Adding in additional redundant sensors can lower the detection score, relieving some of the risks present in this system.

8.2.7. Radiation Detection - Product Verification

SYS-OP-SFT-R-1 has been satisfied. With the regolith layer set at 1.2 m as can be found in section 7.3, the habitat is adequately protected against the radiation environment (78.3 mGy/yr < 165 mGy/yr). SYS-OP-SFT-R-2 has also been satisfied. The mentioned HERA system is the follow up of the BIRD system. This system measured within a 1.45 mGy during a 253 day mission. This equals roughly 2.1 $mGy/year$ [73].

8.2.8. Radiation Detection - Product Validation

All the instruments stated in this section have been tested and used in the ISS or similar missions. However, some of the instruments are currently being tested, demonstrated or developed. Therefore, the products can only be properly validated after their development phase, most likely to end after 2021 [71]. Additional testing is required and is most likely to be done during EM-2.

8.3. Environment Control and Life Support System

The Environmental Control and Life Support System (ECLSS) creates a safe and comfortable environment for the astronauts to live in. It comprises of multiple systems namely: food management, water management, waste management, pressure regulation, atmospheric management, inner thermal control, environmental monitoring, exercise equipment, medical equipment and noise control. This section details the designs of these systems and presents the considerations made.

8.3.1. ECLSS - Inputs and Requirements

The following requirements were set up regarding life support systems. Some requirements are used as they were stated in the Midterm Report [3] and some changes were made, requirements were updated and requirements were added. The ECLSS requirements can be found in appendix B.

8.3.2. ECLSS - Functional Flow Structure

It was chosen not to elaborate upon the installation of the systems and the end of life procedures, since it is the same for all subsystems within the ECLSS, as well as the majority of the other systems which are pre-installed. The FFDs of the food supply, the waste management, atmospheric management, medical, and fire repression systems are visualised in figure 8.6, 8.7, 8.8, 8.9, and 8.10, respectively.

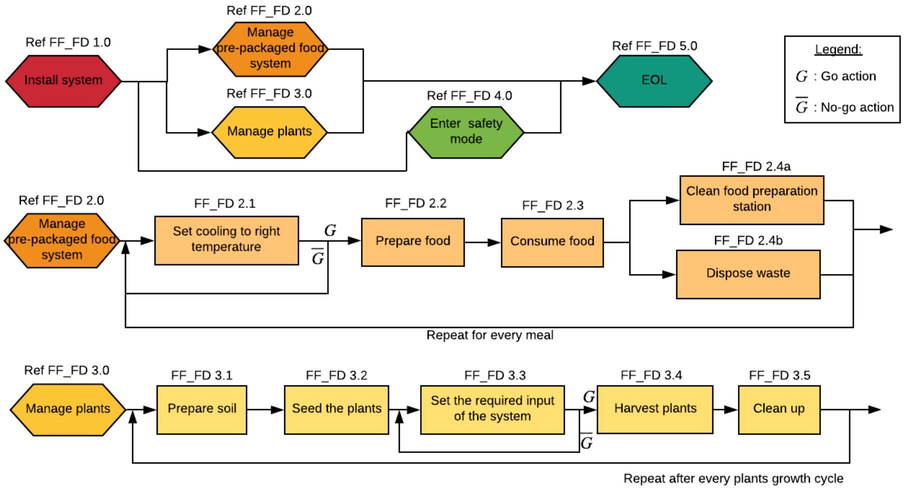


Figure 8.6: Functional flow diagram of the food system

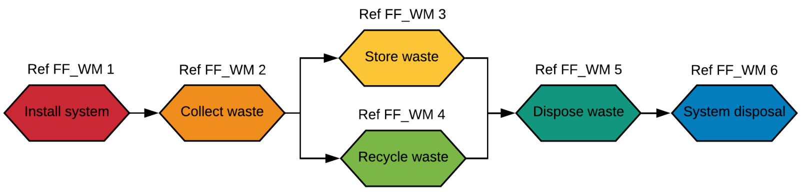


Figure 8.7: Functional flow diagram for the waste management system

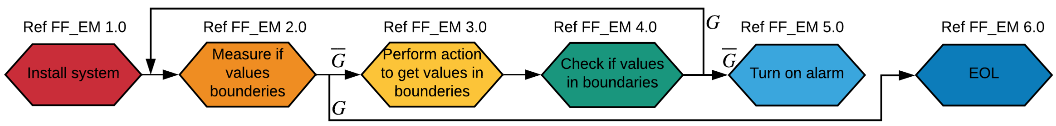


Figure 8.8: Functional flow diagram of the atmospheric management system

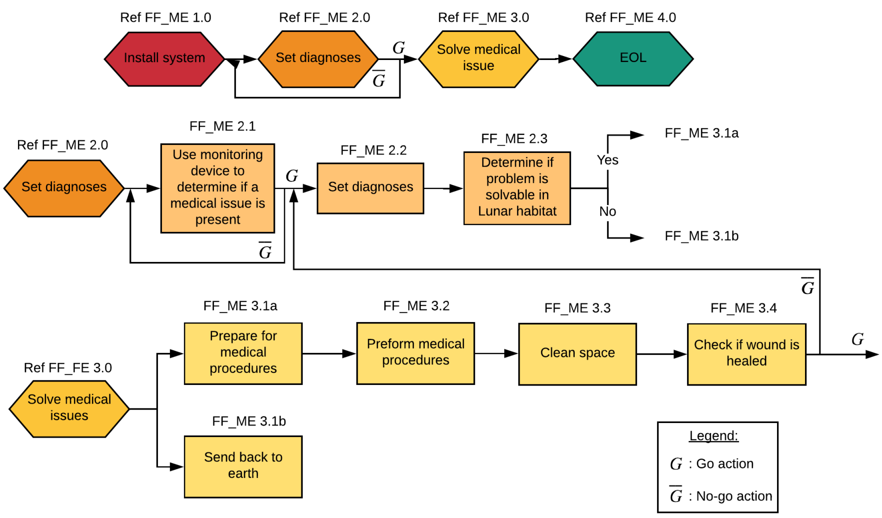


Figure 8.9: Functional flow diagram of the medical system

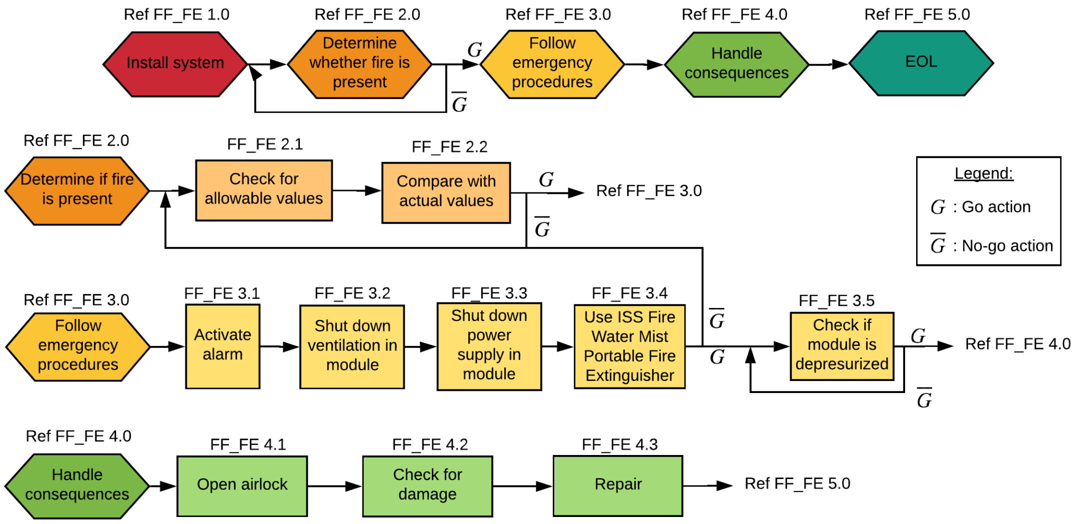


Figure 8.10: Functional flow diagram of the fire repression system

8.3.3. ECLSS - Interfaces

From the product N2 chart, presented in the Midterm Report [3], the interfaces of the ECLSS can be identified. For the sake of brevity, only the interfaces that are driving the design are discussed below.

- **Design power system**: the power available restricts the design space for the life support system, yet the life support system also imposes requirements on the power system. Especially the ways of providing food are extremely dependent on the power available. While growing food inside the habitat saves mass on the long term, it is a power intensive process;
- **Design safety system**: in case of emergency, emergency supplies need to be present. Also, backup systems need to be in place to ensure the water, food, waste, atmospheric management, medical and inner thermal systems continue to function adequately;
- **Design computer and sensor system**: as the ECLSS mainly depends on operating and keeping the habitat within a narrowly defined margin of values (such as atmospheric pressure), the central computer will constantly monitor all ECLSS parameters and will also take appropriate action in the event that a parameter deviates from its expected or required margin;
- **Design astronomic characteristics**: the volume and weight of life support systems will influence the amount of launch vehicles needed. Furthermore, if part of the food supply would be provided by growing plants, requirements on the launch conditions might be imposed;
- **Determine layout**: since the ECLSS systems take up volume and must be accessible for the crew, they will influence the internal layout of the habitat. Special attention must be paid to the ability to perform maintenance for systems which are prone to failure;
- **Crew**: the reason for having life support is the inclusion of humans into the mission. The ECLSS should therefore be easy to use and maintain, and provide the crew with a comfortable living space. Furthermore, in case growing plants is included in the food supply, this will have a positive influence on the happiness of astronauts;
- **Internal interfaces**: since the ECLSS comprises of many elements, internal interfaces need to be considered. The integration of the systems within the ECLSS will be discussed in chapter 10.

8.3.4. ECLSS - Design Layout

This section discusses the design layout of the subsystems of the ECLSS.

Food

Firstly, a trade-off has been made on the supply method of the food. The two main options are: sending pre-packaged food or growing vegetables in addition to the food supply. The influence of a microgravity environment on plants can result in the regulation of genes related to the response to stress, plant development and cell propagation [74]. By modification of the metabolism, cell wall rigidity will be increased

[75], resulting in similar plant growth as observed on Earth. Two systems designed by NASA, to stimulate plant growth in space, are considered for the Lunar habitat: VEGGIE and the Advanced Plant Habitat (APH). VEGGIE is $0.02m^3$, weighs $720kg$ and uses 115 Watts. To grow sufficient food for half of the dietary need of one astronaut, $20m^2$ of growing surface is needed, which would result in 125 VEGGIE modules consuming $14375W$ of power and weighing $900kg$ [27, 76]. This is considered as an unfeasible solution.

The APH module is a more advanced, recently developed plant growing chamber developed to do research. APH is currently being tested in the ISS, meaning that not all specifications are currently known. The temperature, humidity levels and air composition can be adjusted, allowing for the growth of different plants. Furthermore, the power supply to each subsystem can be turned on and off separately [77]. As stated in *SYS-OP-LS-F-3*, astronauts need an average of 3000 $kcal$ per day. Assuming 50% of the food is fresh or frozen, and 50% is dehydrated, the weight of the food will approximately total 4200 kg [27]. Furthermore, the kitchen equipment is estimated to weigh 375 kg. Bringing all food from Earth is thus beneficial for a one year mission. However, it was decided to send two APH modules to perform research on the possibility of growing vegetables on the Moon. The first reason is that the plants provide psychological benefits for the astronauts [27]. Secondly, the texture and taste of plants is a well received addition to the diet. Lastly, as the habitat is expected to remain operational for 10 years, growing plants would effectively lower the launch mass in future missions. In case the APH usage is expanded, it should be investigated if the power required can be lowered by utilising the heat inherently generated by the habitat systems.

Water

Two options are considered for the water recovery system, namely: an improved version of the water recovery system as is currently present in the US segment of the ISS, the Water Recovery System (WRS) [78] and an improved version of the Alternative Water Processor (AWP) as is currently developed by NASA. The high-level architecture of the WRS can be seen in figure 8.11. The main advantage of this system is the high TRL. Although currently the recovery rate from urine is lower than the expected 85%, the system continues to maintain an acceptable water balance. This lack of performance can be explained due to an underestimation of calcium sulphate being present in urine, caused by increased bone de-calcification in low-gravity environments. NASA engineers are far along in improving the system's performance, it is expected that the habitat could be equipped with the improved system. The main disadvantage of the WRS system is the total recovery rate. Although the system recovers around 85% of the water it processes, it is only able to process 15-20% of the anticipated water usage of an exploration mission [79]. The reason is that the system can not process all hygiene and laundry waste water. Furthermore, a toxic solution is used to stabilise the urine to prevent biological growth and urea hydrolysis [80].

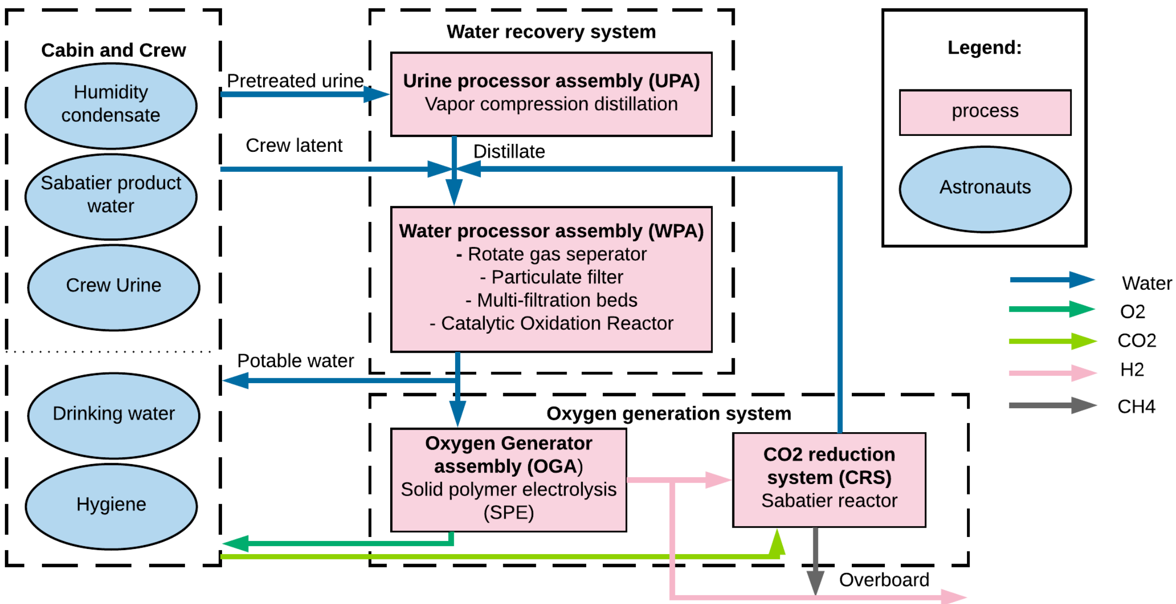


Figure 8.11: High level architecture of the water recovery system, based on [78]

The alternative water processor (AWP) uses four bio-reactors which utilise bacterial metabolic processes to remove carbon and nitrogen from water by means of nitrification and de-nitrification [80]. Afterwards, the residual water goes through a process of forwarded and reversed osmosis to assure removal of large molecules. For this process the Hydration Technology Innovation FO membrane was used. A high-level architecture can be seen in figure 8.12. The system is able to not only process the traditionally recyclable water, but also waste water at a 90% recovery rate, using the most optimal configuration. During testing, the carbon and ammonia filter rates were 55.5% and 83%, which were below the required goals of 75% and 90%. A solution would be to incorporate additional carbon in the water to increase the carbon to nitrogen ratio, resulting in better performance.

A more difficult to solve problem of the AWP is that it currently can not process enough water. An advantage of the system is that it does not use chemical substances, and it consumes 64% less resources. Combining everything, it can be seen that the system is not yet fully capable for this mission. However the issues are expected to be resolved in the coming four years.

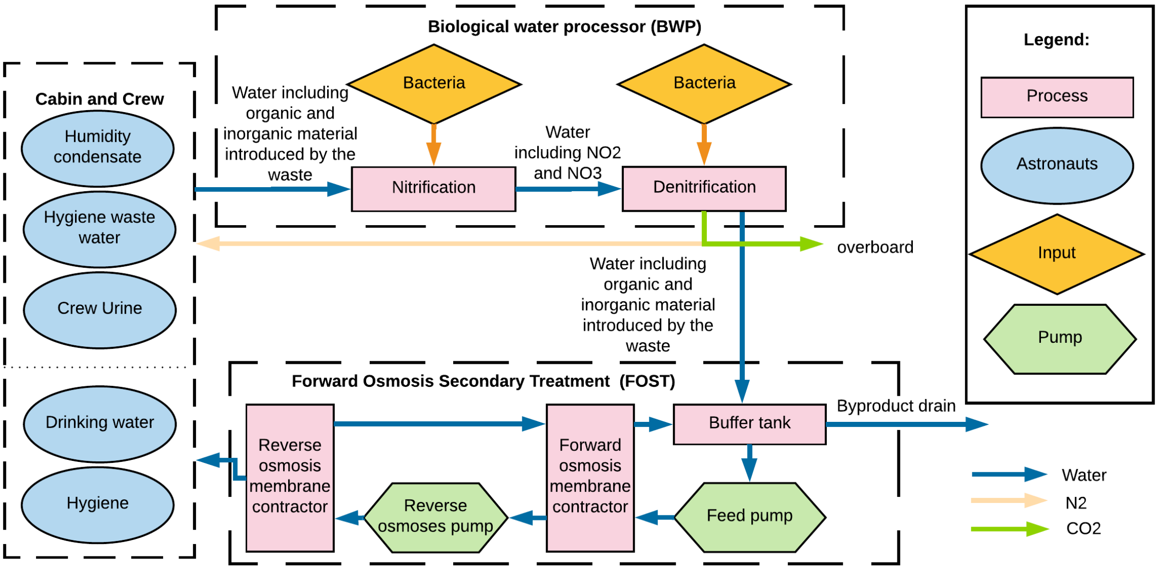


Figure 8.12: High level architecture of the alternative water processor, based on Meyer *et al.* [80]

Conclusively, it was chosen to use the alternative water processor, the main reason being that it has a higher recovery rate while using less resources. If the WRS was chosen as the working system, around 9800 kg of water would had to be transported, in comparison to the 1200 kg of water required for the AWP. However, at least four years of further development is deemed necessary to overcome the issues currently present in the system.

Waste Management

Based on Allen *et al.* [27], it is estimated that a crew of four astronauts, eating only pre-packaged food, will produce 3.2 kg organic and non-organic waste every day. Packing materials for food and non-consumables can be engineered and manufactured in such a way, that they are easily recyclable and can be fed to 3D printers. This will greatly increase the expandability and efficiency of the habitat, since new elements for future modules can be made instead of launching them. However, since it cannot be assumed that the 3D printer material has a purpose directly after creation, it does not necessarily reduce the mass of waste produced. Therefore it is not directly feasible for the one-year mission, but it might be interesting for a bigger, longer sustainable settlement on the Moon or another celestial body. Thus, recycling will not be taken into account for the sizing of the system because of the lack of research on this topic.

With the number of 3.2 kg per person per day, an estimated 4.7 tonnes of waste will be produced over the one-year mission. When compacted, the waste will have a density of approximately 500 kg/m^3, and thus 9.4 m^3 of waste will be produced over a one-year period. As will be elaborated upon in section 9.3, the return vehicle is capable of transporting 14.5 tonnes. Thus, the return vehicle will also be used as waste transport back to Earth after one year of operations. Waste storage spaces can be installed in the habitat, but given the return vehicle is present at all times, these storage spaces do not have to account for all waste. Rather, they would only need to be 1 m^3 in volume. Roughly every 4-5 weeks, astronauts perform an EVA to empty the in-habitat storage spaces in the return vehicle.

Pressure

It was found that the human body's functionality will decrease when being in a low pressure environment for prolonged periods of time. Therefore, the internal pressure has to equal 101.4kPa. Furthermore, many systems are build to function at this specific pressure reducing the need for development of new systems. However, having a relatively high pressure means the pre-breathing time for an EVA will be substantial, due to low pressures in EVA suits of around 25 kPa. However, NASA is currently in the final phase of developing space suits operating at a pressure level of 56.8 kPa [81], which would reduce the required pre-breathing time. The pressurisation system consists of a ventilation system, valves and a controlling mechanism. In order to pressurise the habitat, the ventilation system blows more air into the habitat than it takes out. The valves ensure a maintained pressure difference between parts in the system. Furthermore, the system works closely together with the atmospheric management and environmental monitoring system to measure the pressure in all modules of the habitat and ensure proper ventilation. Finally, the system contributes to the maintenance of a 21% oxygen level [82]. It measures the oxygen levels in the modules and modifies the incoming air when changes are needed.

The complete ventilation system will consist of two subsystems. Both subsystems can be specifically installed for one module, meaning different temperature and ventilation combinations are possible. The first subsystem will ensure pressurisation of the habitat and a correct air composition. The second subsystem will ensure the air is distributed properly throughout the habitat by means of ventilators, creating a homogeneous temperature distribution [82]. Following [27], the ventilation system needs to refresh 0.17 m^3 air every minute. Next to this, the maximum velocity of the air at the ventilator is 0.2 m/s. This means the maximum diameter of the ventilation system will be 14 cm.

Atmospheric Management

To ensure an Earth-like atmosphere, the internal air composition was chosen to have 21% oxygen and 79% nitrogen. The humidity level is set at 50% since this has found to be most comfortable for the astronauts [27]. As air will leak out the habitat and assuming approximately 2 kg of air will be lost each day, extra nitrogen and oxygen need to be taken to the Moon to make up for this loss [83, 84].

The first design option would be to use the same system as is currently being used on the ISS [84]. This option was disregarded since the current oxygen system has a too low recovery rate and the reliability of parts of the CO_2 system are too low as well [85]. Most suitable systems NASA is developing to increase the performance have a low TRL [86, 87]. Since only limited budget and time are available it was chosen to use the ACLS [88], which is currently undergoing the last tests under contract of ESA. The ACLS has three main functions: control the CO_2 levels to acceptable numbers (CCA), process the CO_2 to create water (CRA) and by use of an electrolyte create oxygen (OGA). A high level architecture of these processes can be seen in figure 8.13.

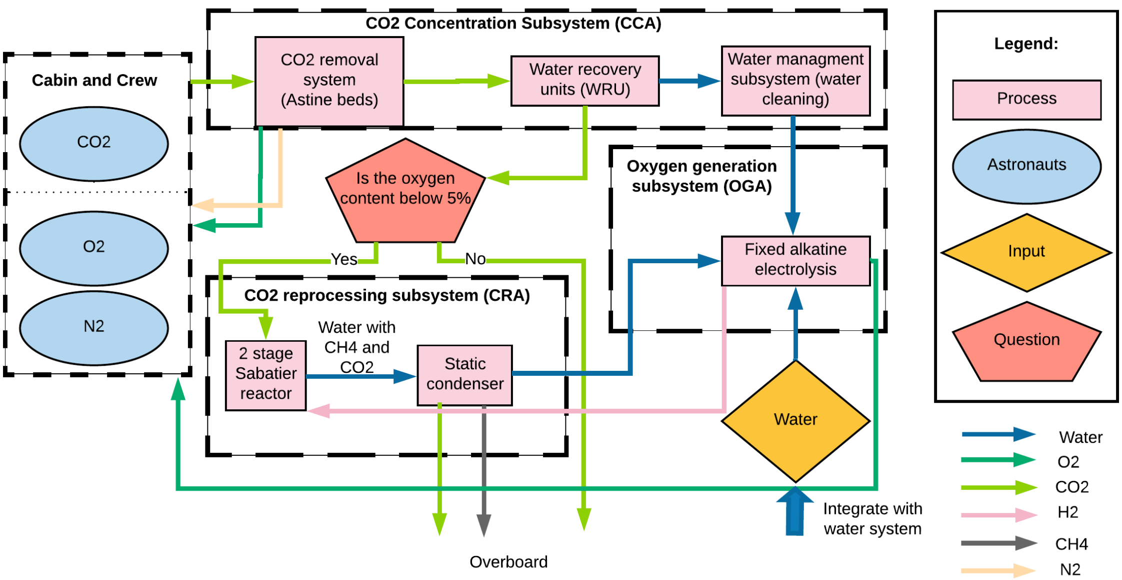


Figure 8.13: High level architecture of the oxygen generation and CO_2 removal system

Both the CO_2 removal system and the oxygen generation system are working sufficiently well for an Lunar habitation mission. The CO_2 reprocessing however does not reach the set target of 75%. This target cannot be reached by using sabatier reactors alone due to the reaction ratios. For this reason an additional interface to allow an add-on device has been integrated. Tests on the add-on device are planned to take place on the ISS from 2019 onwards. In a later stage of the design, an add-on system is to be chosen. For now, all calculations as performed for the budget will be based on the system without an add-on device to account for the worst case scenario. A total amount of 876 kg of water needs to be taken for the system to supply oxygen for a year. The ACLS is designed for three people and therefore needs to up-scaled. Furthermore, an integration with the water and temperature and humidity control system needs to be developed. In the event of a fire, the emergency plan as used in the the ISS will be followed [89], as can be seen in figure 8.10.

Inner Thermal Control

As the temperature underneath 120 cm regolith is found to be 248 K, a temperature control system is needed. This thermal control system will consist of two active thermal control systems (ATCs), as shown in figure 8.14, and a passive thermal control system in the form of regolith as insulation. ATCS 1 will provide liquid cooling by pumping ammonia through heat pipes. Ammonia passes all thermal and safety requirements as set by NASA [90]. It has to be ensured however, that no leaks will cause the ammonia to enter the habitable space of the modules. This means the heat pipes have to be properly integrated into the structure. By insulating all systems within the habitat, the created heat of the system is trapped. This ensures the ammonia to easily take up the heat when it passes by the system, effectively cooling the system down. Even though it is assumed the systems only have to be cooled down, a heater is available to provide for the option to heat up the systems.

ATCS 2 is present to ensure the right temperature is achieved within the habitat itself. This is done by another set of heat pumps and heat pipes, filled with water. As explained in section 7.4 the water shall not drop beneath 273.15 K. The water can be actively heated or cooled and then distributed throughout the habitat. To cool the system, the water pipes will be guided through the outside shell, 1.20 m under the ground. There, the average temperature equals 248 K thus cooling the water rapidly. This was fully elaborated upon in section 7.4.

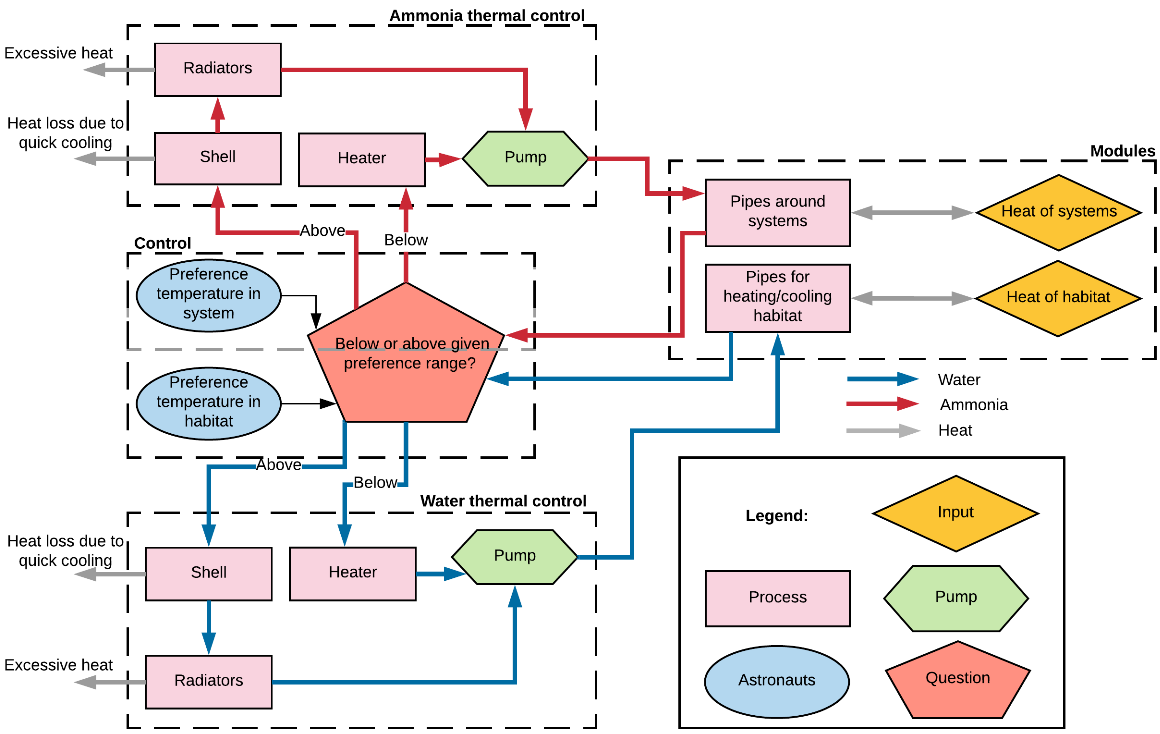


Figure 8.14: Top level architecture of the thermal control system

Environmental Monitoring

To make sure all described subsystems are functioning properly, the environment in the habitat has to be monitored closely. This means in every module sensors will be present to measure the temperature, humidity, oxygen, nitrogen and toxicity levels as well as pressure levels, ventilation rates and fire detection available. Furthermore, to analyse the measured values, a voltaic organic analyser which uses an ionmobility spectrometer is needed as well as a major constituent analyser and compound specific combustion product analyser [91]. However, these devices are not able to measure biological growth. To do so a sample can be taken, using an adhesive sheet as used in KIBO [92]. These samples need to be send back to Earth every three to six moths. Since this is not ideal, further development needs to be done in order to create a lightweight system to perform on board processing. If, in future expansion, a laboratory would be included less development is necessary for such systems.

A GUI will be installed so the astronauts have the ability to change the temperature and ventilation in every module and sleeping cabin as they seem fit. The fire detection sensors will be placed in the ventilation shafts, as hot air does not go up, but is sucked into the ventilation system. After one year of habitation, the habitat will be cleaned thoroughly to ensure no contamination is left in the habitat when a new mission starts. This is done by using the same scanning system that is used in hospitals [93]. When no contamination can be found anymore, the habitat is accepted as clean.

Exercise Equipment

A compartment including exercise equipment is designed to limit muscle atrophy due to micro gravity. The first piece of sporting equipment that will be present is the Treadmill with Vibration Isolation and Stabilisation System [94]. The second is the ISS Advanced Resistive Exercise Device, which focuses on training all main muscle groups [95]. Lastly, the Cycle Ergometer with Vibration Isolation and Stabilisation will be installed for endurance training [96]. This combination of sporting equipment is chosen upon to ensure varied and proper training to maintain a good overall physical health, while at the same time not interfering with scientific equipment. Even though the systems are designed to limit the vibrations and noise levels, they will be placed away from the scientific equipment and the sleeping quarters. Extra ventilation will be present to counteract excessive sweat formation [27].

Medical Equipment

In the ISS, only a limited amount of medical equipment is available including a first aid kit, equipment to sew wounds, a book with all possible diagnoses, and some powerful equipment, such as a defibrillator [97]. During more serious events, which cannot be handled by the crew, the harmed crewmember will either be send to Earth via the Soyuz docked to the ISS, or pass away. During the ride home the astronaut will experience many G's, and this too can cause the crewmember to die [98]. Next to the medical care during injuries, the crew is constantly monitored [99]. This ensures diseases can be found in early stages, ensuring countermeasures can be performed before it becomes a problem.

For a longer mission at a greater distance, this will impose problems. Right now, the chance on a serious, life threatening injury is estimated to be 1 to 2% for an astronaut per year [97]. Crewmembers will not be able to get back to Earth in time, and more dangerous situations are expected to occur. This means more medical equipment will be needed for the Lunar mission.

To stabilise a crewmember after an injury, the complete medical equipment of an Advanced Life Support Ambulance is needed [100]. However, one problem exists. Blood in zero or micro-gravity behaves differently than it does on Earth, making it impossible to perform surgery during the mission as is done on Earth. Also, many particles are floating around the station that will not be around in a clean operating room, mostly being dead skin cells [98]. This can cause infections.

Many research is done in different ways of performing an operation, including robotic operation and laparoscopy [101, 102]. However, the aforementioned problems remain. A partial solution would be to design a clean room, only used for this kind of operations, separated from the rest of the habitat with an airlock. While artificial gravity is far from being feasible, it would be the perfect solution for the problem. However, it will not be considered further in this project. Other possible solutions have not been found yet, and more research will be needed for the habitat to become a reality. Therefore a research and development budget has been set aside to solve the mentioned issues.

Noise Control

For missions with a duration of one year, the ambient noise level must be below noise level NC-50, with the availability of a quiet room. In the quiet room and the sleeping cabin, the noise level shall be below NC-40 [27]. In order to do so, the same structure as used in the ISS will be used, which is shown in figure 8.15 [103]. It consists of three parts. Part 1 is the left acoustic barrier, made from five layers of material, having a total thickness of 9 mm and a mass of 3.17 kg/m^2. Part 2 is the wall itself. The thickness of this wall depends on the structure and strength it needs, dimensions are provided in section 7.5. Part 3 is the right acoustic barrier, consisting of three layers of material, with a total thickness of 13 mm and a mass of 1.22 kg/m^2. The outer layers and cores for both acoustic barriers are the same. As outer layer, HT90-40 Nomex is chosen, as this has low permeability, it is easy to clean and it ensures no water will get into the isolation material. The cores are made of F400-11 Durette Felt. SOLIMIDE TA-301 was considered, but disregarded after it was found to be twice as heavy. In part 1, two extra layers of BISCO are applied. BISCO material blocks the lower frequencies, while Nomex and F400-11 Durette Felt block the higher frequencies.

The first part of the isolation will be placed on the inside of the outer walls, where systems are creating noise. Part 3 will be placed on the outside of the wall, absorbing noise created in the habitat itself. While passive noise control is already applied in the ISS, active noise control of complete modules is still being researched. Right now, there are companies (such as Silentium Ltd.) that have systems that can cancel around 90% of the noise in the low/middle frequency regions. Whether this also works in microgravity will have to be researched, but this is a promising concept in which quiet places in the habitat can be created. Lastly, hearing protection needs to be provided in case the noise levels are higher during certain activities.

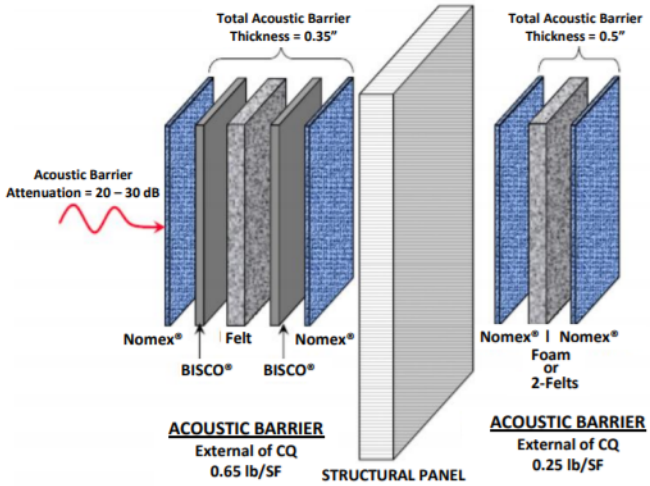


Figure 8.15: Multi-layer passive noise control [103]

8.3.5. ECLSS - Technical Budgets

In this subsection, the respective calculated values for the ECLSS and consumables are given:

- **Mass:** 19938 kg;
- **Volume:** 78 m^3;
- **Power:** 6028 W, with an additional 5200 W usage during specific situations;
- **Cost:** €5.1 billion, or 213 $k€/kg$.

The budgets are explained in the following sections. All values given are excluding contingency factors, except for the cost.

Mass

Based on the number of astronauts, an estimation on the weights of the ECLSS and all consumables was made using the Boden [104] guidelines. The weight of the ECLSS includes the inner thermal control system, the atmospheric composition and pressurisation system, ventilation system and the recycling systems. The weight of the ECLSS (M_{ECLSS}), based on four astronauts and a regenerative system, equals 2531 kg.

The weights of the consumables are based on the recovery rate that can be achieved with the used systems. That is: 90% of the water and 42% of the oxygen is recycled. All tanks for water, oxygen and nitrogen are included in their respective weights. All food shall be brought from Earth. Also, a leakage of 2 kg of air per day is assumed. It follows that M_{water} equals 3025 kg, M_{oxy} equals 1340 kg and that M_{nit} equals 1339 kg. These estimations are based on the calculations on the tanks as explained later in 8.6. M_{food}, with the packaging included, equals 4198 kg [105].

Mass of the other consumables like medical and cleaning supplies can be estimated on basis of a long- or short-term mission and whether laundry can be done in the habitat. This results in a M_{misc} of 3032 kg. The fixed accommodations (freezers, showers and exercise material) are estimated to have a weight of 1843 kg. The medical equipment needed for the mission ($M_{medical}$) is estimated to be equal to 750 kg [27]. The extra mass needed for the noise control is equal to 455 kg per module and is accounted for in the weight of the secondary structure of the module. Lastly, the thermal control subsystem is estimated to weigh 1880 kg, making the total mass of ECLSS related components 19938 kg.

Volume

Based on the available pictures of the complete ECLSS, V_{ECLSS} is estimated to be 10.8 m^3 [106]. Assuming a food and packaging volume of 0.01 $m^3/p/d$, V_{food} is estimated to be 14.6 m^3. Taking the density of oxygen to be at 200 bar and nitrogen at 230 bar, V_{oxy} equals 3.62 m^3 and V_{nit} equals 6.61 m^3. V_{water} and V_{misc} are 2.69 m^3 and 23.62 m^3 respectively. The fixed accommodations are estimated to be 8.52 m^3 [104]. $V_{medical}$ equals 3.3 m^3 [27]. The volume needed for noise control is equal to 2.3 m^3 per module, but this is again taken account of within the structure itself. Adding 4 m^3 for the control subsystem, a total volume of 78 m^3.

Power

Based on the mission characteristics, the power of the ECLSS, P_{ECLSS}, is estimated to equal 3957 W. Furthermore, the fixed accommodations are estimated to continuously need 350 W [104]. $P_{medical}$ equals 300 W of continuous power [27]. The thermal control system needs a continuous power provision of 1420 W. However, An additional 5200 W power usage is estimated to be needed when extra systems are used, such as a laundry machine and some supplies from the medical suite. In total the system requires an operating power of 6028 W, and a peak power of 11228 W.

Cost

The cost of the development of a Lunar life support system within the next 5 years is estimated to equal 4.3 billion euros [107]. Then, the costs of the consumables are estimated. C_{water} equals €3000 , C_{oxy} equals €600000, C_{nit} equals €120000 and C_{food} equals €230000 [105]. As the cost of the development of the system and the supplies is expected to be much bigger than the miscellaneous supplied and fixed supplies, it was decided to neglect these two costs. Adding in a 20% contingency leads to a total estimate of €5.1 billion.

8.3.6. ECLSS - RAMS

Given that the ECLSS has a lot of different subsystems, only the most critical systems haven been analysed using a FMEA. In figure 8.16, 8.17, and 8.18 the FMEA for the medical suite, water processing system and environment regulation systems are presented. The explanation for typical scores can be found in subsection 8.1.5.

Function ID:	Function	Failure Mode	Potential Effect of failure	SEV	Potential causes of failure	OCC	Current Control	DET	RPN
FF_FE1	Install system	Installation failure	Incomplete medical suite	3	Medical equipment broken	1	Redundant suite and network inspection	1	3
			No medical suite	3	Connection error	1	Network inspection	1	3
FF_FE2	Set diagnoses	Wrong diagnoses	Loss of life	4	Human error	2	Double-check by Earth-based doctor	2	16
					Lack of communication with doctor	1	Network inspection	1	4
FF_FE3	Solve medical issues	Unsolved medical issues	Loss of life	4	Human error	2	Double-check by Earth-based doctor	2	16
					Unnoticed medical condition	2	Network inspection	1	4

Figure 8.16: FMEA of the medical suite

Function ID:	Function	Failure Mode	Potential Effect of failure	SEV	Potential causes of failure	OCC	Current Control	DET	RPN
AWP1	Process water using bacteria	First purification failure	Damage to FOST system	2	Not enough bacteria	2	Network inspection	1	4
					Extreme pollution of water	2			4
AWP2	Process water using osmosis	Second purification failure	No clean water	2	Non-processable input	2	Network inspection	1	4
					Wear & Tear	2			4
AWP3	Pump water AWP system	No water circulation	No clean water	2	Pump failure	2	Network inspection	1	4
					Pipe congestion	2			4

Figure 8.17: FMEA of the alternative water processor

Function ID:	Function	Failure Mode	Potential Effect of failure	SEV	Potential causes of failure	OCC	Current Control	DET	RPN
AR1	Recycle internal atmosphere	Incomplete/no recycling of internal atmosphere	CO2 Poisoning	3	No CO2 scrubbing	2	Network Inspection	1	6
			Lower system efficiency	2	CRA Failure	2	Network Inspection	1	4
			Use of emergency supply	2	Lack of water supply	2	Network Inspection	1	4
AR2	Regulate internal pressure	No/Incomplete regulation of internal pressure	Local pressure difference	1	Vent congestion	2	Manual inspection	2	4
			Severe loss of pressure	3	Wrong measurements	1	Network Inspection	1	3
AR3	Regulate internal environment	No/Incomplete regulation of environment	Hazardous environment	3	Wrong measurements	1	Network Inspection	1	3
			Loss of life	4	Undetectable lethal element	2	Network Inspection	1	8
			Loss of life	4	Control system failure	1	Network Inspection	1	4

Figure 8.18: FMEA of environmental management system

8.3.7. ECLSS - Risk & Mitigation

As can be observed in the various FMEA's presented above, there are numerous failure modes that have a high Risk Priority Number. The next few sections deal with mitigating these risks.

Medical Suite

Clearly, the highest RPN can be found for a wrong diagnosis or medical issue caused by human error, which is currently controlled by having a doctor double-check any diagnosis made by the astronaut. Some ways to mitigate this risk are to let multiple doctors check diagnoses and to make sure that at least one of the astronauts has received extensive medical training.

Water Processing System

Given that all the risks presented in the FMEA for the alternative water processor have similar RPN's, there is no specific risk that needs to be mitigated. However, the entire system is rather sensitive, meaning it requires regular inspection and maintenance.

Environmental Management System

High RPN's can be found for two risks in figure 8.18. Firstly, there is a risk of CO_2 poisoning caused by failure of the scrubbers. This risk can be mitigated by having redundant scrubbers and by performing regular maintenance on the system. The next risk that has a high RPN is the loss of life due to an undetectable lethal element being present in the atmosphere. The only way to truly mitigate such a risk is by making sure the measuring system has extremely high accuracy.

8.3.8. ECLSS - Product Verification

For all but two requirements on SYS-OP-S-IA are validated by the fact that there are sensors accurate enough to measure all values. Because all systems required to react to possible unwanted value, are present, these requirements are met. SYS-OP-LS-IA-4 is met by the fact that an oxygen supply of 880 kg is present. SYS-OP-LS-IA-5 is met by applying sufficient ventilation in every module, which is achieved by the ventilation system that blows around the air. With 2700 kg of water, enough is supplied to satisfy SYS-OP-LS-W-1, 4 and 8. SYS-OP-LS-W-6, 7, 9-12 are met, based on the fact that sensors are installed in the habitat that make sure the right values are achieved. All systems needed to achieve the values are incorporated in the design. SYS-OP-LS-W-3 and 5 are met in the safety module, where the complete ECLSS is installed again, but downscaled for emergency events. This redundant system is explained in subsection 10.4.4.

As a redundant food supply is always available at the return vehicle, SYS-OP-LS-F-1 is met. SYS-OP-LS-F-2 until 6 are met by choosing the food based on these requirements. As no preliminary foodlist is available yet, these requirements are assumed to be validated when the next level of design is achieved. SYS-OP-LS-WST-1, 2 and 4 are met by collecting all waste in the waste disposal vehicle and sending this vehicle back to Earth, where it burns up in the atmosphere. Using hospital equipment and towels, SYS-OP-LS-WST-3 is met too.

As modules can be disconnected by means of a double, airtight door, single modules can be depressurised, meeting SYS-OP-LS-P-2. SYS-OP-LS-P-2 and 3 are met by means of an airlock with 10.6 m^3 volume and a sensor measuring the pressurise inside the airlock. All sensors measuring pressure have a accuracy of 1 Pa, meeting SYS-OP-LS-P-4. With sensors having an accuracy of 1 K, both SYS-OP-LS-T-1 and 3 are met. SYS-OP-LS-T-2 is met with the downscaled, redundant ECLSS. Both SYS-OP-LS-CF-1 and 2 are met by design. The design itself is based on a noise-level of 40. With all systems included in the design, SYS-OP-LS-HC-1 and 3-10 are met. SYS-OP-LS-HC-2 is met by having equipment in the medical suite with all systems needed to measure all data.

8.3.9. ECLSS - Product Validation

System validation will be preformed in multiple ways. First, as most systems are developed by partners, they will validate their product. Then, several systems are planned on being tested on the ISS, which would also serve as a validation. Finally, during the year, mock up mission in Antarctica, the ECLS will be validated.

8.4. Communications System

This section entails the design of the communications subsystem.

8.4.1. Communications System - Inputs and Requirements

The first step to designing this subsystem is identifying the requirements and constraints regarding the system. The relevant requirements from the Baseline Report are stated in appendix B [2]. Some of them have been revised, this is discussed later in the text.

Requirements SYS-OP-COM-9 and -10 are updated from 5 $Mbps$ and 3 $Mbps$ to 100 $Mbps$ each. More elaborate research towards the data rate has revealed the preliminary estimation to be too small. Moreover, SYS-OP-COM-7 is interpreted in such way that all astronauts have access to the communication subsystem, but not at all times. However, the communication subsystem has to be operational at all times, to satisfy SYS-OP-COM-8.

8.4.2. Communications System - Functional Flow Structure

Figure 8.19 shows a functional flow diagram of the communication subsystem. It is very global, and can be interpreted for both the uplink, the downlink and interpersonal communication between EVA astronauts.

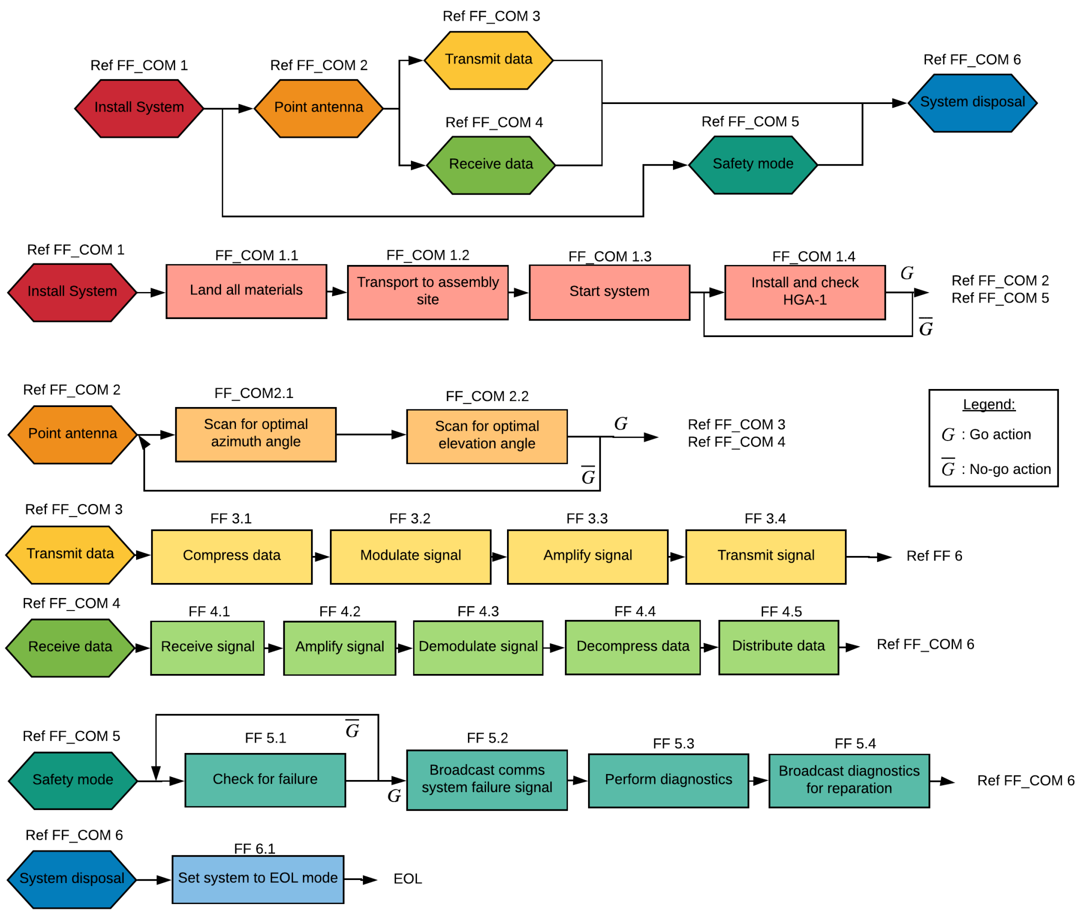


Figure 8.19: Functional flow of the communications subsystem

Figure 8.20 shows the communication flow diagram, which was updated from the diagram presented in the Midterm Report [3].

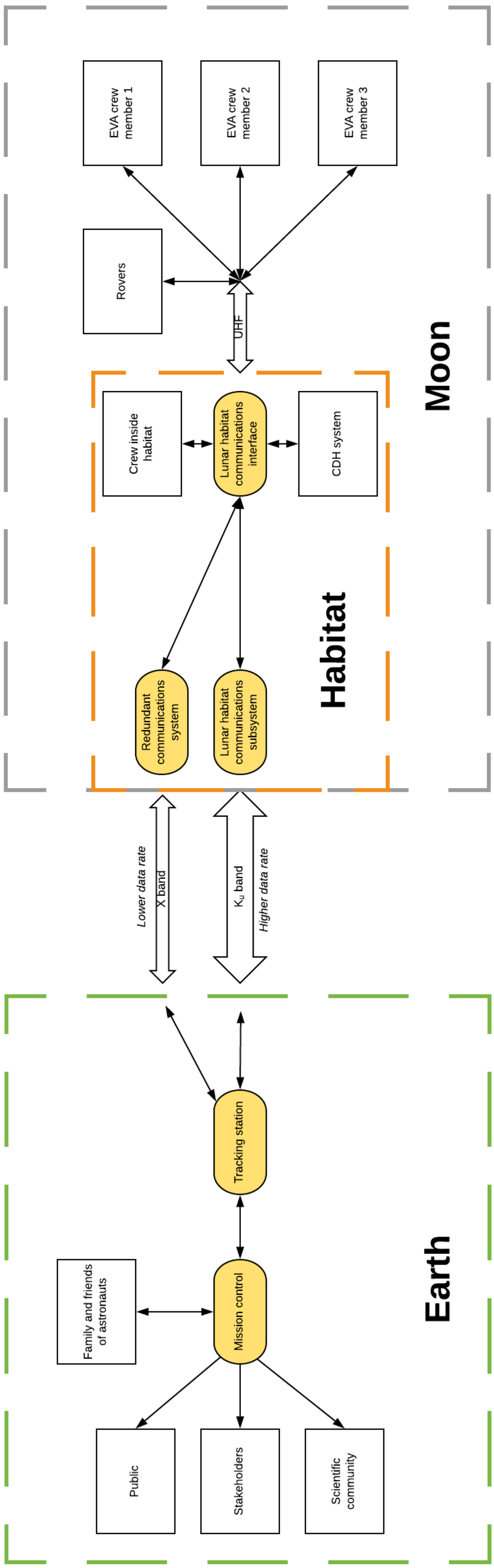


Figure 8.20: Updated communication flow diagram

8.4.3. Communications System - Interfaces

The design process N2 chart presented in the Midterm Report [3] identifies five main design interfaces that give inputs to the design of the communications subsystem. These will be listed and it will be explained how they affect the communications subsystem.

- **Design power and thermal subsystem**: the performance of the communications system is very dependent on the available power for radiation and also on the temperature of the system, since a higher system temperature induces a higher noise temperature, which makes the system require more power for sending a detectable signal;
- **Design safety system**: for safety, the communications system should most importantly be redundant. Since the main system will be tailored for high data rates and will thus be large and heavy, it is beneficial to also have a lower data rate system that is more conservative with power and is more mobile;
- **Design computer and sensor system**: most data that is gathered by the sensors and processed by the computer will have to be sent to Earth. Therefore it is important to know how much of this 'housekeeping' data the sensors and computers collect during a specified time. Usually, the housekeeping data is a secondary data type, not taking up much of the bandwidth, but since this mission is a pioneering effort, it may found to be different;
- **Determine astrodynamic characteristics**: the astrodynamics influence the communications of the mission by distance. It determines the sizing and power consumption of the system greatly, and with size also the mass increases;
- **Determine landing site**: the landing site determines the architecture of the communications because if a landing site is chosen that offers no direct line of sight to Earth, relayed communications are necessary. This imposes more complexity, cost and less performance on the system, and thus is highly undesirable for the communications system. Fortunately, the Apollo 11 landing site faces the Earth permanently, due to the tidal lock of the Moon.

8.4.4. Communications System - Design Layout

The communication subsystem will consist of the following components:

- A 1.5 m diameter main parabolic antenna (High-Gain Antenna 1, referred to as HGA-1);
- A 0.5 m diameter secondary parabolic antenna (High-Gain Antenna 2, referred to as HGA-2);
- A 0.4 m long helical antenna (Low-Gain Antenna, referred to as LGA).

Given the size and mass of HGA-1, it will be stowed during transport to the Moon, and will have to be manually installed during an EVA. It will be stowed on top of the airlock module or placed on the ground outside the habitat. This package will then be delivered to the building site in the same fashion as the other parts. The dish will be mounted on a support pointing structure.

HGA-2 will be mounted on the outside of the cylinder, to be deployed automatically. It will be used to establish the first communication with Earth until the crew arrives to install HGA-1. During the mission, it will also serve as an extra channel for housekeeping data. A LGA will also be readily deployable upon placement of the main cylinder, but is only to be used after the crew arrives because it is optimised for EVA voice communications over a distance of 5 km maximum in all directions.

8.4.5. Communications System - Installation

Manufacturing of the antenna is not a challenge, given the knowledge about space communication systems is readily available. Installation once on the Moon, however, proves to be more of a challenge. HGA-1 will be stowed in a separate package delivered with the airlock module, and will have to be installed manually before being fully operational. This will be done by a team of astronauts on EVA, after all internal systems are verified by the crew. HGA-1 can be dismounted from its stowage location and mounted on the pointing structure that can either be mounted on the main cylinder or placed on the Lunar surface, the latter configuration is shown in figure 8.21

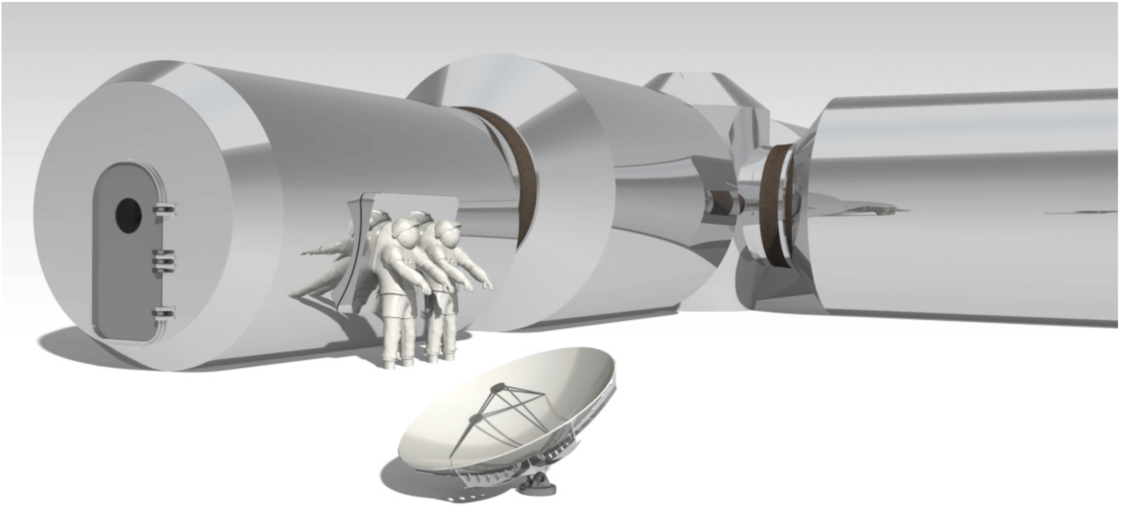


Figure 8.21: HGA-1 placement with respect to the airlock

HGA-2 can be deployed automatically once the main cylinder is in place and will start connecting to the ground stations to provide them with housekeeping data. The helical antenna for EVA communication can also be deployed automatically upon placement of the main cylinder.

8.4.6. Communications System - Technical Budgets

In this subsection, the technical quantification of the system will be presented.

Link

The link budget described in the Midterm Report [3] has been updated. Additionally, a link budget for the secondary communication system has been set up, both link budgets are presented in table 8.2. From the resulting signal-to-noise ratio (SNR), the channel capacity can be determined by the Shannon-Hartley theorem, summarised by equation 8.1.

$$C = B \cdot \log_2 (1 + SNR) \tag{8.1}$$

In which C denotes the channel capacity in bps and B denotes the bandwidth that can be found in table 8.4.

Table 8.2: Link budgets for the main and redundant communications channels

Parameter	**Unit**	**Main channel**		**Secondary channel**	
		Downlink	**Uplink**	**Downlink**	**Uplink**
TX power	dBW	20.00	48.90	16.99	48.90
TX loss	dBi	-0.97	-0.46	-0.97	-0.46
TX gain	dBi	44.85	65.58	29.85	60.12
Pointing loss	dB	-0.14	-0.14	0.00	0.00
Free space loss	dB	-228.16	-228.16	-222.70	-222.70
RX gain	dBi	65.58	44.85	60.12	29.85
RX loss	dBi	-0.46	-0.97	-0.46	-0.97
Signal strength	dBW	-99.29	-70.39	-117.17	-85.26
Noise strength	dBW	-131.73	-125.16	-131.73	-125.16
SNR	dBW	32.45	54.77	14.57	39.90
Link margin	dBW	22.15	44.47	5.37	30.90

From this link budget, the main channel has an estimated capacity of 388 $Mbps$ and the secondary channel has a capacity of 176 $Mbps$.

Mass

A mass estimation based on the antenna dimensions is made for all three antennae [108]. The results are given in table 8.3. A contingency factor of 2 was applied to this estimated total mass to account for the mass of the pointing structure. Given the uniqueness of this mission, it was impossible to give estimates from historical data.

Table 8.3: Estimated mass of the components of the communications subsystems

Item	**Mass** $[kg]$
HGA-1	14.07
HGA-2	2.16
LGA	0.02
Subtotal	16.25
Contingency factor	2
Total	32.50

Volume

Similar antennae to HGA-1 (that are not specifically optimised for space use which would not increase size significantly), have a stowed size of about $2 \times 1.5 \times 0.5\ m$, equalling 1.5 m^3 of volume. For HGA-2, it is assumed that one third of the volume is required (0.5 m^3). With the volume of LGA being negligibly small, the total volume of the antennae will be 2 m^3.

Cost

Given that space communication is a field that has widespread commercial applications (phones, television signals), it is expected that the communications system will be rather cheap. Antennae similar to the ones that are used on the habitat (although, not space-grade), are available in the range of € 100 - € 2500, so assuming that space-grade technology will be ten times as expensive, for two antennae, the total cost of the communication system will be € 50 thousand.

Specifications

All remaining specifications regarding the communication subsystem are collected in table 8.4.

Table 8.4: High level specifications of the main channels and antennae of the communications subsystem

Parameter	**Unit**	**HGA-1**	**HGA-2**	**LGA**
Radius	m	0.75	0.25	0.005
Length	m	-	-	0.40
Peak power	W	100	50	10
Band	–	K_u	X	UHF
Frequency	Hz	15 G	8 G	400 M
Bandwidth	Hz	36 M	36 M	5 k
Modulation	–	DPSK	8FSK	BFSK
Data rate	bps	388 M	176 M	97 k
Round-trip time	s	2.71	2.71	≈0

8.4.7. Communications System - Sensitivity Analysis

The sensitivity analysis as described in section 7.1 was conducted on the link budget tool with which the communication system was designed. Only the HGA-1 downlink was investigated, as the principle governing this channel is the same for all channels. The input variables were chosen to be either design variables, or variables that have been estimated and thus have a significant uncertainty, they are listed below. The output variable was chosen to be the data rate supported by the channel.

- Communication distance;
- Pointing accuracy of the transmitting antenna on the Moon;
- Transmitter efficiency;
- Receiver efficiency;
- Transmitter loss factor;
- Receiver loss factor;
- Noise temperature;
- Transmitter diameter;
- Receiver diameter;
- Transmission frequency;
- Radiated power.

The sensitivity analysis shows that the first-order sensitivity indices for the transmitter diameter, the frequency and the communication distance are especially high: 0.23, 0.17 and 0.18 respectively. This was to be expected since the diameter of the transmitter contributes strongly to the ability to direct the signal. Increasing communication distance and frequency greatly increases free space loss. Besides, the frequency also influences many parts of the link budget, which can be seen in the total sensitivity index, which is the highest of all alongside the transmitter diameter.

8.4.8. Communications System - RAMS

The FMEA for the communication system can be found in figure 8.22. Again, the scoring scheme can be found in subsection 8.1.5.

Function ID:	Function	Failure Mode	Potential Effect of failure	SEV	Potential causes of failure	OCC	Current Control	DET	RPN
FF_COM2	Point Antenna	Pointing Error	Low data rate	2	Scanning Failure	1	Measure Data Rate	1	2
			No comms.	3	Motor Failure	2	Visual Inspection	2	12
FF_COM3	Transmit Data	Failed Transmission	Low Downlink Data rate	2	Low Power	1	Measure Amplifier Power	1	2
			No 2-way comms.	3	Software Error in modulation	1	Test Signal	2	6
FF_COM4	Receive Data	Incomplete Received Data Package	Low Downlink Data rate	2	Low Power	1	Measure Amplifier Power	1	2
			No 2-way comms.	3	Software Error in modulation	1	Test Signal	2	6
FF_COM5	Operate safety mode	Inability to operate safety mode	Irreparable damage to comms.	3	Damage to all dishes	2	Visual Inspection	2	12
			Irreparable damage to comms.	3	Software error	1	Scheduled safety mode checks	2	6

Figure 8.22: FMEA of the communications system

8.4.9. Communications System - Risk & Mitigation

As can be observed in the FMEA for the communications system, a motor failure can cause a no communication situation, which is a critical risk. Mitigation can be achieved by two main actions: firstly, the communication system should be manually adjustable, and the motors should be checked up in a regular maintenance schedule.

Next, a software error in the modulation can cause a one-way communication situation. This is not ideal, and can be mitigated by including a back-up communication system (as is the case), thus decreasing severity. Damage to all the dishes can cause a critical failure situation with no communication at all. However, such a risk is easily mitigated by having a back-up system, as has been designed. The same goes for a software error causing a similar situation.

8.4.10. Communications System - Verification & Validation

Verification

Because the model that calculates the link budget is the same for every channel, only one channel was verified. Using the inputs as used in the calculation for this channel in another link budget calculator [109], a difference in received power of 0.15% was found, verifying the calculations from this model.

Validation

To validate the link budget model, a comparison was made with existing published link budgets for Moon-based missions [110–112]. Because link budgets often work with different assumptions regarding the technical specifications of the system itself, it is rather hard to compare them. For the link budgets cited, the given input specifications were fed to the model, and the resulting SNR was compared. All the investigated link budgets managed to achieve a higher SNR, even with the same input specifications. This difference can be attributed to the different assumptions made for these budgets. It does, however, show that the data rate estimations made in this section are rather conservative, and will not be deficient to the mission's requirements.

8.4.11. Communications System - Product Verification

The first set of requirements, SYS-OP-COM-1 until -3 are met because there are several ground stations providing 360° coverage around Earth at all times. In addition to that, the tidal lock of the Moon and the location of the habitat ensure that a direct line of sight is always available for communication.

The habitat sports the LGA for compliance with SYS-OP-COM-4. Given that the suits have adequate communication equipment, communication over 5 km is possible. HGA-2 provides a redundant data channel, validating SYS-OP-5. Requirement SYS-OP-COM-6 can be complied with because HGA-2 is mounted and ready as soon as the module is in place, so a smaller bandwidth communication channel can be opened with the habitat very early on in the mission.

Given that the channel capacity of the main channel exceeds the requirements, a bandwidth can be reserved to accommodate personal communications at all times, as per SYS-OP-COM-7 and -8. Finally, the data rate requirements SYS-OP-COM-9 and -10 are simply validated by calculating the data rates that are to be expected, which are compliant with the requirements.

8.4.12. Communications System - Product Validation

Within the tight time-schedule, validating the communication system would be very hard to do analytically. The link budget should be refined first to account for unpredictable attenuation, such as weather attenuation. Once the theoretical model has been worked out in sufficient detail, it can be validated by means of testing.

8.5. Command, Data Handling and Computer System

In this section, the CDHCS has been designed and its direct relation to all the other subsystems in terms of connection, power and feedback has been established.

8.5.1. Command, Data Handling and Computer System - Inputs and Requirements

This system interfaces with all other systems, and the requirements for these are mentioned in the relevant subsystem section. A failure in this system can be catastrophic, so extremely high reliability is required.The main constraints for this subsystem are the available power and the radiation level in the environment. Secondary constraints are the interfaces with other systems, however these can be adapted to suit the needs. The requirements for this system are mentioned in appendix B.

8.5.2. Command, Data Handling and Computer System - Functional Flow Structure

The functional flow of this system is very complex as it can be seen trough multiple perspectives. In order to fit the whole diagram, it was decided to split it up. The functional flow of the CDHS is shown in figure 8.23. Ref_CDH 2.0 in this figure encompasses what pertains to the management of specific subsystems, which are shown in the functional flow diagrams of the relevant subsystems.

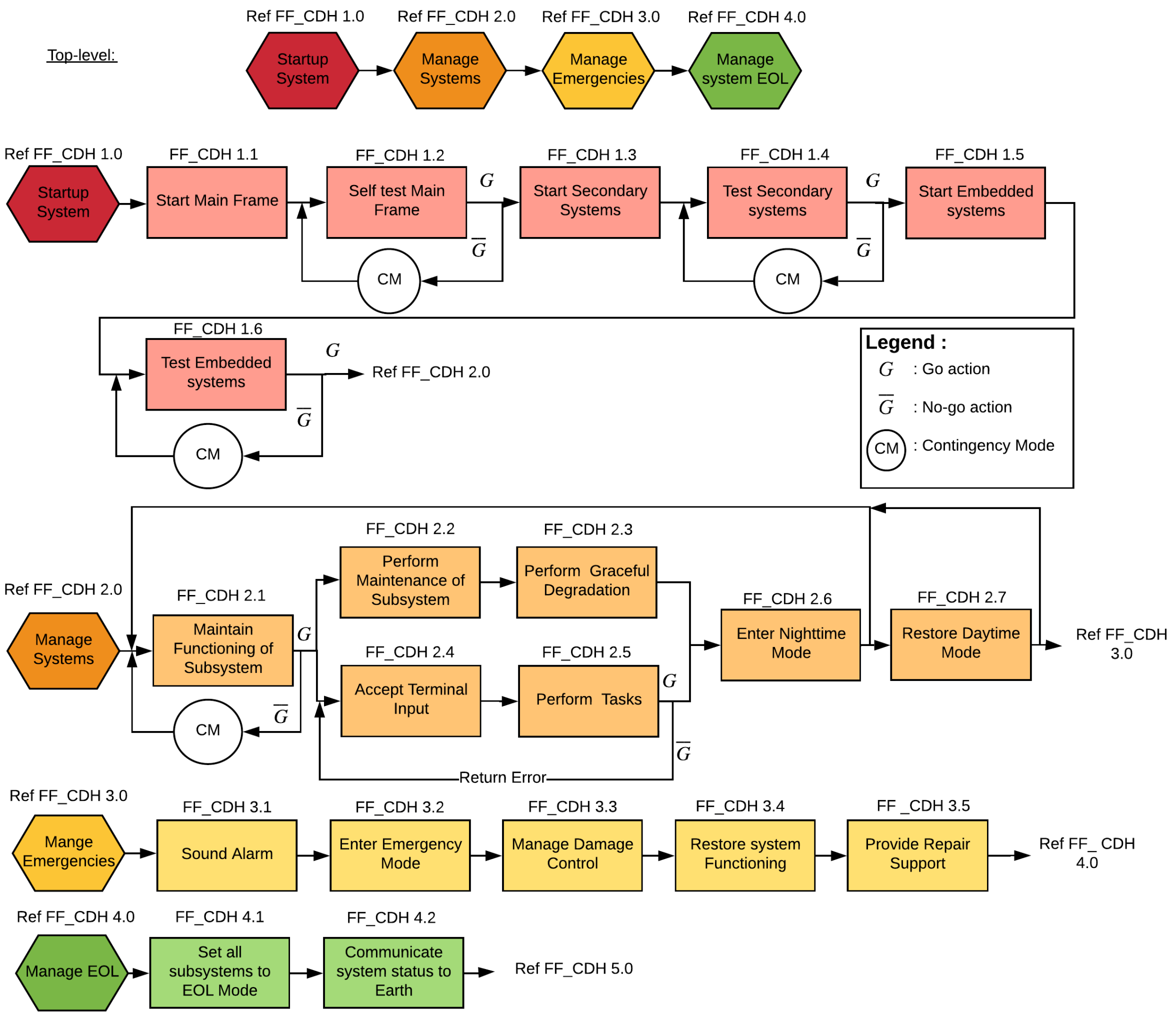


Figure 8.23: Functional flow of the command, data handling and computer system

8.5.3. Command, Data Handling and Computer System - Interfaces

As CDHS controls all other subsystems, it is interfaced to all other subsystems that consume power. The investigation of the interfaces was done using the N2 Chart of the system shown in Figure 9.1 of the Midterm Report [3]. Only the architecture diagram is shown, which contains all the interfaces and can be seen in figure 8.24.

8.5.4. Command, Data Handling and Computer System - Design Layout

The main challenge in designing spacefaring computer systems is the trade-off between computing power, energy consumption and radiation protection (reliability). The more radiation protection is provided to a system, making it Radiation Hardened (Rad-Hard), the higher the power consumption due to the more sturdy design of transistors and redundancy considerations and the lower computing power capacity, as less transistors can be fit in the same space [113]. As the habitat is designed to protect astronauts from radiation, the radiation levels within are significantly below the values which would merit radiation hardened components. However, due to the aforementioned critical nature of this system, it is designed for a worst case scenario in which radiation does seep into the system unobstructed (e.g. breach of one of the modules) and the system needs to continue to function.

To complicate matters further, the computer system has to handle a vast amount of extra functions since there are astronauts present in the system. Taking into account life support considerations, entertainment, emergency and contingency management, great amounts of sensors and computing power are required. The design is not comparable to traditional satellite computer/CDH systems. The closest comparison is the ISS, which uses more than 50 computers controlling 350,000 sensors using more than 1.5 million lines of flight software code [114]. Since very little information is available about the ISS' system specifications, the only alternative is to expand the architecture of traditional satellites and design improvements to cope with the more stringent requirements.

Architecture

After the requirements, interfaces and flow chart were drawn up, the functions were grouped together according to their similarities. With all this information at hand the high level architecture was drawn up and finally safety and redundancy was built into it. The architecture can be seen in figure 8.24. In order to grasp the architecture, first the components will be explained and then the explanation for the configuration will be given. Data handling block diagram is not presented in this project as it is so complex that two Doctorates from the TU Delft (mentioned in the preface), one in aerospace systems engineering and one in integrated circuit design, strongly stated that the design of such as system would require significantly more time than the scope of the DSE allows.

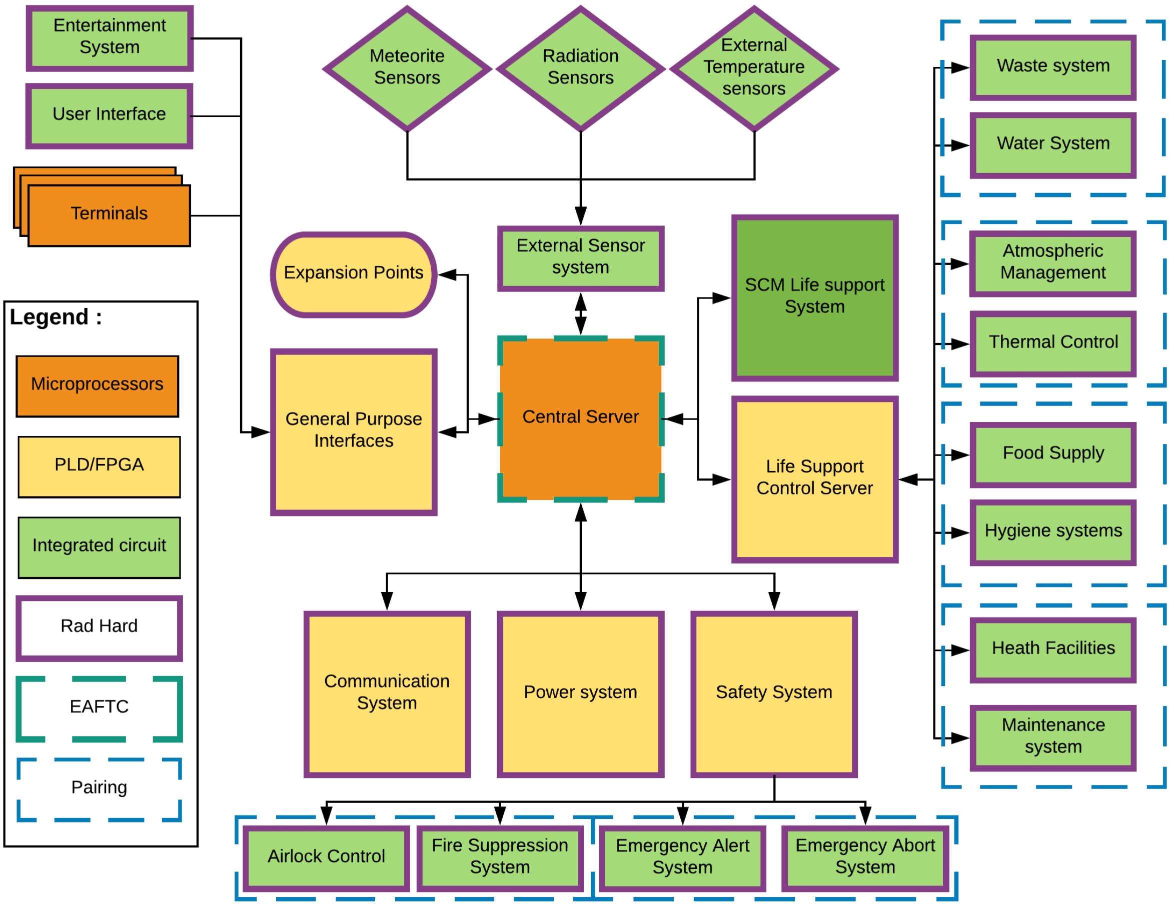


Figure 8.24: High level architecture of the computer system

In the above diagram there are three types of division, the type of system is indicated with the block colours, the radiation management mechanism indicated by the block borders and finally the redundancy approach marked by the bounding boxes (or absence thereof).

Three types of processing devices are used, namely, general microprocessors, Programmable Logic Devices (PLD) and Integrated Circuits. The microprocessors have the advantage that they can run any type of software that is designed for the operating system running on top of them. However, they consume more power than the other two device types. PLDs, of which Field Programmable Logic Arrays are a subset, are devices in between Microprocessors and Integrated Circuits, which can be conceptualised as re-programmable integrated circuits. They have the advantage of easy expansion and since they can be designed to perform one single task, they are quicker than microprocessors. Finally, Integrated Circuits are designed to perform one task, which they can perform very efficiently, however changing their functioning would require building a new circuit.

Two types of radiation management philosophies are implemented, using radiation hardened circuits and Environmentally Adaptive Fault Tolerant Computing (EAFTC). As mentioned before, radiation hardening requires a higher power consumption and the use of older microprocessors which are radiation hardened and thoroughly tested. NASA uses microprocessors designed in 2002 for their upcoming Orion capsules and the ISS uses processors from 1985 [113]. Given that Moore's law states that computation power doubles every six months, there is a huge gain to be made by using 2018 processors. This can be done using EAFTC, which is a computer architecture developed by a group lead by Honeywell. It allows the use of commercial off-the-shelf (COTS) equipment in radiation environments, by computing the same task on multiple cores and adapting the number of cores per process, depending on critical and current radiation levels [115]. The disadvantage is that the TRL is only 4.

Finally, the redundancy options are threefold: pair redundancy, full redundancy and EAFTC redundancy. In the case of pair redundancy, two systems are designed to take over the other system in case of failure. Full redundancy simply consists of a back up system, which is identical or designed using a different approach.

The core architecture consist of a Central Server housed in the SCM module, using the EAFTC system which allows the use of COTS microprocessors, making it the most powerful computation block on board. It holds a central overview of the functioning of the whole habitat system, it is the first point of contact between the systems and the astronauts. It detects faults that might require repairs/maintenance and interlinks inputs/outputs from other systems. On the next ring around the Central Server, the general subsystem control servers can be found which are PLD based and radiation hardened. They are located close to to where the relevant subsystems are housed and are fully redundant. They hold the overview of all subsystems they control. Finally, on the most outer ring radiation hardened integrated circuits that control the actual subsystems are present. They work with pair redundancy, marked by the blue discontinuous lines. If one of integrated circuits fails, the pair takes over the tasks. They are grouped according to similar functions and similar computational needs.

The whole system architecture allows for very high reliability and effective failure containment, while providing high computation power. It is a distributed architecture, and all components can work independently, albeit with a degraded performance. This design follows a three point of failure resistant philosophy. In order for any subsystem to fail, there has to be a chain of failures. First the integrated circuit has to fail, then its pair needs to fail, then the PLD circuit controlling them has to fail and finally the central server has to fail. If all of these failures happen, the SCM life support system also has to fail, in order for a catastrophic failure to happen. This chance is further reduced by the fact that there are several different design approaches per redundancy level. The system also provides relatively simple unit and integration testing possibilities, by starting the testing campaign from the outer edges inwards.

The disadvantages of this system are the relatively high development costs, the high software development costs and the low TRL of the EAFTC system. The development costs are high due to the fact that three types of computer systems are to be developed, however once one of the systems per type is developed, replicating it should not be very costly. The software costs are mainly related to the central server, which has to have software replicating the functions of all the other systems. As this is a critical system, these costs can be justified by the extra safety. More about this can be found in subsection 8.5.6.

8.5.5. Command, Data Handling and Computer System - Production

The production and installation of this system is relatively simple, as it is installed in the cylinders during ground construction. The systems have to be specifically developed for this application, with some COTS components used in the system. All interfaces between cylinders can be built into the coupling mechanisms, allowing for a plug and play construction.

8.5.6. Command, Data Handling and Computer System - Technical Budgets

For the budgets of this system, only the central server and the control servers are considered, as the integrated circuits are already budgeted in the relevant subsystems. In order to estimate the budgets, an estimation of the number of components is made. For the central server the same architecture is used that is proposed by the EAFTC designers is used [115], using four computing nodes and three controllers. In the case of the FPGA servers, sample architectures were analysed and it was determined that approximately 4 FPGA are necessary per server [116].

Mass, Volume and Power

The mass of the system is negligible, as most components are measured in grams, with the largest components weighing less than 2 kg. Most of the weight of the system are in the enclosures surrounding them. The estimation procedure was conducted based on analogous components from commercially available systems, and is shown in table 8.5. The volume of the components is negligible, so the sizing constraint is again the enclosures. Two server racks are budgeted for the central server, and individual enclosures are used for the FPGA [115, 117–120]. The components for the central server are the ones used in the original architecture paper [115]. Values are provided for the worst case scenario without any optimisation, however they do not account for contingencies, as they are accounted for in the technical budget section.

Table 8.5: Technical budget breakdown for the communications and data handling subsystem

System	Component	Quantity	Mass $[Kg]$	Volume $[m^3]$	Max Power $[W]$
Central server	- System controller	3	1	In enclosure	20
	- Adaptive processing	4	1.5	In enclosure	15
	- Computer	4	1.5	In enclosure	15
	- Packet switch	2	6	In enclosure	175
	- Radiation sensors	30	< 0.01	In enclosure	0.1
	- Enclosure	2	125	1.14	0
Control servers	- FPGA	20	0.1	In enclosure	5
	- Enclosure	20	10	< 0.0007	0
Total			475	2.42	553

Cost

The cost of the radiation hardened microprocessors can vary from €10 thousand to €200 thousand [113]. Since the main microprocessors are COTS components which cost in the range of €1000, none of the €200 thousand components are going to be used, generating a significant cost saving. The 20 FPGAs that can resist 1000 rads cost about €10 thousand each, hence their total cost is about €200 thousand. In the case of the central server, no cost estimate exist for this architecture. In order to remain conservative, it will be assumed that the cost of the whole system is as much as it would cost to use 4 radiation hardened cores, totalling € 800 thousand. As mentioned previously, rad-hard components are not going to be used, meaning the total cost will probably drop. The cost of all remaining components can be considered negligible. Applying a safety factor of 2, the total hardware cost would reach €2 million. Taking into account the scale of this project, this cost is completely negligible.

The second element to consider are the software costs, as there is no reliable way to estimate the size of the code base needed, the best alternative solution is to estimate the cost based on the ISS. This will most likely lead to an overestimation of the costs, as the ISS is much larger and contains elements from multiple Space Agencies that lead to significant duplication and interfacing. As previously mentioned the ISS runs 1.5 million lines of code [114]. In 1992 it was estimated that each line of flight code cost 654 USD [121]. Assuming that the price did not go up, other than to inflation, this would imply €562 in 2018. At this price level, the cost of the in flight software would be €847.5 million.

8.5.7. Command, Data Handling and Computer System - RAMS

The FMEA for the CDH system can be found in figure 8.25, with the scoring scheme as explained in subsection 8.1.5.

Function ID:	Function	Failure Mode	Potential Effect of failure	SEV	Potential causes of failure	OCC	Current Control	DET	RPN
FF_CDH1	Start-up system	Startup failure	No CDH	3	PSU failure	1	Obvious visual Inspection	1	3
			No central server	2	Damage to system controller	1	Network Inspection	1	2
			Fast system degradation	3	Underestimated radiation environment	1	Overdesign for radiation	2	6
FF_CDH2	Manage systems	Incomplete/non-management of systems	Degraded performance	2	Connection/Cabling failure	1	Network Inspection	1	2
			No central server	2	Full degradation of all computing cores	1	Network Inspection	1	2
			Loss of a system	4	Integrated Circuit failure	1	Network Inspection	1	4
FF_CDH3	Manage emergency systems	Incomplete/non-management of emergencies	Loss of life	4	Failure of detecting emergency situation	1	Astronaut Inspection	3	12
			Loss of life	4	Failure to enter emergency mode	1	Obvious	1	4
			No more nominal system operations	3	Failure to restore to nominal operations	1	Obvious visual Inspection	1	3

Figure 8.25: FMEA of command, data handling and computer system

8.5.8. Command, Data Handling and Computer System - Risk & Mitigation

Four risks appear to be critical based on the CDH FMEA. Firstly, an underestimation of the radiation environment can cause fast system degradation, leading to a premature mission end. Mitigation of this risk can be achieved by overdesigning the system against radiation, decreasing severity.

Integrated circuit failure can cause complete loss of a system, which can in turn lead to loss of life or destruction of the habitat. The only way to mitigate this risk is by having multiple layers of system redundancy, which is currently the case. Next, failures in detecting when something is an emergency situation can cause loss of life. Mitigation can be achieved by having a back-up system dedicated to detecting emergency situations and entering emergency mode.

8.5.9. Command, Data Handling and Computer System - Product Verification

The verification procedure to check requirement compliance can be see in table 8.6.

Table 8.6: CDH system verification procedure

Requirement	Verification procedure	Method
SYS-OP-CDH-1	- Check that the output of every system is present in the main server	Analysis
SYS-OP-CDH-2	- Run the full system and check the power consumption	Test
SYS-OP-CDH-3	- Analyse the reliability of every component, and compute the total system reliability	Analysis
SYS-OP-CDH-4	- Turn off the main system and check that the redundant system provides all functionality	Test
SYS-OP-CDH-5	- Damage or disconnect one section of the system at a time and check that the damage is reported	Test
SYS-OP-CDH-6	- Expose the system to a dose of 100 rad	Test

8.5.10. Command, Data Handling and Computer System - Product Validation

The validation of the CDH system is done during the validation campaign described in section 13.4. After verifying that the system is performing as expected on the ground, the validation of the system is conducted during the one month trial run on the Moon. The difficulty of the validation of this system is the fact that it is the same system that transmits all the data back to Earth, hence the verification of its functioning on Earth must be done extensively.

8.6. Electrical Power System

As opposed to the other subsystems, the electrical power system (EPS) still has several options to be considered. In this section, these options will be explained and evaluated after which one is chosen and further elaborated upon. The requirements for the EPS can be found in appendix B.

8.6.1. Electrical Power System - Design Options

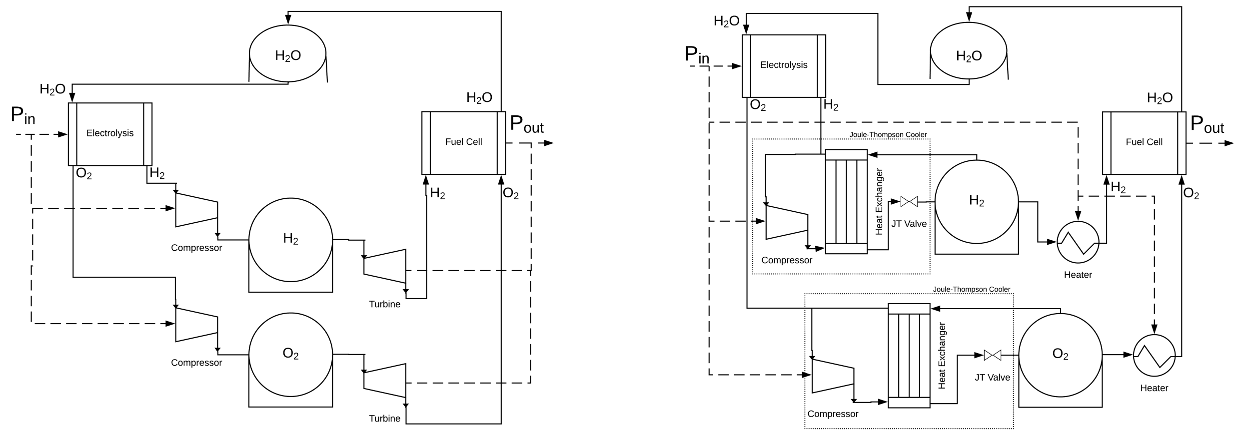
Globally two options are available for the EPS: solar or nuclear power. For nuclear, only a fission reactor is deemed feasible as outlined in Midterm Report [3]. For solar panels a energy storage system is necessary. Two main options exist: batteries or fuel cells. In this section all these options will be briefly discussed.

Batteries

In the battery category four different options have been investigated. The key parameter being considered is the specific energy, or the amount of energy that can be stored per unit mass. Option one is a classic flight proven Li-ion battery with a specific energy of 280 Wh/kg. Next is a more futuristic solid state battery. These so called Li-Metal batteries are currently being produced [122] and have a promising specific energy of 500 Wh/kg. The last technology is a lithium-sulfur battery. The aim of [123] is to get these batteries with a specific energy of 500 Wh/kg ready for space applications. However the potential of Li-S technology can reach a massive specific energy of 600 Wh/kg [124] and thus will also be considered. The subsystem masses were simply determined using these specific energies and the amount of energy that is needed to store. This was then added to the mass of the solar panels.

Fuel Cells

The several options for the fuel cells naturally came up when encountering complications with the previous option. The general approach, however, is the same for all three. First a reversible fuel cell is used, which converts energy and inert chemicals into reactive chemicals. These will then be stored and when power is needed they will run through a fuel cell to generate power and the same initial inert chemical. The first two options contain the largest amount of energy per unit mass: a hydrogen-oxygen fuel cell and electrolysis cell system. One issue that arises is that both hydrogen and oxygen are gases and therefore require a large volume and heavy tank to store them. For option one, shown in figure 8.26a, the tank mass has been optimised, as described later in this section. The complications, due to a low gas density, could be solved by using cryogenic storage tanks. This system is shown in figure 8.26b. The dotted arrows portray power flows, whereas the solid ones represent chemical flows. A cryo-cooler would be used to cool the propellant coming out of the electrolysis cell and is heated again when it exits. The Joule-Thompson cooler is proposed, due to its relative simplicity. The gas is compressed using a turbine and then expanded through a Joule-Thompson valve. A certain fraction of the gas will be liquefied and remains in the storage tank. The gaseous counterpart will be vented back into the cyclus through a heat exchanger with the compressed gas. A complication with this method is that the compressor induces an added required power of 100 kW to be able to run the cooler unit.



(a) Block diagram of the gasesously stored hydrogen fuel cell system

(b) Block diagram of the cryogenically stored hydrogen fuel cell system

Figure 8.26: Two proposed hydrogen fuel cell systems

A third option would be to use the power to ammonia principle which is used for grid balancing and power storage purposes [125]. In this system both the water and ammonia are easy to store in liquid form allowing for small sized tanks. However, complications arise due to the fact that ammonia does not pack quite the amount of energy per unit mass as hydrogen, 18.6 MJ/kg versus 142 MJ/kg respectively. Additionally, the system would still make use of gaseous chemicals. This is partially solved by running a hydrogen fuel cell during power storage mode to power a compressor. However, in addition to the oxygen, the system would need to store nitrogen as well. All these components make a very complex system, shown in figure 8.27, that is potentially very prone to error and can therefore only be implemented with redundancy at every component.

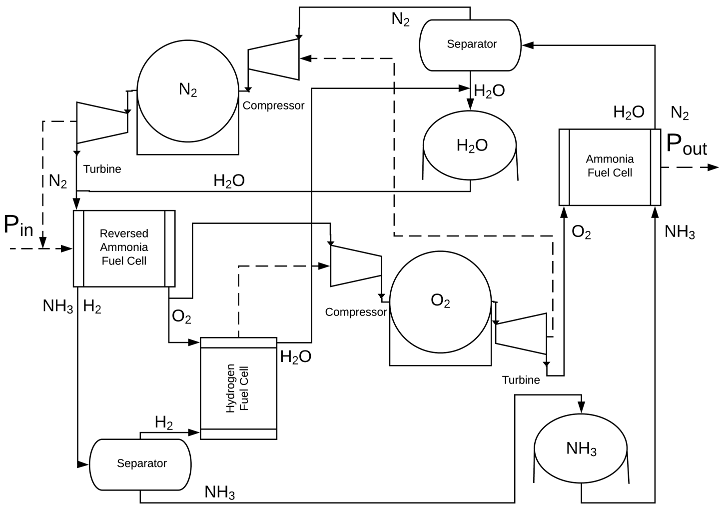


Figure 8.27: Block diagram of the ammonia fuel cell system with liquid storage

Estimating the masses for these systems is lot more complicated than for batteries. It contains masses for the solar panels, chemicals, tanks, fuel cells, compressors and turbines.

The solar panel mass is dependent on the round trip efficiency of the system as it determines the amount of power that needs to be generated. This is 60% for hydrogen and 39% for ammonia according to Wang *et al.* [126] and Davis *et al.* [127]. The chemical mass is calculated similarly to that of batteries, using the specific energies of the fuels. The oxidiser mass is determined by taking the stoichiometric molar ratios between the fuel and oxidiser.

The tank masses depend mainly on four parameters: material density and yield strength and the gas' density and pressure. To optimise the first two titanium is chosen. It is a more expensive material, but offers outstanding strength to weight properties. Since hydrogen and oxygen do not behave as ideal gasses when put under high pressure a more intricate model is used to determine the pressure that allows for the lightest possible tank. This is due to the balance of lower thickness for lower pressures but higher radius for lower densities and pressures. This model is retrieved from Mench [128] and uses the Van der Waal's equation of state, shown in equation 8.2.

$$P = \frac{\rho R T}{1 - b\rho} - a\rho^2, \qquad with \qquad a = \frac{27}{64}\frac{R^2 T_c^2}{P_c} \qquad and \qquad b = 0.125\frac{RT_c}{P_c} \tag{8.2}$$

This equation can be solved for ρ in cubic root form and was done by 1728 Software Systems [129] method. Using the most shape efficient tank type, a sphere, the required tank thickness is calculated using the first formula in equation 8.3 and the tank mass using the second.

$$t_{tank} = \frac{P R_{tank}}{2\sigma_{Ti}}, \qquad m_{tank} = \frac{4}{3}\pi \cdot \left[\left(R_{tank} + t_{tank}\right)^3 - R_{tank}^3 \right] \cdot \rho_{Ti} \tag{8.3}$$

Now the pressure could be adjusted and by trial and error the tank mass could be minimised, taking into account that the diameter should not exceed 4.6 m, a requirement that was the limiting factor only for the very sparse hydrogen gas. The tank mass for the water and ammonia storage is very low as they are dense and can be stored at low pressures.

For estimating the mass of the fuel cells, turbines and compressors, Tornabene *et al.* [130] was used. It provides curves estimating these fuel cell components on a same scale system. The reversible fuel cells have been estimated to be 118 kg, turbines are a mere 6 kg and the compressors 10 kg each. Given the result of the trade-off in subsection 8.6.2, no sensitivity analysis was warranted on the model estimating the tank masses for the different fuel cell designs.

Nuclear

The last option would be the use of a small nuclear reactor. This allows for a closer cooperation with NASA, since it is currently developing and testing a small nuclear fission reactor KRUSTY (Kilowatt Reactor Utilising

Stirling Technology) under a project called Kilopower. Currently one unit produces 1 kW, but this can be upscaled to around 10 kW according to [131]. Hence, a total of two units would be required. Globally the system consists of three big parts displayed in figure 8.28: a nuclear fission core providing heat to the system; 4 Stirling generators to transform the thermal power into electrical power and a large radiator to provide a cold heat sink on the other side of the generator. This heatsink has two functions: radiating all the excess heat away from the system which originates from the power converters that do not run at 100% efficiency and to provide low cold temperature to increase the efficiency of the generators following the fundamental thermodynamic formula in equation 8.4.

$$\eta_{th,stirling} = \frac{T_H - T_L}{T_H} = 1 - \frac{T_L}{T_H} \tag{8.4}$$

Of course, one of the first questions arising when hearing the word nuclear is the systems safety. Therefore, the next section will mainly focus on explaining how the nuclear system works and why it is appropriate to consider it to be sufficiently safe for the mission.

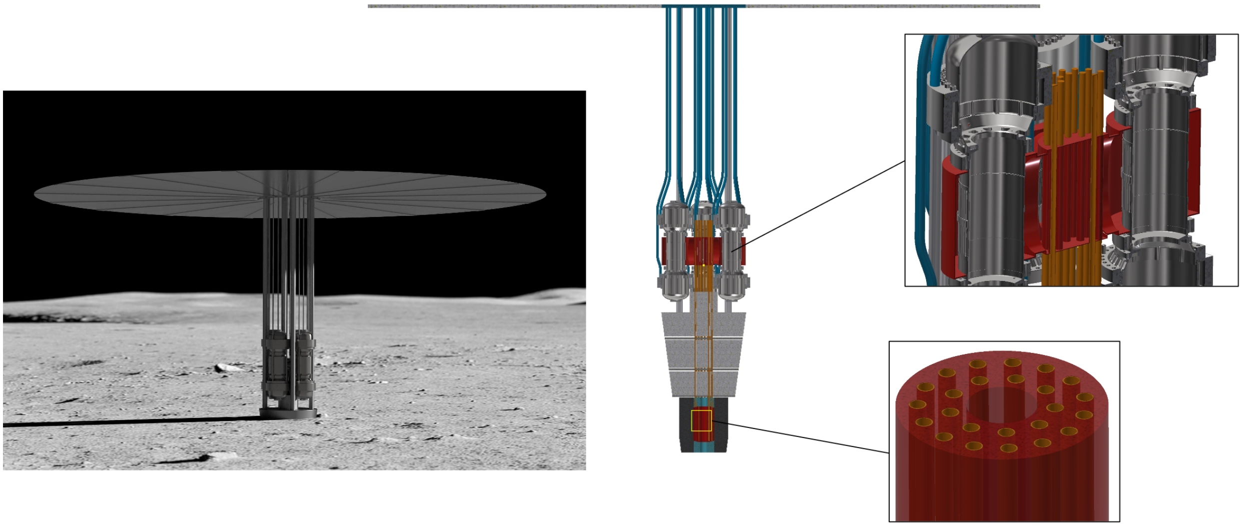
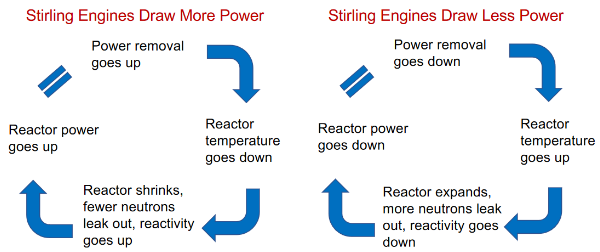


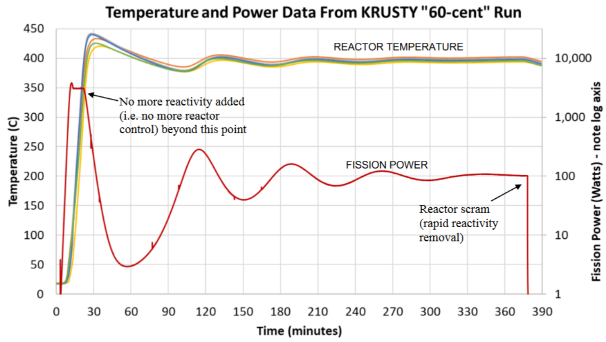
Figure 8.28: External and internal view of the kilopower reactor [132]

Safety of Nuclear Fission

The reactor is fundamentally different from the large thermal nuclear reactors on Earth. It is a so-called fast neutron reactor. This means it does not require a moderator to slow down the faster neutrons into so-called thermal neutrons, required for a thermal nuclear reactor. This omits the required enormous basin of water, which is of course a huge benefit considering transportation to the Moon. The heat is transferred from the nuclear core using sodium heat pipes and therefore it does not require high pressure steam to operate. This means it is much less prone to explosive behaviour, which is one of the reasons constant monitoring of Earth-bound reactors is necessary. Contrarily, KRUSTY has a cycle, that corrects for any disturbances it might undergo during operation. After the reactor is made operational it requires no to minimal service and astronaut interference for it to function within its given constraints. This reliability is more elaborated upon in equation 12.1.1. The nature of this passive stability is explained in figure 8.29a. This cycle occurs and dampens just like a classical underdamped stable system. This response is plotted in figure 8.29b. Please note that this period is rather long due to the test's low power output of 100 W. This effect is comparable to a pendulum that also takes longer to swing back and forth in lower gravity [132].



(a) KRUSTY's stable system cycle [132]



(b) Reactor power and temperature response in uncontrolled conditions [132]

Figure 8.29: KRUSTY's inherently safe system cycle

Additionally, the nuclear material, U^{235}, is far less radioactive than the plutonium used in RTG's the last decades, when the core is not under a sustained nuclear chain reaction. During launch, and any other occasions where the reactor might need to be shut down, this is ensured by a boron carbide control rod inserted in the core. This keeps it from maintaining a stable nuclear chain reaction or unstable increased reactivity (more neutrons released than required to maintain the current amount of reactions). In case the launch vehicle explodes on the pad, with the inactive reactor on top of it, a person standing 1 kilometer away from the pad will receive a dosage around 1 millirem. This is the same dose a whole-mouth dental X-ray gives and is vastly lower than what a passenger gets during an airplane flight according to McClure, the project lead for Kilopower at Los Alamos National Laboratory, in Business Insider [133].

Also during operation, the radiation emission will be limited, since a full third of the systems mass originates from radiation shielding, limiting the radiation exposure to less than 3 millirem per hour (mR/hr) within 500 m [134]. This comes down to 0.26 Sv per year, for continuously standing next to it. However, the reactor will also be buried beneath the regolith and a small wall in the direction of the habitat can also be added. On top of that, the astronauts will be mostly inside the habitat, already designed to shield them from the harmful radiative space environment. During EVA's they will not be around the habitat and reactors continuously, when they're exploring more remote regions around the habitat.

8.6.2. Electrical Power System - Trade-off

Now that all the options have been discussed, a trade-off regarding the power supply of the habitat can be performed. The results are shown in table 8.7.

Table 8.7: Visual representation of the trade-off for the electrical power subsystem

Aspect	**Power source**							
	Solar power							**Nuclear fission**
Storage	Li-				Reversible fuel cell			N/A
	Ion	Metal	S Current	S Future	H_2 Compressed	H_2 Cryo-cooled	NH_3 Compressed	
Mass $[ton]$	15.8	9.0	11.2	7.6	5.6	3.5	8.9	3.5
Safety	Safe with protection	Inherently safe			Potentially hazardous to use			Safe by design
					Large tanks and turbines	High temp. fluxes, fast rotation	Less stable chemical process	
TRL	9	4	5	3	6	3	4	5
Sustai-nability	Considerable amount of Li				Closed loop system with non-toxic materials			Disposal of nuclear material
Oper-ations	Can't be fully emptied for long duration	No moving parts or maintenance needed, can sit empty without degradation			High operating temp.	Requires additional 100 kW for cooling	Faster membrane binder degrad.	Stable and steady state
Insta-llation	One part				Assembly required			Digging

Additionally to the information in the table, solar panels will require a lot of maintenance. Regolith could cover the panels, cling to joints and micrometeoroids could puncture the panels. Also due to the presence of gravity the mechanism of the panel should be redesigned to take into the forces due to its own mass, which is unnecessary for satellites in orbit. The radiator of the nuclear system in comparison is much smaller, and won't cause failure when some damage is sustained as explained in figure 8.6.4. Also the fuel cell systems are prone to error due to the criticality of all of its components and the large size of the total system.

In conclusion, by means of voting, the whole group decided to go for a nuclear power source. The main arguments were its simplicity versus a rather complex and risky fuel cell system, its safety and the higher TRL. The high performance Li-S batteries are currently just formulated as a potential limit, which according to experts is reachable for this technology somewhere in the future. However, no concrete attempts have been made yet, since the industry is still working on developing the current 400 Wh/kg batteries. The team has attempted to a great extend to avoid using a nuclear power source due to the social complications it brings. Even though its technology is well established, public outcry and fear for catastrophic accidents hinder the use of such systems. The main hinder is the combination of the words space and nuclear. Separately they are generally accepted. Nuclear reactors still widely exist around the world and at KRUSTY's scale and distance from home, they are massively being implemented in nuclear powered submarines and aircraft carriers. Now the teams job is to ensure the public of a safe system with these two words combined and showing them the technological advances it brings to human advances in spaceflight.

8.6.3. Electrical Power System - Functional Flow Structure

Figure 8.30 shows the FFD of the nuclear fission plant.

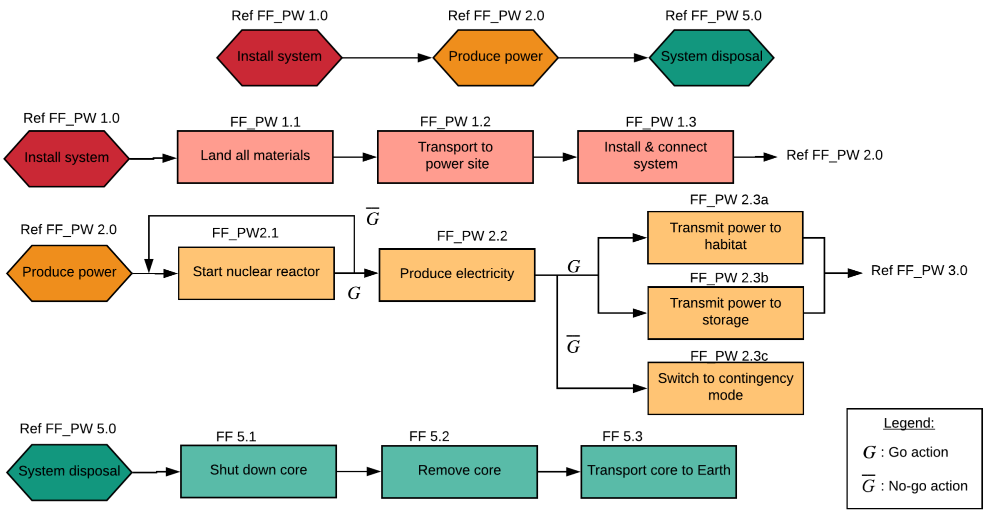


Figure 8.30: Functional flow of the power supply subsystem

8.6.4. Electrical Power System - Layout

The reactors will emit radiation, and it is therefore required they have a certain distance from the habitat to protect the astronauts during EVA. When the astronauts are inside the habitat, they won't be affected critically. According to Rucker *et al.* [134], one nuclear plants emits 0.263 $Sv/year$ at 500 m distance each. Radiation in strength is related to distance according to the inverse square law 8.5.

$$intensity \propto \frac{1}{distance^2} \tag{8.5}$$

For a distance of 100 m this results in a total of 6.57 Sv per reactor, and 13.14 $Sv/year$ in total. This is the same as 4.2 $\cdot 10^{-7}$ Sv/s, which is within the human exposure limits. In addition, the core is going to be dug in and a small regolith wall will be built between the reactor and the habitat. The astronaut's space suits will also provide some additional protection. The reactors must have a certain distance with regard to each other in case one has to be shut off for maintenance or repair. The same distance of 100m was selected in order to minimise the distance for cabling, while being safe. The two reactors and the habitat form an equilateral triangle if seen from above. With simple optimisation of trigonometry the minimal desired length of the cabling was calculated as is shown in figure 8.31.

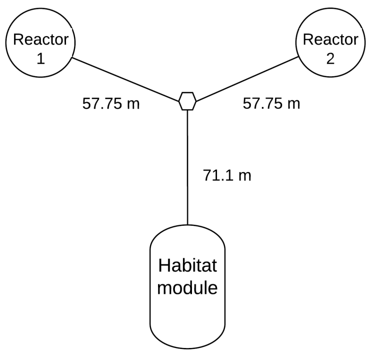


Figure 8.31: Schematic view configuration of cabling between reactors and habitat

A total of 186.6 m of cabling is required. Cable loss can be minimised by using a current as low as possible and a cross sectional area of the cable that is as large as possible. Since the output Voltages and Amperes are not known for KRUSTY and AC is chosen as output for minimising transportation loss, either the use of transformers and smaller cross-sectional area cabling or the use of no transformer and cabling with a larger cross-sectional area has to be chosen. The material of the cables is most likely a copper or a copper-silver alloy. Keep in mind that a power station near the habitat is required in both situations to provide a voltage of $220 - 230V$ to the habitat. For certain systems DC is required and therefore another AC-DC transformer is required in both situations as well. Total power loss of the system is assumed to be 5%.

Meteoroid Protection

The power source is very valuable for the missions continuation and was therefore evaluated separately for possible meteoroid impacts. To analyse whether this is required, the titanium radiators are assumed to resemble titanium monolithic shields. Using Christiansen *et al.* [43], the formula to calculate the minimum diameter (d_p) of a spherical particle that produces penetration failure damage to the impacted titanium sheet with thickness (t_{Ti}) is equation 8.6.

$$d_p = t_{ti} K^{-1} \cdot 5.24^{-1} \cdot BHN^{0.25} \left(\frac{\rho_p}{\rho_{Ti}}\right)^{-0.5} \left(\frac{v \cdot cos\theta}{C_t}\right)^{-2/3} \tag{8.6}$$

Where:

- ρ_p = density projectile $[g/cm^3]$;
- ρ_{Ti} = density titanium alloy $[g/cm^3]$;
- K = Damage parameter for titanium alloy;
- C_t = Speed of sound in target $[km/s]$;
- v = projectile velocity $[km/s]$;
- BHN = Brinell Hardness $[-]$;
- θ = impact angle from target normal $[°]$.

The input values have been stated in table 7.1. Hence, using equation 8.6 and using v = 70 km/s, a critical projectile diameter of 0.037 cm is obtained which corresponds to a meteoroid particle with mass $2.725 \cdot 10^{-5} g$. From equation 5.6, a flux of 0.0570 $[Impacts/m^2 - year]$ was obtained which results in a total of 23 penetrations for both nuclear fission radiators in 10 years, using a total area of 20 m^2 per fission plant, hence 40 m^2 in total for the use of two. Projectile speeds of 70 km/s are rare, hence the value in reality shall be lower. For the average of v = 23.9 km/s, the total number of penetrations is 7 for the two radiators. Because the size of the impacting particles are small, and are considered to not critically influence the function of the radiators, it was decided no extra protection is required for the nuclear plants. The actual nuclear core is dug in and its area is smaller and therefore considered protected sufficiently. The Stirling engines are exposed, but again, their exposed areas are very small and they are redundantly built in. EVA visual inspection shall be performed to monitor their health status. The radiators could be repaired in case of severe damage. If this is to be done, the reactor has to be shut of before on of the astronauts moves there. The second fission plant shall have a distance of 100 m to the other, in order to be able to remain operational whilst the other is being inspected or repaired.

This model does not include cumulative damage caused by reoccurring impacts of smaller particles. For future design, it is advised to include cumulative damage model.

8.6.5. Electrical Power System - Production

Manufacturing and construction of the EPS will be outsourced to NASA. A close cooperation will result in a beneficial mission for all parties. NASA will have a testbed for their system to reach TRL 9, before implementing it on a more distant mission to Mars, whereas the habitat is provided with a constant, reliable and safe means of power generation. Installation will be done during the initial phases of the mission and will include a minor digging operation. Burying the reactor will increase its safety even more and serve as a foundations for the "umbrella" design.

8.6.6. Electrical Power System - Technical Budgets

Mass

The mass of the power system encompasses the two plants and the transformers and cabling. The estimated mass of the two reactors is 3508 kg [134]. An additional contingency of 15 % is added because the system hasn't been fully developed yet, resulting in a total of 4034.2 kg. Assuming an additional 1200 kg for all power distribution machinery, total mass comes down to a total of 4708 kg.

Volume

The total volume again encompasses the two plants and the transformers and cabling. The total volume of the folded plants equals 20 m^3. Because the cabling and transformers are assumed to have a volume of 0.06 m^3: 186.6 m of cable with diameter of 2 cm and converters with a total volume of 1 m^3. Total estimated volume: 21.06 m^3

Cost

The estimated cost of the current running prototype is around $20 million [133], transforming with today's currency exchange rate, this equals €17.2 million. This concerns a system with an electrically heated depleted uranium core, to validate performance of the power converters. Because the complete system itself is currently still being developed a contingency of 20% was added. Total estimated cost: €21 million.

8.6.7. Electrical Power System - RAMS

The meaning of each of the criteria is explained in subsection 8.1.5.

Function ID:	Function	Failure Mode	Potential Effect of failure	SEV	Potential causes of failure	OCC	Current Control	DET	RPN
FF_PW1	Install System	Installation Failure	Only emergency backup power available	3	Damage to materials due to travel	1	Video and relayed inspection	2	6
					Digging in does not work	2	Testing on Earth	3	18
FF_PW2	Produce Power	No power produced	No power available	4	Start-up failure	1	Network inspection	1	4
					Transmission of power failure	1	Manual inspection	2	8
FF_PW3	System Disposal	Incomplete or no disposal	Leave reactor on moon	2	No shutdown	1	Network inspection	1	2
			New disposal location required	1	Core more radioactive than expected	1	Manual inspection	2	2

Figure 8.32: FMEA of the electrical power system

8.6.8. Electrical Power System - Risk & Mitigation

The two potential risks that need mitigation are: digging in does not work, with a RPN of 18, and transmission of power failure, with a RPN of 8. The mechanism for digging in the reactors could be demonstrated and tested on Earth in similar conditions, however, the composition of the Lunar surface structure at the digging site could differ from the expectations. It is therefore advised to have a mission ahead of this mission for to fully test the operationality of the rovers and sampling the sites by drilling. The digging in of the reactor should be simulated with a comparable structure. This is a costly solution but considered necessary to ensure a safe setup and functionality of the mission. Transmission of power failure is either failure of the converters or the cables. In both cases, high TRL, space proven or tested units must be used. This adds extra weight to the system but negligible to the total and necessary for continuous power supply.

8.6.9. Electrical Power System - Product Validation

All information used regarding the nuclear fission plant KRUSTY was gathered from NASA. This includes tests, sizing and practical information. It is therefore assumed that the model is validated to a certain extend. However, as mentioned, KRUSTY is currently still being developed and tested. The actual parameters might chance once its development has been finalised.

9
Operations Design

Having designed all subsystems that are included in the interior of the habitat, there are a few more elements that need to be investigated before the integration and the logistics of the whole system can commence. This chapter deals with a brief discussion of the technology needed for the Lunar landing in section 9.1, the Lunar rovers for any sort of operation in section 9.2, and the return vehicle for the astronauts in section 9.3.

9.1. Precision Landing Technology

The first design to be considered with regards to the operations of the habitat is the technology that will be used to successfully land as close to the desired location as possible. Hence, two prevision landing possibilities are investigated: the advanced technology of terrain relative navigation or the use of beacons.

9.1.1. Terrain Relative Navigation

Terrain Relative Navigation (TRN) is an emerging landing technique that shows promising results. It is a system with TRL of 6 which is set to be used on the Mars 2020 mission to demonstrate its capabilities. It makes use of a Lander Vision System (LVS), a smart sensor which combines and matches images it makes during descent with a pre-loaded high-res landing site image. Together with precision inertial measurements it enables to estimate velocity profiles, attitude position and relative map position with high precision. The system works at any altitude but has to be specifically calibrated for different mission profiles, i.e. for the Mars 2020 mission it only starts working when the lander reaches an altitude of 3 km. This mission envelope got tested extensively: at high altitudes a coarse match between images is made, when lower altitudes are reached a finer match between landmarks is made. During these tests, several simulations estimated a landing accuracy of 40 m which would suit the needs of the Moon landing system [135]. This system will be accompanied by the Navigation Doppler Lidar (NDL), which is specifically designed by NASA Langley for precision landing on other planets. It is a system that comprises of three laser beams which provide ultra-high precision velocity, direction and altitude measurements without having high weight or power penalties. The system weighs around 13.5 kg and needs 90 W of power to operate [136]. The combined system is currently being tested and should be ready before 2020.

9.1.2. Beacons

When talking about beacons, an important concept to understand is that of trilateration, which is based on the trigonometric measurements of three points in a space domain to calculate absolute or relative locations. If enough sides and angles are known, the remaining distances can be computed. This is the simple, geometrical idea behind beacons. A minimum of three points, hence three beacons, are needed to successfully identify the exact location where the system is desired to land. However, the information could be affected by reading errors. In order to negate this error, a fourth beacon is preferred for minimising uncertainty [137]. The main pro of using this technology for precision landing is that it can be extensively tested on Earth to reach the perfect, desired accuracy. As with TRN and NDL, state of the art beacons have a TRL of 6 as the technology has been tested in a relevant environment. However, a remarkable con is that they have to be placed at significant distance from one another to be useful. Finally, deploying the beacons would require more mass and system systems compared to using TRN.

9.1.3. The Choice

Based on the discussion provided above, the choice is made to follow NASA's vision and use a combination of LSV and NDL. The total mass estimate for this system would not exceed 20 kg and would facilitate landing procedures, as no extra system has to be developed for deployment.

9.2. Lunar Rovers

Ideally, the second most important thing that shall land on the Moon after the beacons are the Lunar rovers. For the scope of this mission three vehicles will be discussed: one for astronauts' transportation purposes, one for the installation of the habitat, and one that takes care of all regolith-related activities.

One problem that arises for all the rovers is the clinging and abrasive nature of the regolith, which leads to the risk of compromising the performance of the vehicle. To resolve this issue, all the rovers will be equipped with Electrodynamic Dust Shields (EDS), a technology that has been successfully tested already for rigid systems [138]. The rovers will be equipped with thin conductive wires, when an AC-current is passed through the wires they will effectively dust off the vehicle. The power required for operating the EDS equals 0.02 W/cm^2 as determined by Calle *et al.* [139].

9.2.1. Apollo's Lunar Roving Vehicle

Despite several designs of Lunar rovers having been developed, the most reliable reference for this assignment is without any doubt Apollo's Lunar Roving Vehicle (LRV). This 3.0 x 2.3 x 1.1 m electric-powered vehicle has a ground clearance of 36 cm, and has successfully performed in Apollo 15, 16, and 17 missions. Showed in figure 9.1, the LRV weighs a total of 210 kg and can carry a payload of 490 kg, which equals 116 kgf when converted into Lunar pound-force. The frame has been manufactured with 2219 aluminium alloy and it folds semi-automatically inside the Apollo's lander. However, assistance from the astronauts is required for deployment, activation and operation. The LRV includes a Velcro-fastened seat belt for safety purposes.

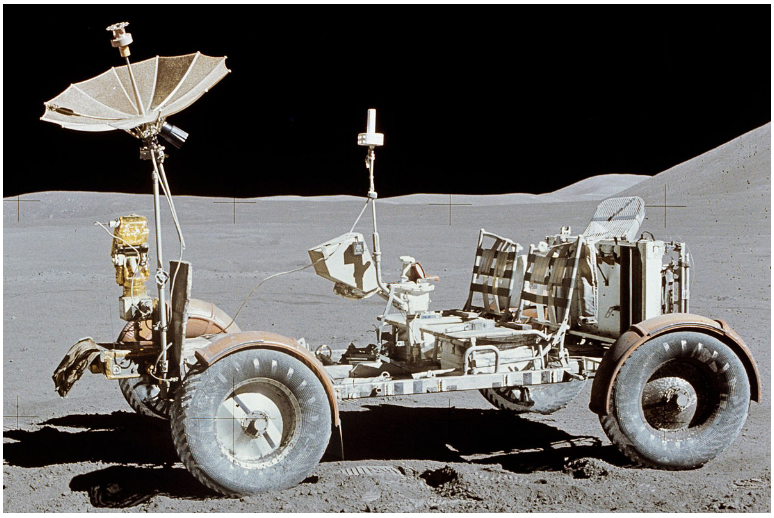


Figure 9.1: Apollo's Lunar Roving Vehicle

The vehicles used in the Apollo missions have been somehow different in terms of performance. These inequalities can be seen in table 9.1. The vehicle used in Apollo 14 was not an actual LRV, but a rather simple cart with wheels, pulled around by the astronauts to contain the needed equipment and collect samples [140].

Table 9.1: Comparison between Apollo's Lunar Roving Vehicles in terms of performance Morea [140]

Characteristic	Unit	Apollo mission 14	15	16	17
Driving time	$hr:min$	*N/A*	03:02	03:26	04:26
Surface distance travelled	km	5.31 (estimated)	27.84	26.71	35.88
EVA duration	$hr:min$	09:23	18:33	21:00	21:30
Average speed	km/hr	*N/A*	9.17	7.72	8.04
Max range from Lunar module	km	*Unknown*	4.98	4.50	7.56 (EVA #2)
Longest EVA traverse	km	2.41 (estimated)	12.47	11.58	20.11 (EVA #2)
Rock samples returned	kg	42.63	77.11	96.61	112.94

9.2.2. Astronauts Rover Vehicle

The first vehicle to be discussed is conceptually very similar to the one of the Apollo's missions: a rover that can transport astronauts on the Moon. From this point onward, this vehicle will be referred to as

Astronauts Rover Vehicle (ARV). In order to develop a preliminary design of the ARV, a brief list of the driving requirements has been generated [140] and put into the list of requirements appendix B.
Since it is beyond the scope of this project to fully design a roving vehicle, the dimensions, weight, and R&D cost budgets follow directly from Apollo's LRV [141]. It should be noted that estimating R&D costs is extremely difficult. To add contingency it was assumed that one dollar spent on R&D is the same as one euro spent on R&D (adding percentage wise contingency with respect to the exchange rate of both currencies). The rover will consists of three parts: the front of the vehicle will contain a navigation unit, high-gain antenna for all data processing, low-gain antenna for communications with the habitat, and a power unit. The two astronauts' seats, the controller unit and the display monitor will be in the middle, whereas the storage for the payload is located in the back section. All these components will be upgraded to meet today's technological standards. As stated, conductive wires will be placed around the vehicle to create the EDS. An overview of the specification parameters are given in table 9.2 and will be used for organising the system's logistics and operations of the habitat. At least two ARVs will be needed for the first year of the mission for redundancy purposes.

Table 9.2: Specification parameters of the Astronauts Rover Vehicle

Parameter	**Unit**	**Value**
Empty mass	kg	210
Max payload mass	kg	490
Max total mass	kg	700
Frame length	m	3.0
Frame width	m	2.3
Frame height	m	1.1
High-gain antenna diameter	m	0.55
Volume	m^3	7.59
Total cost	$€M$	250

9.2.3. Installation Rovers

The second vehicle to be discussed is the vehicle whose main responsibility is to install the habitat. A big challenge in doing so is the transportation from the landing to the building site. First, the landed modules have to be tilted horizontally, after which the system transports them to the designated habitat location site and places them on the Lunar surface. This process requires a specific rover, specifically designed for such purpose.

The Options

The main dilemma that arises when designing an installation rover is whether this should be one large single system, hereby defined as the the Mammoth, or multiple smaller systems, called the Ants. The ants in their turn could be automated or individually controlled from Earth. As the technology behind automated capabilities is already advanced [142, 143]. It is assumed that automated control of the Ants is more readily available and reliant by the time this mission launches and is therefore the desired option.

The pros and cons of a having a single system versus multiple ones can be found in table 9.3. The main pro of the Mammoth is that is only consist of one single operating system, effectively making the overall system less complex. However, its cons outweigh this pro: A heavy, single purpose, short operational life system is undesired.

The Ants can be designed in such a way that, after installation of the habitat, they can be reused for other purposes, like exploration, whereas the Mammoth is too big to expand with additional options and functions. The Ant's dimensions are limited, making its transport inside the rockets less complicated. The cons mainly relate to the control system programming challenge. Several Ants precisely working together in a different environment than Earth is very challenging. However, looking at state of the art technology, this is deemed feasible [142]. Therefore, the Ant design was chosen. It should be noted that this kind of rover has a TRL of 1 and is therefore very conceptual and shall only be qualitatively elaborated upon. Working out all the details would not be realistic taking into account the tight time schedule of the project.

Table 9.3: Pros and cons of the installation rovers

System	Pros	Cons
Mammoth	- Easier to perform the task at hand - High structural rigidity - Less complex mechanism - Extensively applied in other fields	- No immediate failure redundancy - High weight and large dimensions - Single purpose - More difficult to traverse rough terrain - Safe life - Short operational need
Ant	- High failure redundancy - Relatively low weight - Logistically friendly - Multi-functional platform - Small individual dimensions - Fail safe	- More prone to individual failure - Complex technology - More difficult to co-operate

The Ant

A preliminary CAD drawing of the Ant vehicle is presented in figure 9.2: It is 3 m wide and 2.1 m long, it consists of six pairs of omni-directional wheels, each equipped with its own electrical motor for full mobility. The can also adjust their height by a vertical piston and a pair of struts to angle them up and down. Furthermore, a battery is located in its base plate, which enhances stability due to the lower centre of gravity. When the batteries are depleted, they can be charged through a charging station connected to the power plant. Another advantage of this system is that the rovers can serve as power grid balancers. The charging system could be compared to that of an autonomous lawnmower, where the Ants will automatically drive themselves to. The most interesting fact about the Ant is that it can co-operate with other systems. In case this is not needed, for example once the transportation of all modules is complete, they can also operate individually. The final feature worth mentioning of the Ant is the connection points on top of the baseplate. Due to this configuration, the Ant could be equipped with different kinds of equipment for several other purposes. How the Ant will be operated will be elaborated upon in section 11.3.

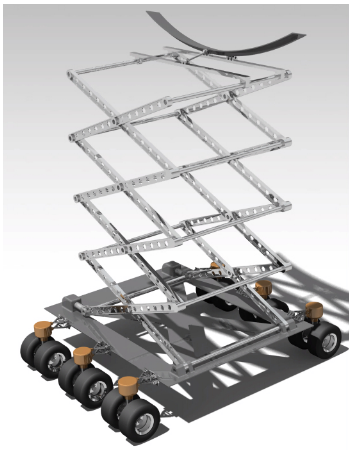
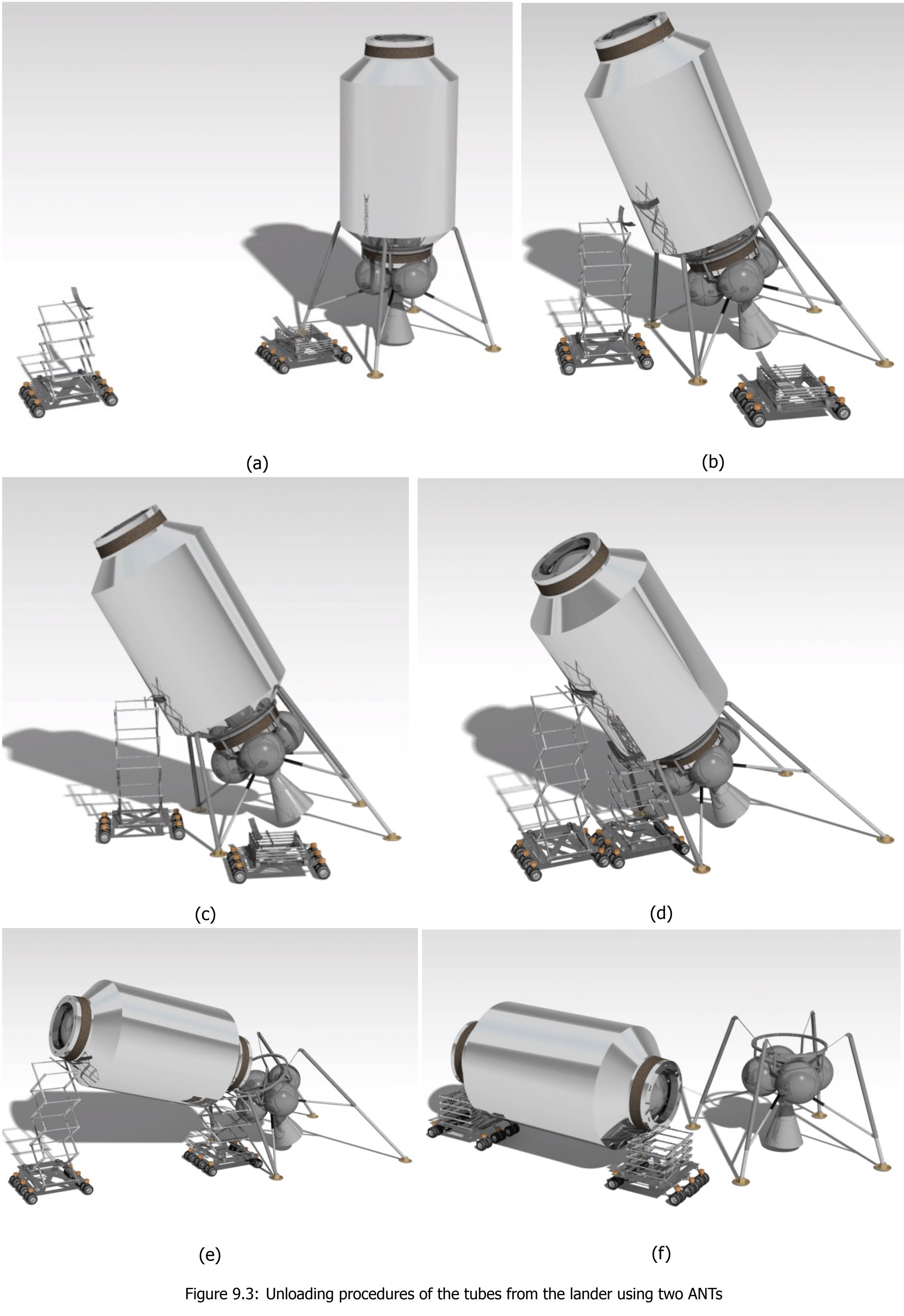


Figure 9.2: CAD render of the Ant system

The Ant used in installation, is equipped with a scissor lift to carry the modules, with a total of sixteen struts. At the base of the struts, an electric actuator is placed with a stroke-length of 40 cm, enabling the lift to deflect up to a maximum of 5.40 m. By performing simple force analysis taking the Moon's gravity into account, it was calculated that the two actuators should have a combined force output of 35000 N: with current actuator technology, this is viable as only one electric actuator can already exert this force [144]. Chapter 11 develops a more elaborate explanation on how the Ant rovers perform their designated tasks. Finally, based on the Mars curiosity rover, the R&D expenses for the Ant are expected to be around €700 million. It should be noted that the curiosity's mission budget was $2.5 billion, however the breakdown of this budget is found to be classified. Common believe is that around 30% went to R&D of the rover itself [145]. As the Ant has similar dimensions to the Mars Curiosity rover as well, the mass of the Ants is estimated to have the same mass of around 900 kg [146].



(a) (b)

(c) (d)

(e) (f)

Figure 9.3: Unloading procedures of the tubes from the lander using two ANTs

9.2.4. Regolith Rover

For smoother transfer of the heavy modules to the designated habitat location, a road has to be paved to facilitate mobility on the rough terrain of the Moon. Additionally, it is highly desired to minimise dust cloud formation due to the rocket's exhaust during landing. Furthermore, the layers on top of the modules will have to be solidified to ensure rigidity of the regolith and appropriate and additional protective layer against meteoroids. This could be achieved by using the sintering technology as explained by Taylor and Meek [25]. For this specific purpose, a preliminary design of the Regolith Rover (RR) has been created.

The RR will be based on a sketch provided by Taylor and Meek [25]. However mass and R&D will be evaluated based on the information previously provided for the Ant. The RR will be equipped with a blade and a sintering machine to perform its tasks. From this point onwards, the latter will be referred to as "microwave". The blade will be used to move the regolith to the modules after which the microwave will sinter it.

A robust structure is needed to make sure the RR can gather the regolith without getting stuck. The width of the entire structure will be 1.32 m and the length will be 2.1 m. The wheels are based on the wheels of the ARV, however are less thick: 14 cm wide. The microwave can sinter a width of 1.0 m regolith, with 2 cm on both sides left, to ensure the back wheels will not touch the hot, sintered soil. The length of the microwave will be 60 cm. The roller and microwave will be positioned underneath the RR, both with a width of 1.0 m. A CAD drawing of the RR can be seen in figure 9.4. It is assumed that regolith needs 5 minutes of microwaving to melt and become hard. Combining this with the length of the microwave, the sintering speed of the RR is equal to 2 mm/s.

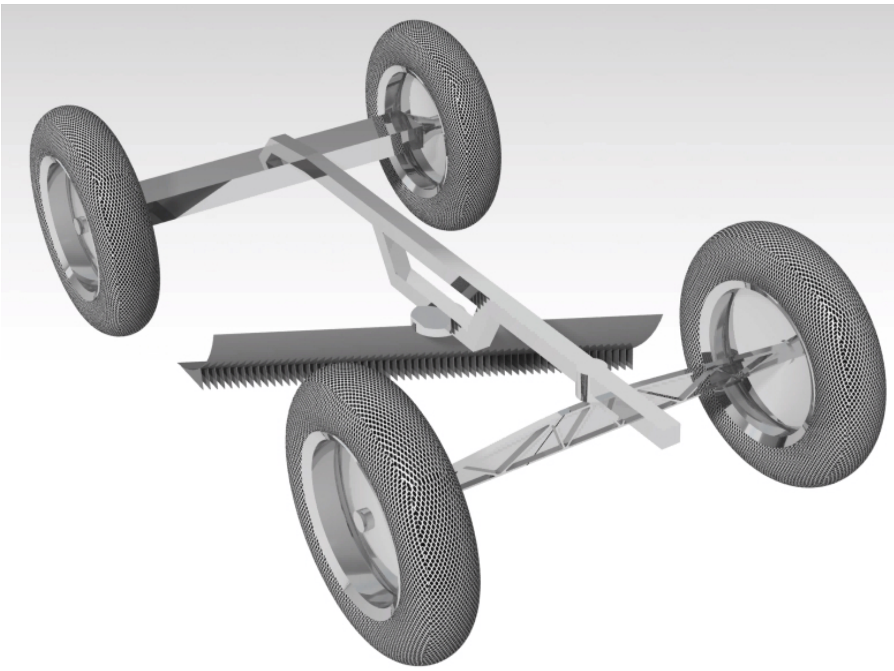


Figure 9.4: CAD render of the Regolith Rover based on the concept proposed by Taylor and Meek [25]

The first step for covering the modules with regolith is to create a ramp, this will be done layer by layer, each time making the ramp a bit more steep and long. Then, two layers of regolith will be sintered on top. To make sure the material of the module does not melt due to the hot temperatures of the microwave, the first layer will have a thickness of 65 cm, as it is estimated that sintering regolith will harden the first 50 cm [25]. Then, another layer of 55 cm will be shoved and sintered on top of the modules. Finally, a third layer of only 25 cm is sintered to create a final thickness of 120 cm. To ensure all elements outside of the habitat can still be connected to the modules, spots on the habitat will not be sintered. To ensure the rovers do not sinter these parts, tubes are put on the connection points. These tubes will stick out for 1.5 m, to ensure no regolith can get into the connection points. During sintering, the rovers drive around the connection point, leaving them open for further expansion purposes. It should be noted that power requirements are not given since no information about this matter is readily available. This should most definitely be looked at in the future as it will be the main driver in the choice for power supply.

The basis of the design is already estimated within the budget of the Ants, and the specific systems are based on already existing technologies. However, the combination and appliance of a sintering mechanism together with a rover has never been tested before. Therefore, only the R&D cost of the microwave technology is taken into account and is estimated to be around 50% of the Ant's total expense, resulting into a cost of €350 millions. The mass of the RR is assumed to be equal to that of the Ant, equalling 900 kg.

9.3. Return Vehicle

As part of the mission, the astronauts will have to get back to Earth after one year. For this reason and for any emergency situation in which an immediate return to the ground station has to take place, a return vehicle shall be present at all time. For simplicity, it is chosen to use the lander that brought the astronauts to the Moon, which is based on the design of the Altaïr Lunar lander [147]. Altaïr is a developed descend and ascend system within the constellation programme of NASA, supposed to fly in 2018 but eventually cancelled due to budgeting issues [148].

Despite this, Altaïr is a fully developed concept, which only needs construction and testing. The weight of the lander is estimated to equal 37050 kg, of which 6150 kg is given for the ascending stage. This stage has a pressurised volume of 17.5 m^3 and a payload capacity of 14.5 $tonnes$ which can bring four astronauts to the Moon and back, meeting all applicable requirements of this mission [147]. The total development costs for Altaïr are estimated to equal €10.3 billions [149]. This estimation includes the first stages of the development, which have already been completed. However, this will still be included in the lander's expense, as Altaïr was designed for missions no longer than 210 days. Hence, some design changes will be needed to be used for one full year, including extra protection against radiation and meteoroids.

9.4. Concluding Remarks

In order to land the payload with precision, the lander will be equipped with a Lander Vision System and a Navigation Doppler Lidar which uses the concept of terrain relative navigation. The Payload will consist of three different kinds of vehicles: an ARV to carry the astronauts, a regolith rover to pave the landing site and roads, and finally an Ant rover to carry and transport the modules for installation. For emergency safety measures, a return vehicle based on the Altaïr concept will be used, as this has already been fully developed. Estimates of R&D costs were provided per system and would equal €10.85 billions in total for the precision landing mechanism, all the rovers, and the return vehicle.

10
System Integration

Now that all the required subsystems have been designed to an acceptable level of detail, a full integration of systems and budgets can take place. This chapter will first present all the Technical Resource Budgets and their management history in section 10.1, followed by an explanation of the integration of the designed elements in section 10.2. Lastly, a detailed description of the internal layout can be found in section 10.4.

10.1. Resource Budgets

Resource budgets established in the Baseline Report [2] and Midterm Report [3] have been continually updated throughout the development process. To not have exponential growth of design parameters such as mass and power, frequent checkups of the current values have been made, and adjusted when necessary.

10.1.1. Budget History and Contingency Management

Over the course of the project, the team has kept track of two main parameters, mass and volume. Because cost is so interlinked to mass, and given the fact that no adequate cost estimation has been made up to now, the cost budget has not been included yet. It is, however, recommended to take this report's estimation as a baseline for such a budget in the future development of the project.

Mass Budget History

The team has made estimations of total mass during the three last phases of the project. In figure 10.1, the change of the project's total mass can be found for each of the checkup moments.

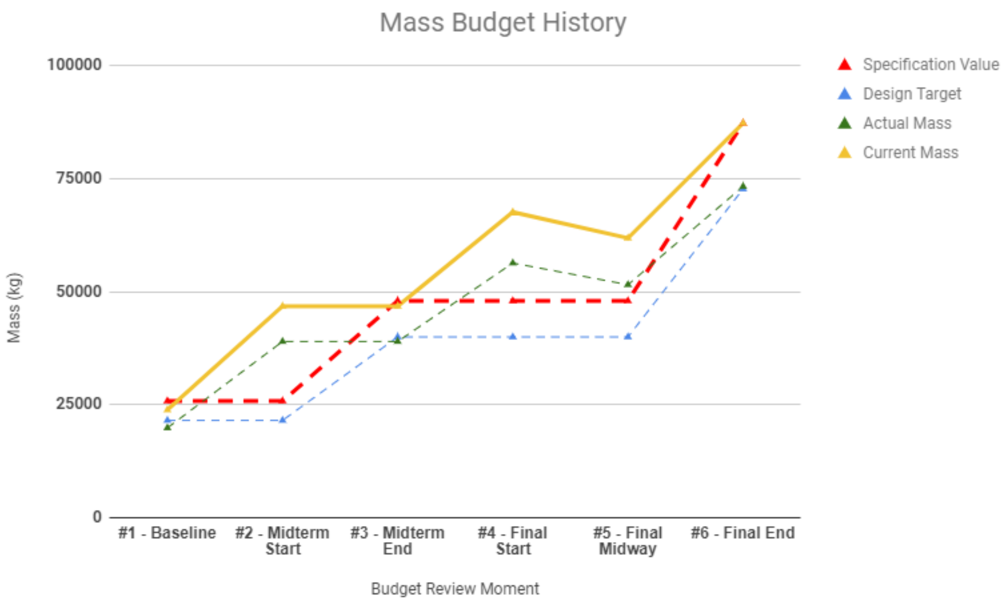


Figure 10.1: Budget history - total mass

Clearly, mass has been growing considerably during the project. At the start of the midterm phase, a new estimate of total system mass was made using a more refined parametric model. As the team deemed this estimate to be more accurate, the design target and specification value were updated at the end of the midterm phase. As the project moved through the final phase, the actual mass increased quite rapidly. During the first control moment, the total estimate of structural mass was brought down by performing more detailed design work. At the final control moment, however, the team included the airlock, node and water tank modules, and the rovers and astronaut vehicle in the total mass calculation. Given that these elements are required for mission success, the design target was adjusted to include the new modules. Currently, the mission is about 3 tonnes over budget concerning mass of the cylindrical modules.

Volume Budget History

The volume estimations made during the different phases of the project can be found in figure 10.2.

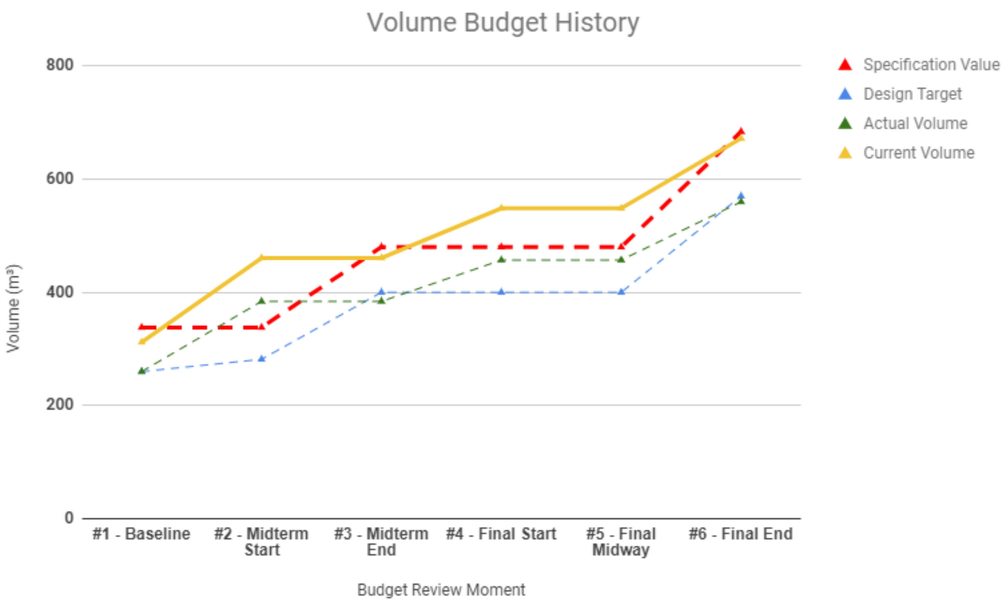


Figure 10.2: Budget history - total volume

Clearly, the total volume of the habitat has also grown considerably over the course of the project. An explanation for the first significant growth from baseline to midterm and the following specification adjustment is the fact that during the baseline phase of the project the estimation was based on requirements only, whereas the first estimation made in midterm was based on the preliminary layout. During the final phase of the project, these layouts were redone, and an airlock and node module were added, leading to the final volume of 672 m^3. Note that only 78% of the 672 m^3 can be counted as pressurised volume (The inner diameter of 4.08 m vs the outer diameter of 4.6 m). Combined with the fact that the secondary structure takes up almost 50% of the inner cylinder, the team has decided to adjust the target and specification value accordingly.

Recommendations for Future Budgets

As the project moves into the next design phases, a couple of budgets become attractive to monitor in a similar manner. Firstly, as mentioned earlier, a cost budget can benefit from strict control, keeping the total project cost within acceptable limits. Next, with an initial setup of the power system available, keeping track of the total power budget is recommended. Lastly, a suggested mission schedule will be established in chapter 13, and keeping track of estimations for this timeline can help the project meet new deadlines in the future.

10.1.2. Mass Budget and Distribution

The final version of the mass budget, after having implemented the last contingency measures, is presented in table 10.1. It implements all the subsystem budgets as mentioned in chapter 8, and shows the contingencies based on design maturity. The ECLSS systems are made up of the ECLSS rack, medical suite and the fixed accommodations.

Table 10.1: Final mass budget

System	Calculated value $[kg]$	Target value $[kg]$	Margin $[\%]$	Current value $[kg]$	Budget $[kg]$
Primary Structure	5360	5120	20	6432	6144
Secondary Structure	12640	12000	20	15168	14400
Radiation protection	0	2000	20	0	2400
Meteoroid protection	0	2000	20	0	2400
EPS	4093.9	1080	15	4708	1242
Thermal control	2380	1400	20	2856	1680
Communication	40	60	20	48	72
CDH	475	400	20	570	480
ADCS	340	400	20	408	480
Water & oxygen	4471	4000	20	5365	4800
Other consumables	8569	8000	15	9853	9200
ECLSS systems	5124	4000	20	6149	4800
Airlock	5263	0	20	6315	0
Node	6911	0	20	8293	0
Rovers	7183	0	20	8620	0
Protective water	5000	0	20	6000	0
Astronaut vehicle	5458	0	20	6550	0
Total	73308	40460		87309	48070

As can be seen in table 10.1, the cylinder mass is still 3461 kg over budget, ignoring all additional elements introduced by mission logistics. Reduction of the mass can be achieved by further development of the relative subsystems, or by changing the budgeted values. This last action can be justified by the fact that the level of design detail has increased significantly, allowing the team to make a more accurate prediction of the final subsystem & full system mass. Clearly, additional budget needs to be established for the new elements, as was described in subsection 10.1.1. With an established mass budget, a detailed mass distribution can be made. As determined in the Midterm Report [3], the allowable mass for a single cylinder is approximately 11.5 metric tons due to landing constraints. The systems per cylinder are presented in table 10.2.

Table 10.2: Mass distribution for the four cylindrical modules

#1 System	Mass $[kg]$	**#2** System	Mass $[kg]$	**#3** System	Mass $[kg]$	**#4 (SCM)** System	Mass $[kg]$
Structures	5400	Structures	5400	Structures	5400	Structures	5400
ECLSS rack	3038	Thermal	476	Thermal	476	Thermal	476
Thermal	476	CDH	54	CDH	54	Antenna	15
CDH	54	ADCS	100	ADCS	100	Central server	300
ADCS	100	EPS (internals)	300	EPS (internals)	300	ADCS	100
EPS (internals)	300	O_2 tanks	603	O_2 tanks	603	EPS (internals)	300
O_2 tanks	201	N_2 tanks	385	N_2 tanks	385	O_2 tanks	201
N_2 tanks	385	Food	1931	Food	1931	N_2 tanks	385
Food	483	Water	1503	Water	1503	Food	483
Water	376	Misc. supplies	1092	Misc. supplies	1092	Water	376
Misc. supplies	546	Restraints	25	Restraints	25	Misc. supplies	872
Restraints	25	Medical suite	300			Restraints	25
Exercise eqpt.	217.5	Test eqpt.	150			Medical suite	900
Trash compactor	225					SCMLS	1000
Laundry	252					Waste collection	169.5
						Galley	352.5
						Mntn. storage	394.5
Total	12077	**Total**	12318	**Total**	11868	**Total**	11748

Clearly, all the modules are slightly too heavy for the 11.5 tonnes requirement. To get to the correct mass for landing, portable supplies can be taken out of the modules. The number of supplies taken out differ from module to module and are presented in table 10.3.

Table 10.3: Supplies taken out of the modules to meet the landing requirements

System	Unit	Module 1			
		1	2	3	4
Food	$[kg]$	280	500	370	35
Miscellaneous supplies	$[kg]$	300	330	0	200
Total out	$[kg]$	580	830	370	235
Total	$[kg]$	11497	11488	11498	11486

As can be seen in table 10.3, the new total masses meet the required landing mass. The removed consumables will be brought to the Moon by placing them in another launch, which will be described in subsection 11.2.3.

10.1.3. Volume Budget and Distribution

Given that volume is somewhat limited in the habitat system, a volume budget and distribution were made in addition to the mass budget and distribution. The total required volume per system is presented in table 10.4.

Table 10.4: Final volume budget for the cylindrical modules

System	Calculated value $[m^3]$	Target value $[m^3]$	Margin $[\%]$	Current value $[m^3]$	Budget $[m^3]$
Structures	300	322	20	360	386.4
Radiation protection	0	0	20	0	0
Meteoroid protection	0	0	20	0	0
EPS	14.1	8	15	14.8	8.4
Thermal control	0	0	20	0	0
Communication	0	0	20	0	0
CDH	1	1	20	1.2	1.2
ADCS	0	0	20	0	0
Consumables	46.61	50	20	56	60
ECLSS	26.63	30	15	32	36
Total	390	460		466	492

Clearly, a few subsystems do not have a budget currently assigned. For radiation protection, meteoroid protection, thermal control and noise control, the respective volume is included in the structural volume. This can be justified by the fact that these systems are all part of the structural layup of the cylinder. Communication and ADCS are both external subsystems and are not included in the cylinder volume budget for this reason.

All internal systems can also be subdivided per module. This distribution of volumes is presented in table 10.5. The thermal system has been allocated some volume in the distribution to represent internal control systems, whereas the EPS has been allocated no volume since all internals can be integrated into the structural volume. The volume for this system (EPS) presented in table 10.4 represents the actual reactor, which is not an integral part of the cylindrical modules. Lastly, a discrepancy exists between the two CDH budgets, which can be explained by the fact that a small portion of the CDH volume budget has been allocated to the node and airlock elements.

Table 10.5: Volume distribution for the four cylindrical modules

Module							
#1		#2		#3		#4 (SCM)	
System	Volume $[m^3]$	System	Volume $[m^3]$	System	Volume $[m^3]$	System	Volume $[m^3]$
Structures	75	Structures	75	Structures	75	Structures	75
ECLSS Rack	10.8	Thermal	1.2	Thermal	1.2	Thermal	1.2
Thermal	1.2	CDH	0.12	CDH	0.12	Central server	0.631
CDH	0.12	ADCS	0	ADCS	0	ADCS	0
ADCS	0	EPS (internals)	0	EPS (internals)	0	EPS (internals)	0
EPS (internals)	0	O_2 tanks	1.629	O_2 Tanks	1.629	O_2 tanks	0.543
O_2 tanks	0.543	N_2 tanks	1.9	N_2 Tanks	1.9	N_2 Tanks	1.9
N_2 Tanks	1.9	Food	6.72	Food	6.72	Food	1.679
Food	1.679	Water	1.614	Water	1.614	Water	0.4035
Water	0.4035	Misc. supplies	8.15	Misc. supplies	8.15	Misc. Supplies	6.8
Misc. supplies	4.075	Restraints	0.135	Restraints	0.135	Restraints	0.135
Restraints	0.135	Medical suite	1.65			Medical suite	3.96
Exercise eqpt.	0.285	Test eqpt.	1.35			SCMLS	3.55
Trash compactor	0.45					Waste collection	1.11
Laundry	2.25					Galley	1.595
						Mntn. storage	1.47
						Antenna	0
Total	99	**Total**	99	**Total**	96	**Total**	116

10.1.4. Power Budget

The power budget for the current design level can be seen in table 10.6. The total power available follows from the first estimation made in the Midterm Report [3], whereas the respective subsystem budgets have been allocated based on preliminary estimations.

Table 10.6: Final power budget

System	**Calculated value** $[W]$	**Target value** $[W]$	**Margin** [%]	**Current value** $[W]$	**Budget** $[W]$
Structures	0	0	20	0	0
Radiation protection	35	40	20	42	50
Meteoroid protection	0	0	20	0	0
EPS	0	0	15	0	0
Thermal control	1403	1600	20	1684	2000
Communication	160	200	20		250
CDH	0	0	20	0	0
ADCS	553	640	20	664	800
Water & oxygen	0	0	20	0	0
Water System	500	560	15	600	700
ECLSS systems	11228	13000	20	13474	16250
Total	13879	16040		16463	20050

The leftover power can be beneficial to charge up the rovers, even in the case of a "peak power" situation.

10.2. Design Integration

Below, the integration of different elements is discussed.

Integration of the Water System

As water is used in many systems, integration is necessary. First, it is essential to consider the separating of water. Clean water needs to be separated from grey or used water. Furthermore, water used in the inner and outer thermal system also needs to be separated from consumable water and kept in a closed loop system. To separate the different types of water, several separate tank systems are used. The pipeline connections between modules is built into the docking ports.

Routing of Cables

Cables can be routed from one module to the other by the built-in connection system of the docking port. All cabling will be placed behind the walls, under the floor or above the ceiling of the interior module.

Furthermore, redundant cables will be present to limit the consequences of a damaged cable.

Airlock, Module Matching

The door of the airlock will be aligned with the floor of the habitation modules, as already was taken into account during external layout. All cables can thus quickly be routed from the walls and the ceiling through the docking mechanism into the airlock and back.

Integration of the Thermal System with Radiation Protection

In section 7.3 it was explained that a layer of water is needed to protect the astronauts against radiation. This layer will be placed in the fuel tank, which surrounds every module. However, as explained in section 7.3, only the top part of this tank is used to put water in. In section 7.4, it was checked if this water would freeze, which was not the case.

The water used to protect against radiation will be integrated with the thermal control system, which is described in figure 8.3.4. This is beneficial since the water which is warmed up by the heat of the systems will be cooled while circulating through the old fuel tank. The pipelines needed for this circulation system will be built in on Earth, while the water will only be put in on the Moon.

Elements Outside Habitat

As radiators and the communication antenna need to be placed on the outside of the habitat, a connection has to be designed for the cables and pipes to go through both the structure and the layer of regolith. The cables and pipes will enter and leave the structure close to the docking mechanism, as there is no layer of water present in that part of the wall. To be able to guide the cables and pipes through the regolith, the regolith cannot be sintered. Therefore during sintering, a few spots will be skipped as is further explained in subsection 9.2.4. The radiators and communication antenna can be attached to the module close to these openings. The same openings can also be used to guide the power cabling through.

10.3. System Diagram

The last system diagram can be seen in figure 10.3, it shows the interconnections of all the active systems that were designed.

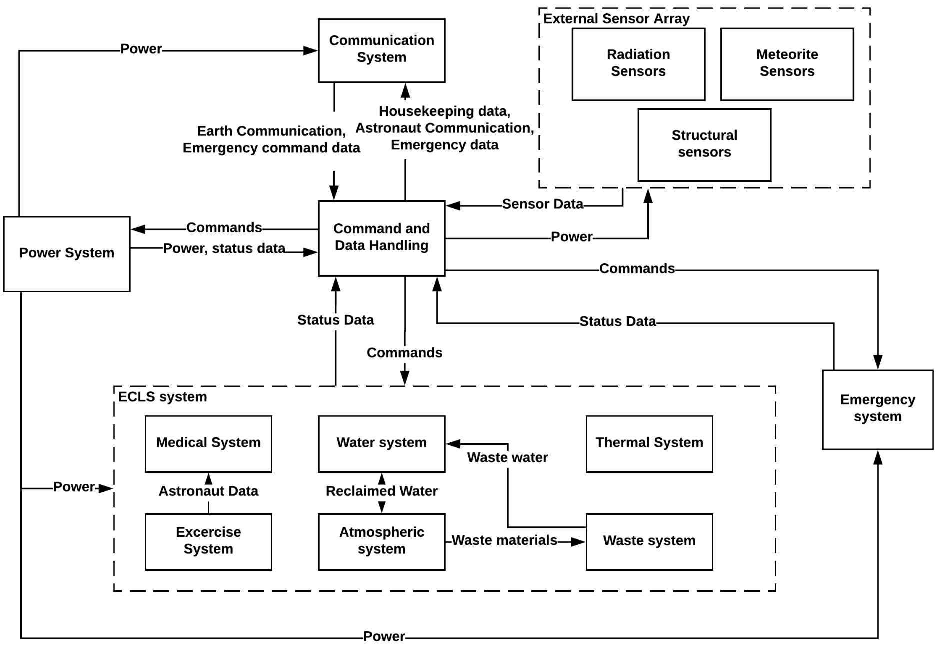


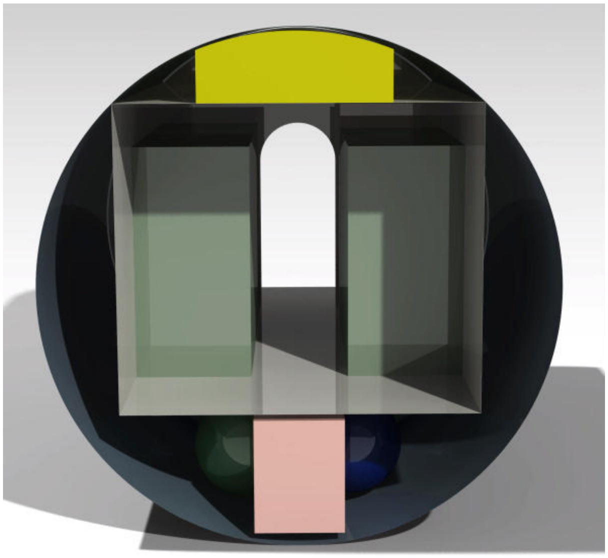
Figure 10.3: System hardware diagram

10.4. Internal Layout

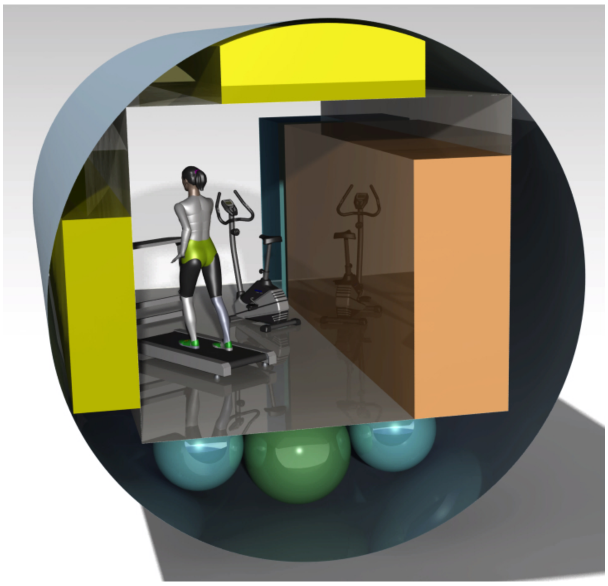
The internal layouts of the modules are discussed in this section.

10.4.1. Module 1

The first module, as can be seen in figure 10.4a, contains the ECLSS system, as well as the bathroom and laundry facilities. Furthermore, storage space is available for the storage of food, water, miscellaneous supplies, tanks and cabling. Lastly, the medical bay is located in this module, separated from the rest of the module by a plastic screen.



(a) Module 1, contains the ECLSS, med-bay, bathroom & laundry and a variety of storage space



(b) Module 2, contains the exercise equipment and a generous amount of storage space

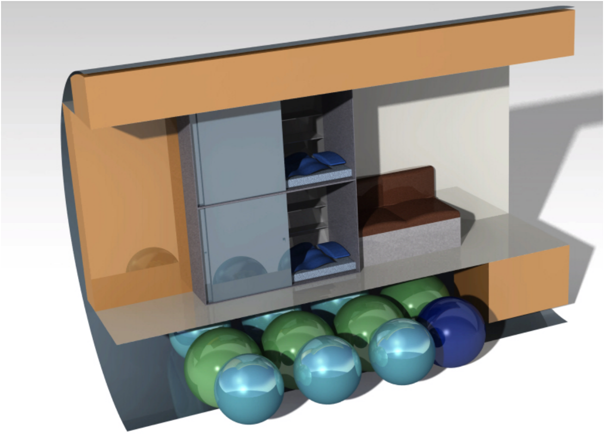
Figure 10.4: Renders of modules 1 and 2

10.4.2. Module 2

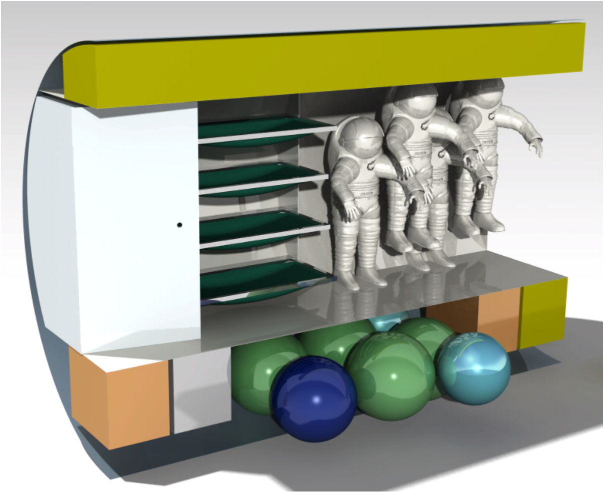
The second module, which can be seen in figure 10.4b, contains a generous amount of storage space. Its most important function, however, will be the exercise compartment. Furthermore, in the evening when some astronauts will go to bed while others still want to talk, space is available to put the foldable table in this module. Lastly, a backup version of the medical equipment is present as well as test equipment to perform basic research.

10.4.3. Module 3

This module is designed to be the module where the astronauts will live and therefore doubles as a quiet room. The module contains four capsules which include a bed and personal storage space. In these capsules, space for photos as well as other personal belongings is present. The design was based on capsule hotels, as the structure allows for privacy, as well as personal regulation of environmental conditions. Next to this a sofa is placed alongside one of the walls. Opposite, a foldable table is placed. Storage for entertainment is present, as well as extra foldable chairs, making it a sociable area. Again storage for essentials is also available. figure 10.5a gives an impression of the module.



(a) Module 3, the astronaut living compartment



(b) Module 4, the ECM. Also contains the galley, central server and other essential systems for standalone operation

Figure 10.5: Renders of modules 3 and 4

10.4.4. Module 4

Subfigure 10.5b shows the safety module. Under normal operating conditions the module will function as kitchen and storage room for medical equipment. The central server is also stored in the module, together with maintenance equipment. To design the module for emergencies, EVA suits, a toilet, a down-scaled ECLSS as well as emergency provisions are present.

11
Logistics and Operations

Having fully developed all subsystems that will be part of the habitat and having thoroughly integrated them into the lunar habitat, the design of the product itself is complete. This chapter discusses an in-depth analysis of the logistics and operations of the mission, describing all the steps that bring the product from merely a design to a ready to be used stage. First, a general approach to the logistics flow is covered in section 11.1, then the chronological sequence of operational events is discussed, from the transfer to the Moon in section 11.2, to the installation plan in section 11.3, which leads the habitat to being in fully operational state.

11.1. General Approach

The general logic behind this chapter is purely chronological. To outline the sequence of events needed to bring the product from design phase to full habitat operation, the mission functional flow in figure A.1 is considered. Out of the six first level stages, only four need to be considered from a logistical point of view, namely the transfer to the Moon (FF 3), the installation plan (FF 4), the operation of the habitat (FF5), and the EOL sustainability (FF 6), as identified in figure 11.1. Hence, the collection of resources and packing of elements, i.e. FF 2, is disregarded based on the fact all equipment needed on the Moon, from construction elements to food supply, is assumed to be already stored at the launching site, ready to be loaded. The four stages mentioned are individually analysed in this chapter, besides FF 6, regarding the maintenance and sustainability of the habitat, which is taken care of in chapter 12.

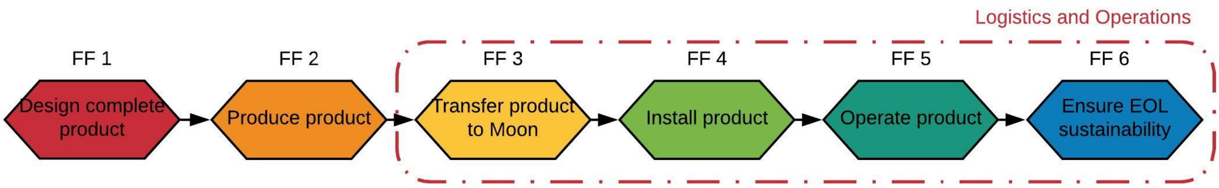


Figure 11.1: High level functional flow diagram in terms of logistics and operations

11.2. Transfer to the Moon

In order to fulfil FF 3, a series of actions needs to be performed, such as establishing the mission astrodynamics, choosing an adequate launch vehicle, determining a launching sequence, and finally the landing procedures.

11.2.1. Mission Astrodynamics

The astrodynamics behind this mission consist of three main phases: Take-off to Low Earth Orbit (LEO), transfer to Low Lunar Orbit (LLO), and landing. The launch is done using a preexisting launcher, which bring a transfer vehicle, a lander and the habitat modules into LEO. One launch is used to bring a fully fuelled lander and habitat module into orbit where they rendezvous with a transfer vehicle, previously launched using a second launcher. The transfer vehicle performs a Trans Lunar Injection (TLI) to bring the vehicles into Lunar Transfer Orbit (LTO) and then executes a second burn to perform a capture into Lunar orbit. This is where the lander and habitat module decouple from the transfer vehicle, in order to de-orbit and proceed for landing on the Lunar surface. In order to determine the characteristics of this transfer, a 2D numerical model was constructed and used to determine the necessary ΔV budgets for the mission, of which the basic assumptions are given below:

- Mission takes place in a two-body system;
- Hohmann transfers are used to minimise the required energy for transportation;
- ΔV can be applied instantaneously;
- Last stage of launcher decouples in LEO;

- Parking orbit around Earth will be at 185 km altitude in LEO;
- The S/C is assumed to be a point mass.

The model calculates the accelerations on the spacecraft due to the influences of the Earth's and Moon's gravity. These values are then integrated twice to get the new position, and finally iterated. The program first uses an Earth-centred reference frame to calculate the trajectory until Lunar orbit and then a Moon-centred reference frame to calculate the landing details. The simulation begins at an altitude of 185 km and at the pericenter velocity of a LTO. The relative position, velocity and accelerations of the Moon, Earth and S/C are calculated for every step unit the spacecraft reaches a distance of 1000 km from the Moon. After that point, the coordinate system is transformed to the Lunar centred one and the landing trajectory is calculated. The final results of the simulation are visualised in figure 11.2. On the left side of the figure, the trajectory to the Moon can be seen. The orange (lighter colour) line indicates the the LTO trajectory without the influence of the Moon, whereas the red line (darker colour) indicates the trajectory due to the influence of the Moon's gravity. On the right side, the trajectories around the Moon are visualised. The velocity increments needed for capture in Lunar orbit, de-orbit and landing are the outputs of the simulation, which are collected in table 11.1.

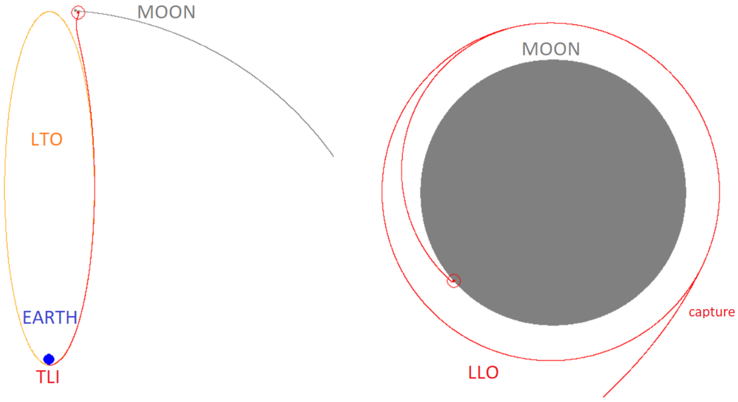


Figure 11.2: Trajectory of the mission to the Moon

Once the required ΔVs are calculated, the mass fractions can be determined using equation 11.1, where M_0 is the initial mass, M_1 the final one, g_0 the gravity on Earth, and I_{sp} the specific impulse (a measure for efficiency) of the rocket engine, in seconds. Following the studies done by Wade [150], values of 324 and 303 s have been selected for the hypergolic second stage and for the Lunar descent propulsion system from Apollo, respectively [151]. The resulting values are reported in table 11.1.

Table 11.1: Delta-V budgets and mission mass fractions

Parameter	**Unit**	**Value**
ΔV transfer	km/s	3.229
ΔV lander	km/s	1.980
M_{LEO}/M_{LLO}	–	2.76
M_{LLO}/M_{surace}	–	1.91

$$\Delta V = I_{sp} \cdot g_0 \cdot \ln\left(\frac{M_0}{M_1}\right) \tag{11.1}$$

Model Flow Chart Verification

Figure 11.3 represents the flow chart of the aerodynamics model, which runs through the main loop twice as shown by "1" and "2" in the last two blocks of the loop. The first is focused at the Earth-Moon spacecraft system for the transfer orbit, while the second one on the Moon-spacecraft system for the landing trajectory.

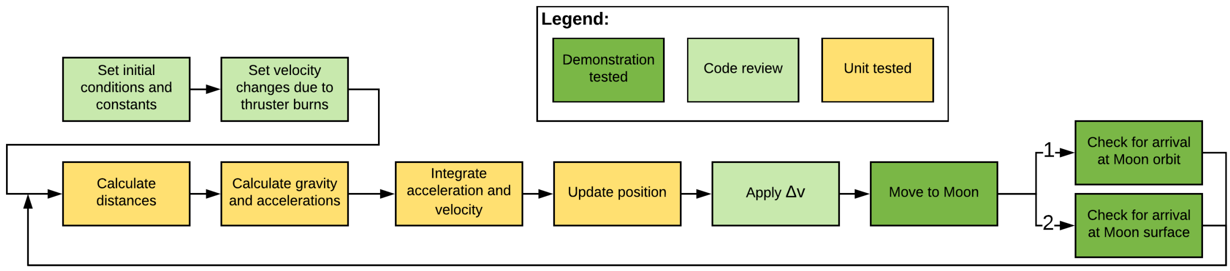


Figure 11.3: Flow chart of the 2D astrodynamics model

Unit Tests

As it can be seen in table 11.2, all unit tests conducted on the astrodynamic model showed positive results. It should be noted that the 'move to Moon' block shown in figure 11.3 is only tested visually, as it uses the same physics engine as the S/C in the model. Given that the Moon in the simulation moves with the correct orbit and it uses the same engine, it was determined that this was sufficient to verify this particular process. The remaining demonstration tests were conducted by printing the system status at regular intervals and checking for expected behaviour. The actual numbers have not been included as they do not add any meaningful value to the results.

Table 11.2: Unit tests conducted on the 2D astrodynamics model

Code tested	**Unit testing procedure**	**Result**
- Gravity field determination	- Using standard input, $\pm$ 1 km, and comparing with verified manual calculations	- 100% match in 3 cases, 7 significant figures
- Distance calculation	- Applying same procedure as before, for 3 different pairs of (x, y) positions	- 100% match in 3 cases, 7 significant figures
- Acceleration calculation	- Checking with manual calculations the directions, signs and magnitude for 3 different object configurations	- 100% match in 3 cases, 6 significant figures
- Velocity determination	- Running 3 cycles for different accelerations and checking with manual calculations	- 100% match in 3 cycles, 6 significant figures

Integration Tests

In order to test the integration of units within the astrodynamic model, a simpler two body system was used by turning the Moon's gravity off. This is because analytical methods cannot solve a three-body problem. Distance, velocity and acceleration at the apogee of the uninfluenced LTO were retrieved and compared to the analytical values, as shown in table 11.3. The small error differences could mainly be due to the fact that the program makes use of small discrete time-steps. Therefore the closest to apogee it can get, with a time-step of 0.1 s, is 1.4 m.

Table 11.3: Percentage differences between conics and the 2D astrodynamics model

Apogee parameter	**Unit**	**Value**		**Difference**
		Model	**Conics**	[%]
Distance	km	382737	382563	0.04558
Velocity	m/s	187.39	187.47	0.04488
Acceleration	m/s^2	0.002724	0.002721	0.09109

11.2.2. Launcher Selection

Since a typical payload mass for space mission of this size lies in the twenty tonnes range, a very powerful vehicle is needed. A study has been performed on the most current and near future launchers to assess their performance and driving characteristics such as cost or experience. These are evaluated in table 11.4, where * refers to the LTO capability, ** to boosters excluded, and *** under critic debate of feasibility.

The stringent typical mass eliminates almost all the existing launchers, excluding a mere five of them: Falcon Heavy (FH), Space Launch System (SLS), New Glenn, Big Falcon Rocket (BFR), and Long March 9. From these, the BFR and Long March 9 are still in the design phase and can therefore not be relied upon to deliver the launcher within any time frame. Even though it is supposed to launch within two years, the same goes for New Glenn, as very little concrete information or design details are available. This leaves FH and SLS as the two remaining options. Because of the ever-increasing importance of sustainability, this will be the final constraint for selecting the launcher. The propellants of FH are liquid oxygen and RP-1, a highly refined version of kerosene, whereas SLS burns on a combination of liquid oxygen and hydrogen, which eliminates the exhaustion of carbon dioxide into the atmosphere. However, the SLS does use two solid rocket boosters using PBAN and APCP, one toxic and one harmful to the environment, respectively. Furthermore, it is entirely expendable, whereas the three boosters and fairing of the FH are recovered, leaving just an expendable small second stage. Taking this into consideration, together with the enormous price tag associated with the SLS, the FH is the most optimal choice and will be the prime candidate for launch vehicle selection.

Table 11.4: Collection of potential existing and near future launchers [152–166]

Launcher	**Company**	**Country**	**Mass** **LEO** $[ton]$	**Mass** **GTO** $[ton]$	**Price** $[M\$]$	**Maiden flight**
Falcon Heavy	SpaceX	USA	63.8	26.7	90	2018
Delta IV Heavy	ULA	USA	28.37	14.21	350	2004
Long March 5	CALT	China	22	14	110	2016
Atlas V	ULA	USA	18.85	8.9	109	2002
Ariane 5	Arianespace	EU	20	10	165-220	1996
Proton-M	Khrunichev	Russia	22	6.6	105	2001
Vulcan Centaur Heavy	ULA	USA	35.38	15.88	99**	2019
SLS	NASA	USA	77+	45*	1500-2500	2020
New Glenn	Blue Origin	USA	44.906	29	-	2020
BFR	Spacex	USA	150	-	62***	2024
Long March 9	CALT	China	140	50*	-	2028

By inspection of the FH fairing as described in chapter 6, it was determined unfeasible to place the habitat modules and the required fuel tanks in the available volume. The required fuel to take 20 tonnes of payload from LEO to LLO is approximately 35 tonnes, using the mass fractions described in subsection 11.2.1. The best alternative is then to use two launches, one to lift the lander and habitat module and one to bring the transfer vehicle to LEO, where they would rendezvous. Hence, if the Falcon 9 is used to launch the habitat and lander, 22.8 tonnes could be brought to LEO, which would imply 40.1 tonnes of fuel to bring this mass to LLO. To lift the fuel and transfer vehicle, the selected launcher is finalised to be the Falcon Heavy.

11.2.3. Launching Timeline

The timeline of the launches shall be coherent to the logistics of the building procedures of the habitat. Logically, the items that will be brought to the Moon with the initial launches are the ones that will be needed first for installation purposes. Hence, the RRs and the Ants will be needed from the very first launch on, to pave the habitat location and to transport the habitat elements, respectively. The nuclear plant will be needed to provide power for communications and data tracking, hence it is vital that this shall arrive in the first launch. The node is the central point of the habitat, therefore it would be convenient if this was the first element to be positioned, after which the four modules can be attached to it. The airlock can only be brought to the Moon once the last module is installed, together with the high-gain antenna that is attached to it. With the airlock being connected, the structure of the habitat is complete and the walls may now be filled with the protective layer of water. Finally, the astronauts arrive to the habitat.

Following this logic, the mission inventory is distributed across several launches in such a way that the weight and volume constraints of the chosen fairing are not exceeded; these are 11500 kg and 166.7 m^3 maximum. The goal is to distribute the total mass of the whole payload in the most effective way possible, such that the number of launches is minimised. The distribution of the payload in the required number and order of launches is collected in table 11.5, whereas figure 11.4 represents it graphically. For clarification purposes, RR refers to regolith rover, NPP to the nuclear power plant, supplies to the food and other miscellaneous payload that could not fit in the four modules, HGA to the high-gain antenna, ATV to astronauts transfer vehicle, and finally RV to the return vehicle. Furthermore, the item envelope for launch 1 and 7 refers to the structure that shall be designed in order to bond the other items present in the launch. Whereas the modules or the node are individual elements, directly attached to the lander's engines, launches 1 and 7 contain multiple parts of payload, hence they need a supporting system that keeps them together during transfer to the Moon. The mass of the envelope has been estimated to be 500 kg, whereas the volume has not been calculated as outside the scope of this project. However, given the remaining volume of the fairing after loading the other items of precisely 127.2 and 89.0 m^3 for the two launches under consideration, it is assumed that such a structure would fit in the fairing. Finally, the ninth launch consisting of the RV and astronauts is assumed to have 0 kg or m^3 left, as this vehicle shall be designed entirely based on the structure of the RV, leaving no extra space in the fairing.

The modules are purposely designed based on the given maximum mass requirement, hence they take up almost all the allowable weight in the fairing. On the other hand, the other launches are not as space or mass efficient, as it can be noticed in table 11.5. A total of 9441 kg and 610.4 m^3 are left out from the maximum allowable payload within the nine launches, which gives the opportunity for further elements such as testing facilities or additional supplies.

Table 11.5: Distribution of resources across a number of launches

Launch	Item	Amount	Item		Launch total		Remaining	
			Mass $[kg]$	Volume $[m^3]$	Mass $[kg]$	Volume $[m^3]$	Mass $[kg]$	Volume $[m^3]$
1	RR	4	900	4	11208	39.5	292	127.2
	Ant	4	900	4				
	NPP	1	3508	7.5				
	Envelope	1	500	-				
2	Node	1	8293	86.42	10308	97,02	1192	69.68
	Supplies	-	2015	10.6				
3	Module 1	1	11497	124.6	11497	124.6	3	42.1
4	Module 2	1	11488	124.6	11488	124.6	12	42.1
5	Module 3	1	11498	124.6	11498	124.6	2	42.1
6	Module 4 (SCM)	1	11486	124.6	11486	124.6	14	42.1
7	Airlock	1	6315	61.07	7274	77.75	4226	89.0
	HGA-1	1	39	1.5				
	ATV	2	210	7.59				
	Envelope	1	500	-				
8	Water tank	1	1000	5,5	6000	10.5	5500	156.2
	Water	-	5000	5				
9	RV	1	6150	31.8	6550	32	0	0
	Astronauts	4	100	0.0498				
Total:					**87309**	**755.2**	**11241**	**610.4**

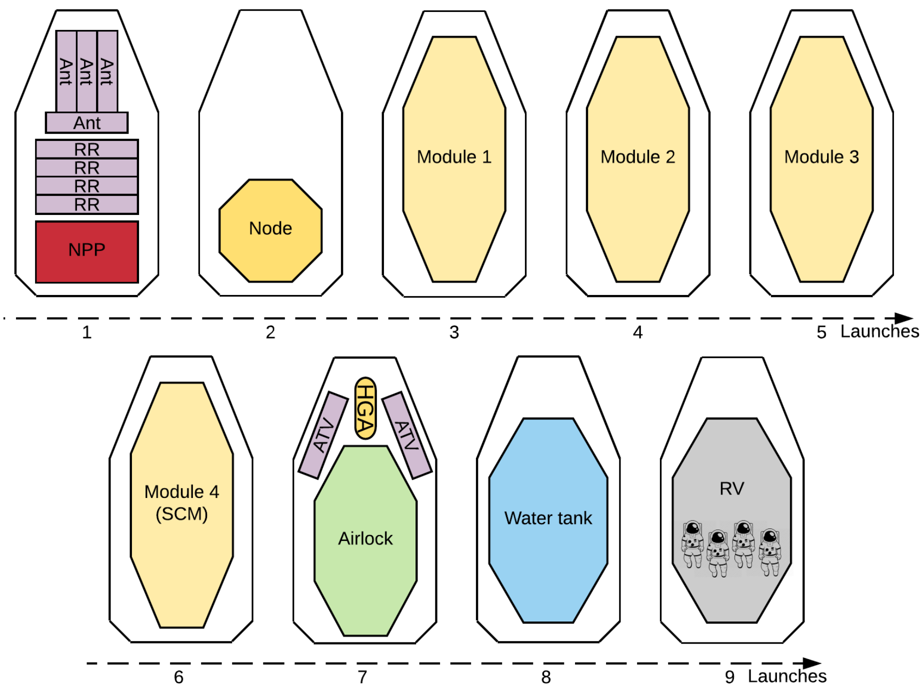


Figure 11.4: Distribution of payload across launches

11.2.4. Landing Procedures

The landing procedure is assumed to be similar to the one designed for the Mars 2020 mission, mainly because the LVS and NDL are being tested and validated for that specific envelope. Once in LLO, at a height of 3 km from the Lunar surface, the LVS will make a coarse image of the Apollo 11 site with several landmarks. The computer of the camera system will already have the real landscape of the surface in its CPU, and that will be compared to the identified landmarks; as the lander descends, more pictures will be made several times per second. Based on these matching results, an input in the differential thrusters and reaction wheels can change the lander's attitude and velocity profile, and therefore its desired landing position. Furthermore, as the vehicle descends even further, the NDL will provide even more accurate data on the altitude, velocity, and attitude characteristics to adopt [167]. For the first landing, an accuracy of around 40 m is assumed to be feasible, whereas it shall increase even higher for the subsequent ones due to the preparation of the site. Once again, it should be noted that this performance is based on test results and it is assumed that the technology will be fully developed at the commencement of this mission.

In terms of landings, the launches described in subsection 11.2.3 are essentially the same, if not for the first one. This is because, as elaborated upon in subsection 9.2.4, regolith can be sintered and thus allowing the creation of paved surfaces. For installation purposes, this shall be done in the earliest possible stage to ease the installation of the habitat, in the first landing. Hence, two different types are distinguished: Unpaved and paved landing. It should be noted that in both cases the procedure is the same, with only one difference relating to the fact that the pre-loaded images will be different.

11.3. Installation Plan

Having established a chronological launching and landing sequence, the installation plan can now be detailed to fulfil FF 4 of figure 11.1. For this purpose, a timetable has been set up in which it is assumed that the mission, besides design phase and collection of resources, will start on day one of week one. The daily schedule is shown in figure 11.7. The index 'P' refers to preparation, 'TO' to take-off, and 'L' to landing: It takes four days to prepare the launcher and six to land it. Every two weeks the Moon undergoes a full Solar eclipse, which is referred to as hibernation (H); these are weeks 2 and 3, 6 and 7, and so on. During this period, the rovers cannot operate because of the lack of Sun for recharging their batteries, hence no installation activity is performed. Also, the first day of 'sunlight' is not considered ideal for landing, given that the temperature is still fairly low and thus not ideal for the performance of the robots. In the list below, a week-by-week description of figure 11.7 is given:

- Week 1 With the RRs, the Ants, and the NPP, the first launcher is prepared on day 1 and launched on day 5;
- Week 2 Six days after take-off, the fist landing of the mission takes place, where one day of margin is given for allowing the regolith to settle back down on the Lunar surface. The entire location is scanned, data is sent back to Earth, and based on these pictures the control team on Earth decides upon the position of the installation site, relative to the lander. Only then, the RRs can off-load and start the paving of a 50 x 50 m square for the landing site and an area of 858 m^2 for the habitat itself;
- Week 3 A 5 x 100 m straight road will connect these two paved locations, resulting in a total area of 4200 m^2 to be sintered, and since the RRs can sinter with a speed of 2 mm/s, four of them will take 150 hrs for the entire surface, resulting in just over six days. Three days of margin are allowed for contingency and charging of the RRs' batteries. Day 7 of the week is saved for transporting the NPP to the desired location, which will ensure the RRs and Ants can charge themselves when out of battery. The transporting of the NPP will include the Ant taking it from the rocket, bringing it to the assigned spot and digging a hole to put it in. Finally, a small wall will be build to protect the astronauts from radiation;
- Week 4 Hibernation week, hence no activity on the Moon can be performed. On day 6, the preparation of the second launchpad begins;
- Week 5 On day 3, the second launcher takes off, immediately after which the third one starts being prepared. Again, this takes off on day 7 and this sequence repeats for launcher number 4;
- Week 6 On day 2 of the week, the second landing takes place. Differently from the first one, it does not require one day of margin for the settling of the regolith, since both the habitat and landing locations been sintered and paved. Hence, the Ants can easily transport the node to its location. Four days later, the first module of the habitat lands, is transported and linked with the node. This moment of the installation plan is visually represented by the render of figure 11.5a. In the meantime, on day 4, the fourth launcher takes off;

Week 7 The second module lands on day 3, which is transported and installed with the node. Now that two modules are fully placed, the RRs can be used again to cover the first and second sides of the habitat, as outlined by figure 11.6. This is again visually represented by figure 11.5b. It is assumed to be taking two days for covering each side, and the regolith used shall be taken at a large enough distance from the modules as to not cause any complications;

Week 8 Same as in week 4, the only action is to start preparing the next launcher on day 6, as hibernation hinders any type of installation;

Week 9 Similarly to week 5, the fifth launcher takes off and the sixth one is prepared on day 3 of the week, with the latter subsequently taking off on day 7, together with the preparation of launcher 7;

Week 10 Module 3 is landed, transported and installed on the second day, after which the third and fourth side of the habitat can be covered with regolith: this moment is figure 11.5c. After placing the last module, which is also the SCM, the fifth and sixth sides can be covered as per figure 11.5d, whereas the seventh launcher with the airlock takes off on day 4;

Week 11 Sides 7 and 8 are fully covered, as shown in figure 11.5e; the airlock, together with the HGA, lands and is transported on day 3;

Week 12 Hibernation week, no action performed on the Moon nor on Earth;

Week 13 Same as in week 12;

Week 14 On the first day, the system is initiated and the habitat starts running. If everything goes according to plan, a check is given at the end of the day, which leads to the preparation of the eight launcher on the morning of day 2. This consists of the water to be used to fill up the walls of the modules for protection and takes off on day 6. On the other hand, both sides of the airlock are covered, i.e. 9 and 10. This is visually seen in figure 11.5f. The habitat undergoes configuration for one whole week to initiate and integrate all subsystems;

Week 15 One week of habitat configuration is complete. The top part of the node is covered on day 2 and the regolith layer of the whole habitat is refined for two days straight, in order to ensure a proper protection. The last three days deal with transporting the water tanks and filling up all modules with water, after which the entire habitat is considered fully assembled;

Week 16 Same as in week 12;

Week 17 Same as in week 12;

Week 18 After four weeks of the habitat being up and running, the check can be given to start preparing the launcher vehicle that will bring the astronauts to the Moon, which takes off on the fifth day;

Week 19 During the final week of the installation process, the astronauts finally land and enter the habitat on the fourth day, after which they have three days to perform any EVA that may be required before the next two weeks of eclipse begin again. From this point onward, the habitation has begun.

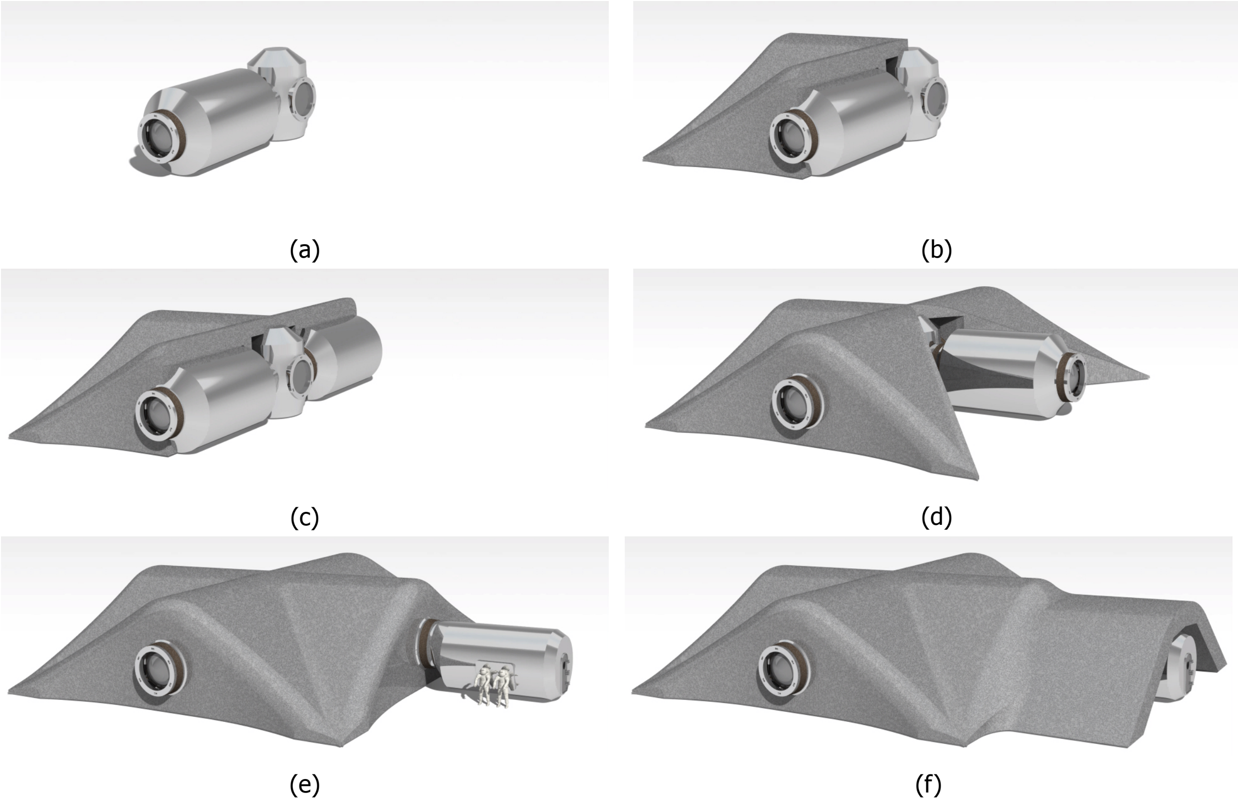


Figure 11.5: Time steps of the installation plan of the habitat

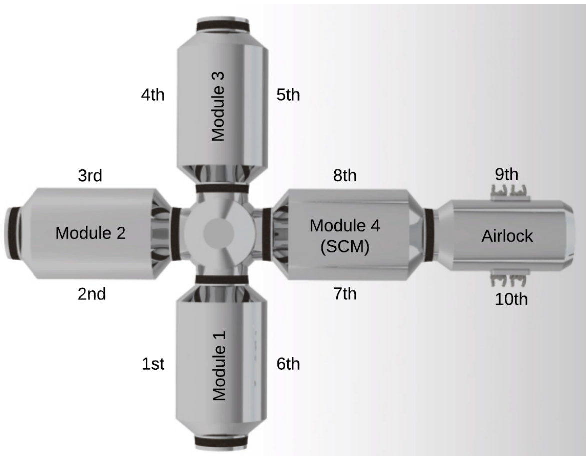


Figure 11.6: Exterior sides of the habitat in the order with which they are covered with regolith during installation

Week	Day 1	Day 2	Day 3	Day 4	Day 5	Day 6	Day 7
1	P 1; H	H	H	H	TO 1; H	H	H
2				L 1; Dust settling; Image mapping	Paving	Paving	Paving
3	Paving	Paving	Paving	Paving	Paving	Paving	Transport NPP
4	H	H	H	H	H	P 2; H	H
5	H	H	TO 2; P 3; H	H	H	H	TO 3; P 4; H
6		L 2; Transport node		TO 4		L 3; Transport module 1	
7			L 4; Transport module 2	Cover side 1	Cover side 1	Cover side 2	Cover side 2
8	H	H	H	H	H	P 5; H	H
9	H	H	TO 5; P 6; H	H	H	H	TO 6; P 7; H
10		L 5; Transport module 3	Cover side 3	TO 7; Cover side 3	Cover side 4	L 6; Cover side 4; Transport module 4	Cover side 5
11	Cover side 5	Cover side 6	L 7; Cover side 6; Transport airlock	Cover side 7	Cover side 7	Cover side 8	Cover side 8
12	H	H	H	H	H	H	H
13	H	H	H	H	H	H	H
14	System start-up; Water check	P 8; Habitat configuration; Cover side 9	Habitat configuration; Cover side 9	Habitat configuration; Cover side 10	Habitat configuration; Cover side 10	TO 8; Habitat configuration	Habitat configuration
15	Habitat configuration	Cover node	Finalise habitat cover	Finalise habitat cover	L 8; Transport water tank	Fill up with water	Fill up with water
16	H	H	H	H	H	H	H
17	H	H	H	H	H	H	H
18	Four weeks check; P 9				TO 9		
19				L 9; Astronauts enter habitat	EVAs	EVAs	EVAs

Figure 11.7: Mission timetable for complete installation plan of the habitat

11.3.1. Habitat Configuration

As shown in week 14 of the logistics timetable, the habitat has to be properly configured. Hence, it is important to discuss in what order the subsystems will be initiated, which is what is provided below:

1. Command, data handling and computer system:

The first system to be turned on, in order to check if all internal connections are linked and data is transmitted properly;

2. Communications system:

The communication system is the second one to start up. It connects to the CDH, and when this is done correctly, data will be sent to Earth. When confirmation of successful receipt is given by the ground station on Earth, the signal is given to start up the life support systems.

3. Pressurisation system:

Initiating the life support systems means starting to pressurise the habitat at first. Since the CDH system is up and running, the internal pressure and atmospheric properties can be measured. When all values are within their required boundaries, the signal is given to start the next system.

4. Water and oxygen recycling systems:

As nothing is yet in use, these cannot be tested on their performance, but a check can be given whether all parts are working correctly, which is measured via the inputs and outputs of the systems. When measurements show positive results, the last subsystem will be started.

5. Thermal control system:

For the last system to be initiated, the protection layer of water within the cylinders has to be present. This is why one week is waited for landing 8 to arrive on the Moon with the water tanks. They will be transported at a temperature of 282 K, in a cube of 5 m^3. At this temperature, the same amount of heat is radiated out as is absorbed. To find this temperature, equation 11.2 is used, which can be rewritten into equation 11.3 for the cube, when using a coating with equal emittance and absorption, such as Martin Black Paint N-150-1 [168, 169]. The internal flux can be neglected, as no systems create any heat inside the cube. For the emitted heat, the complete area of the cube is used, while for the absorpting area only the enlightened side is used. This area is equal to the average area between minimal and maximal enlightening. It was assumed that the transfer from Earth to the Moon will be in sunlight at all times.

However, a contingency plan is needed to ensure the container will stay intact, if it is not possible to pump the water into the habitat before the night starts. This plan consists out of the material chosen for the cube. The actual container is made of a thermoplastic material, which has a so-called plastic envelope, such that it can accommodate the expansion due to ice formation without cracking. The outer shell of the cube will still be rigid, to be able to cope with launching accelerations.

$$A_1 \epsilon \sigma {T_E}^4 = A_2 \alpha_{ab} F_S + A_2 \alpha_{ab} f F_S A_E + A_2 \epsilon F_{E,IR} + Q_i \tag{11.2}$$

$$\sigma T^4 = \frac{\frac{r^2+\sqrt{2r^2}r}{2} \cdot F_S}{6r^2} + \frac{\frac{r^2+\sqrt{2r^2}r}{2} \cdot F_S A_E}{6r^2} + \frac{\frac{r^2+\sqrt{2r^2}r}{2} \cdot r_E^2}{6r^2 \cdot \left(r_E + \frac{D_M}{2}\right)^2} \tag{11.3}$$

With a temperature of 282 K, the water will be liquid when it arrives on the Moon, which means it can directly be put into the habitat. Since at this point the entire habitat will be covered in regolith, besides the front part of the airlock, this will be done via a connection point there, which links all the water tubes and systems to each other, and thus creating a closed loop system. As all systems are running now, the thermal control system is needed to cool down the systems. The CDH system will give a signal to the pumps to initiate them. This will cool down all systems, while the inner temperature of the habitat will converge to the required settings.

6. Astronauts:

When the CDH system measures that the values of all subsystems are within their respective boundaries and the one-month check is sent to Earth, the astronauts will make their way to the Moon, where they will land after ten days. After finally entering the habitat in week 19, they will have three days to perform the last checks with possibly necessary EVAs, before the next two weeks of eclipse begin.

12
Risk Analysis and RAMS

With the subsystem level RAMS & risk characteristics analysed in chapter 8, this chapter will deal with the analysis of system level RAMS and risk characteristics. Firstly, the system level RAMS characteristics, split up into a section on reliability, and safety and maintenance, are presented in section 12.1. Then, partially based on the earlier RAMS analysis, the system level risk is presented in section 12.2.

12.1. System RAMS

A first version of the RAMS analysis was made in the Midterm Report [3], which will now be expanded upon. Mission reliability will be covered first, followed by a system level FMEA and a listing of a preliminary maintenance schedule.

12.1.1. Mission Reliability

According to Larson and Wertz [121], mission reliability is defined as functioning without any failure that impairs mission performance. For the Lunar Habitat mission, three segments of performance have been defined in figure A.1, the transfer to the Moon (FF3), the installation on the Moon (FF4) and the operation on the Moon (FF5). The mission reliability for these three phases can be defined as follows:

$$R_{TTM} = R_{Launcher} \cdot R_{TV} \cdot R_{Lander} \tag{12.1}$$

$$R_{Installation} = R_{Rover} \tag{12.2}$$

$$R_{Operation} = 1 - (1 - R_{Sys}) \cdot (1 - R_{SCM}) \tag{12.3}$$

Both equation 12.1 and 12.2 are cases of series reliability, given that the entire process fails if one of the elements fail. In contrast, equation 12.3 is a case of parallel reliability, given that mission functionality can be covered by both the entire system and the SCM. The system reliability present in this equation can be further split up as follows:

$$R_{Sys} = R_{Power} \cdot R_{Atmos} \cdot R_{Comms} \cdot R_{CDH} \cdot R_{Thermal} \cdot R_{Radiation} \cdot R_{MMP} \cdot R_{Structures} \tag{12.4}$$

Based on the user requirements, U10 in particular, and the fact that this mission is a pioneering mission, a total operational reliability of 99% after 1 year is deemed acceptable. This means that assigning both the system and the SCM a reliability of 90% after one year is sufficient. However, a reliability of 95% is preferred, giving the team some room for errors.

Launcher Reliability

Launcher reliability can be estimated based on historical data. The Falcon 9/Heavy rocket series has performed 56 launches to date, out of which there was a single in-flight failure, and a single pre-flight failure. Given that both failures occurred during the first couple of years of operation (infant mortality), a decreasing failure rate is assumed. Considering this, a launch success of at least 99% is within the reasonable range. The reliability of the launches can then be estimated by equation 12.5.

$$R_{Launcher} = 0.99^{N_{Launch}} \tag{12.5}$$

Transfer Vehicle Reliability

No estimation of the Transfer Vehicle Reliability can be made based on historic data, but according to Hassan and Crossley [170], propulsion systems can have a reliability of up to 99.5%. As Larson and Wertz [121] also present many electric parts having higher reliability in space flight compared to launch (factors range from 10 to approximately 100), it is assumed that the transfer vehicle can be designed with a reliability of 99.9% by implementing multi-level redundancy. Another argument for the feasibility of such a high reliability is the fact that the mission duration for the transfer vehicle is relatively short. Dubos *et al.* [171] present typical satelite reliability, and find that for very short mission duration, reliability is close to 100%.

Lander Reliability

A landing vehicle needs more complex systems compared to a transfer vehicle in order to accurately perform its function. For this reason, extremely high reliability is something that seems unfeasible. Hence the reliability of the lander is assumed to be 99%; similar to a launcher.

Installation Reliability

Installation reliability is rather difficult to estimate because of the fact that all rovers are still in such a preliminary design stage. To keep overall mission reliability high, the rovers need to be designed with maximum reliability in mind.

Power System Reliability

The reliability of the power system can be described by equation 12.6. It follows a K-out-of-N scheme, with 3 out of the 8 sterling generators being required to deliver the emergency power levels of 7500 W.

$$R_{Power} = \sum_{i=3}^{8} \binom{8}{i} \cdot R_{Sterling}^{i} \cdot \left(1 - R_{Sterling}\right)^{8-i} \tag{12.6}$$

Plugging in the numbers, for a single sterling generator reliability ($R_{Sterling}$) of 99%, the total reliability of the power system is more than 99.9999%.

Atmospheric System Reliability

Given that the atmospheric system has one full back-up in the SCM, the total reliability for the system can be described by equation 12.7

$$R_{Atmos} = 1 - (1 - R_{ECLSS})^2 \tag{12.7}$$

Taking an ECLSS reliability of 99%, the total system reliability becomes 99.99%.

Communication System Reliability

The communication system also has one back-up, meaning there is once more a case of parallel reliability. The total reliability for the system can be found in equation 12.8

$$R_{Comms} = 1 - (1 - R_{Dish})^2 \tag{12.8}$$

Taking a reliability of 95% for a single dish, the total system reliability becomes 99.75%.

Command & Data Handling Reliability

The reliability of the CDH system can be approximated by equation 12.9. It is once more a case of parallel reliability with a full level of redundancy.

$$R_{CDH} = 1 - (1 - R_{CDH-System})^2 \tag{12.9}$$

Once more setting a single system reliability to 99%, the total system reliability becomes 99.99%.

Thermal Regulation System Reliability

There is no full redundancy in the thermal system, but it has redundant components. An estimation for the reliability of this system can once more be set at 99%.

Radiation Protection Reliability

The reliability of the radiation protection is extremely difficult to estimate properly. Given that it is a static system and is not really capable of causing failure of other systems except for design flaws, it makes sense to take it out of equation 12.4.

Meteoroid Protection Reliability

The reliability of the meteoroid protection system can be approximated by the PNP as described in section 7.2. This has a value of 0.998, which means the reliability of the system is approximately 99.8%.

Structural Reliability

The reliability of the habitat structure has to be high to ensure 10 years of operating life. Given a high maintainability, having a reliability of approximately 99% after a year is acceptable.

Total system

Plugging in all the suggestions and values found above, we find for the system reliability a value of approximately 97.5%. Assuming a similar reliability for the SCM, the operational reliability becomes approximately 99.9%.

However, the reliability of the installation process is considerably lower, given that 18 consecutive launches, a transfer and a landing lead to a total R_{TTM} of only 82.6%. However, a launch failure in this stage of the project would cause a higher end cost, but it does not cause complete mission failure.

12.1.2. System FMEA

Apart from all the subsystem level FMEA's, a system level FMEA was also made. Once more, severity is judged on a negligible to catastrophic scale as described in subsection 8.1.5. Occurence is based on electronic systems (score 1) and mechanical or biological systems (score 2). Lastly, determination is based on automatic detection (score 1), manual detection (score 2) or pre-testing (score 3). Some cases get a 0 determination score by design.

12.1.3. Safety

Following the risks identified in subsection 12.1.2, safety procedures and features are drawn up for the habitat operation.

12.1.4. Safety Features

- **Radiation sensors** - As described in section 8.2, a extensive set of radiation sensors will be implemented in the habitat to monitor for harmful radiation doses, on top of the radiation protection by water and regolith layers.

- **Fire extinguishers** - Just like aboard the ISS, a fire repression system is integrated into every module of the habitat, as described in figure 8.10. In the event of a fire emergency, the fire needs to be extinguished manually, by use of the fire extinguishers that will be part of the habitat equipment.

- **Fire ports** - Next to the fire extinguishers for fire repression, the habitat will also sport fire ports. These are ISS-derived ports that allow CO_2 to be funnelled behind the interior panels, where fire is likely to occur due to faulty wiring.

- **Emergency egress lighting system** - Another ISS-derived safety measure would be the Emergency Egress Lighting System (EELS), which illuminates the path towards safety equipment (or, in non-space applications, the exit) in case of a power outage.

- **Personal Breathing Devices** - In the case of a depressurisation or the occurrence of toxic fumes in the internal atmosphere, Personal Breathing Devices (PBDs) are present to provide safe air for breathing while the issue is being resolved.

- **Compartmentalisation** - As with the PBDs, the compartmentalisation is important in case of rapid depressurisation (usually due to puncturing of the hull). Due to compartmentalisation, the punctured compartment can be sealed from the rest of the habitat, so depressurisation will be contained to one module while the rest of the habitat will stay mostly unaffected.

- **Module Venting** - In case of an uncontainable fire, venting a module might be the only option left. Therefore, near the docking ports, blowout valves will be present. These can vent the module in a short time, extinguishing all fires present and expelling all toxic gases in the atmosphere. The ECLSS Atmosphere Management will recover the atmosphere after the blowout.

- **Return Vehicle** - A return vehicle will be present to transport the crew back to Earth at end of mission, but in case of a premature mission ending the return vehicle is also to be used as safe transport back to Earth. Therefore the return vehicle should be accessible, operable, and ready for take-off at all times.

- **SCM** - As described in chapter 6, one of the four habitation modules will be the SCM, which is a radiation-hardened, fully autonomous module that is designed to function even while all other modules are inoperable.

- **Medical Suite** - In the case of medical emergency, be it sickness or injury, treatment will be available in the medical suite fitted in the habitat. It will be fitted with some standard equipment so common injuries or conditions can be treated.

Function ID:	Function	Failure Mode	Potential Effect of failure	SEV	Potential causes of failure	OCC	Current Control	DET	RPN
FB_1.2	Navigate through space	Loss of navigation	No command of module	3	Loss of communication with ground segment	1	Automated checkups	1	3
			Wrong transfer or orbit	2	Pointing error in ADCS system	1	Automated checkups	1	2
			Loss of single module	3	No/Too short burn	2	Test on Earth	3	18
			Loss of single module	3	Debris strike in space	1	Automated checkups	1	3
			Loss of single module	3	Power loss in flight	1	Automated checkups	1	3
FB_1.3	Perform landing operations	Loss of landing functionality	Crash landing	3	Power loss in flight	1	Automated checkups	1	3
					Landing system does not deploy	2	Automated checkups	1	0
					No/Too short burn	2	Test on Earth	3	18
					Collision with object on lunar surface	1	Test before landing	3	9
FB_2.0	Install habitat	Loss of installation functionality	Slower installation	2	Rover breakdown	2	Automated checkups	1	4
			Module sinkage	4	Paved regolith cannot bear loads	2	Test on Earth	3	24
			Non-complete protection	3	Sintering damages module	2	Test on Earth	3	18
			Loss of single module	3	Docking/Connect-ion failure	2	Automated checkups	1	6
FB_3.1	Supply atmosphere	Loss of atmosphere	Loss of single module	3	Leak in module	1	Automated checkups	1	3
			Loss of habitat	4	Loss of composition regulation	1	Automated checkups	1	4
			Loss of habitat	4	Airlock explosive decompression	1	Automated checkups	1	4
			Loss of single module	3	Leak in pressure regulation system	1	Automated checkups	1	3
			Loss of habitat	4	Contamination of atmoshpere	2	Automated checkups	1	8
FB_3.2	Provide healthcare	No healthcare	No help from Eath doctors	3	Loss of communication	1	Automated checkups	1	3
			Loss of life	4	No emergency medical supplies left	1	Regular checkups	2	8
			Loss of life	4	No drugs left	1	Regular checkups	2	8
			Decreased astronaut functioning	3	Breakdown of exercise system	2	Automated checkups	1	6
			Loss of life	4	Defect medical equipment	1	Automated checkups	1	4
FB_3.3	Supply provisions	No provisions	Immediate mission end	4	Water supply runs out	1	Automated checkups	1	4
			Immediate resupply required	3	Food supply runs out	1	Regular checkups	2	6
			Decreased astronaut functioning	3	Supplement supply runs out	1	Regular checkups	2	6
			Loss of life	4	Contaminant in provisions	2	Automated checkups	1	8
			Decreased astronaut functioning	3	Provision quality insufficient	1	Regular checkups	2	6
			More water needed	2	Water recycling system fails	1	Automated checkups	1	2

Figure 12.1: FMEA of the full habitat system, part 1

FB_3.6	Provide temperature control	No temperature control	Loss of single module	3	Breakdown of heatflow regulation	2	Automated checkups	1	6
			Loss of single module	3	Temperature sensors broken	1	Automated checkups	1	3
			Complete loss of base	4	Power loss	1	Automated checkups	1	4
			Higher temp. in base	1	Solar weather causes extra heatflow	1	Regular checkups	2	2
FB_3.7	Provide radiation protection	No radiation protection	Slightly higher interior dose	1	Solar weather causes excess radiation	2	Regular checkups	2	4
			Immediate mission end	4	Regolith destroyed by large meteoroid	1	Automated checkups	1	4
			Premature mission end	2	Extra EVA activity required	1	Automated checkups	1	2
FB_3.11.1	Provide emergency crew return vehicle	No possibility of crew return	Complete loss of life	4	Return vehicle transportation failure	1	Regular checkups	2	8
			Loss of life	4	Emergency escape route blocked	1	Regular emergency exercises	1	4
FB_3.11.2	Provide compartme-ntalisation	No habitat compartment-alisation	Loss of single module	3	Single compartment door failure	2	Automated checkups	1	6
			Complete loss of habitat	4	Power loss	1	Automated checkups	1	4
FB_3.11.3	Provide fire suppression	No fire suppression	Loss of single module	3	Fire detection failure	1	Automated checkups	1	3
			Complete loss of base	4	Not enough firefighting supplies	1	Automated checkups	1	4
FB_3.13	Supply power	No power supply	Complete loss of base	4	Nuclear meltdown	1	Reactor inherently stable	0	0
			No emergency power	3	Power storage empty	1	Automated checkups	1	3
			Loss of single module	3	Power distribution failure	1	Automated checkups	1	3
FB_3.14	Provide communication	No communication	Premature mission end	3	Antenna failure	1	Automated checkups	1	3
			Start of emergency procedure	3	Ground support failure	1	Automated checkups	1	3
			Manual antenna adjustment	1	Antenna unsufficient pointing accuracy	1	Automated checkups	1	1
			Manual antenna adjustment	1	Line of sight obstruction	1	Regular checkups	2	2
			Premature mission end	3	Power loss	1	Automated checkups	1	3
FB_3.16	Provide maintenance capability	No maintenance	Premature mission end	3	Maintenance tools break down	2	Automated checkups		0
			Premature mission end	3	Spare parts run out	1	Regular checkups		0
FB_3.18	Provide impact protection	No protection against impacts	Slow degradation of protection	2	Damage detection system fails	1	Automated checkups	1	2
			Loss of single module	3	HE-MM strike	1	Extremely low probability	0	0
			Loss of single module	3	MM strike	1	Sufficient protection	0	0
FB_3.19	Operate CDH	No CDH	Complete loss of base	4	Power Loss	1	Automated checkups	1	4
			Premature mission end	3	CDH fast degradation	2	Automated checkups	1	6
			Loss of redundancy	2	Central Server damaged	1	Automated checkups	1	2

Figure 12.2: FMEA of the full habitat system, part 2

Safety Procedures

These can be found in subsection 12.1.3, where moderate, critical and catastrophic failures are defined as in subsection 12.1.2

Table 12.1: Safety procedures for emergency events

Events		Failure procedure	
	Moderate	**Critical**	**Catastrophic**
CDH failure Power Failure Comms Failure	Repair Damage	Evacuate to SCM module, repair damage	Evacuate the base with return vehicle
Solar Flare Meteoroid Impact Radiation Penetration	Evacuate to SCM	Evacuate to SCM, wait till emergency passes and repair damage	Evacuate the base with return vehicle
Depressurization Explosion	Seal off affected module, move affected crew members to med bay, repair damage from inside using EVA suits	Seal off affected modules, evacuate to SCM, tend to woundedcrew members, repair damage	Asses damage and mission continuity, evacuate the base if neccessary
Medical Emergency	Tend to crew member using medicalsupplies	Sterilize med bay and place wounded crew member in an isolated environment	Evacuate the base with return vehicle
Fire in Modules	Crew puts on breathing device, Suppress the fire with fire extinguishers, cutthe ventilation to module, restore atmosphere	Crew puts on breathing device,seal off the affected compartment, vent the compartment, restore atmosphere and repair damage	Evacuate the base and vent all modules
Contaminated Atmosphere	Crew puts on breathing device, Seal off contaminated module, recover air quality with ECLSS system	Crew puts on breathing device,seal off contaminated modules, vent atmosphere	Evacuate the base and vent all modules

12.2. Risk Assessment

In this section a risk identification is performed. Most risks are taken from the Midterm report [3], but some new ones are added based on the detailed FMEA's performed in the RAMS analysis.

12.2.1. Risk Identification

The risks written in bold are new risks which are identified during the progress of designing the final concept. The underlined risks are updated versions of similar risks from the midterm report. The risks which are struck through have been eliminated due to design choices.

1. Transport product to Lunar building site:
   (a) Wrong/absent parts in transport vehicle to launch site;
   (b) Delay of transport vehicle to launch site;
   (c) Accident of transport vehicle;
   (d) Elements do not comply with launch requirements (size/mass);
   (e) Rocket failure;
   (f) Wrong ΔV in TLI;
   (g) Transfer vehicle booster failure;
   (h) Non-detectable debris strike;
   (i) Detectable debris strike;
   (j) Subsystem failure during transport to Moon;
   (k) Re-ignition failure;
   (l) Unforeseen geographical conditions on landing site;
   (m) Wrong landing spot;
   (n) Damaged payload impacting the mission.
2. Install product:
   (a) Impassable terrain on planned route to building site;
   (b) Performance 'ant' rovers not as required;
   (c) Payload elements not arrived;
   (d) **Sintering process damages underlying module**;
   (e) Module assembly mechanism fails;
   (f) Building site surface characteristics different from expectation;
   (g) Structural integrity does not comply with requirements;

(h) Surrounded by impassable terrain on landing site;
(i) **Regolith rover breakdown**;
(j) **'Ant' rover breakdown**;
(k) **Paved regolith does not meet expected material properties**;

3. Perform pre-operational procedures:
   (a) ~~Solar panels do not deploy~~;
   (b) Insufficient power supply for pre-operational procedures;
   (c) No communication possible before operational phase;
   (d) Leakage in habitat;
   (e) Non-fixable leakage in habitat;
4. Operational phase:
   (a) Emergency before escape route/ safety shelter is finished;
   (b) Non-repairable damages to the habitat or other systems;
   (c) Components for repairs not available;
   (d) Emergency escape vehicle is broken;
   (e) Fire;
   (f) Toxic spill;
   (g) Too high radiation levels in habitat;
   (h) Too high radiation levels in space suit;
   (i) Detectable meteoroid/debris strike;
   (j) Non-detectable meteoroid or debris strike;
   (k) Consumables run out;
   (l) Habitat environmental contamination;
   (m) Habitat noise contamination;
   (n) No health monitoring available;
   (o) Unforeseen increased radiation;
5. Perform post operation procedure:
   (a) Habitat is not (correctly) sterilised;
   (b) Waste is not correctly disposed.
6. EOL:
   (a) Not all organic substances collected;
   (b) Not all collected organic substances fit in return vehicle;
   (c) Something going wrong with the return to Earth;
   (d) **Nuclear core disposal issues**.
7. Astronauts' risks:
   (a) Astronaut is unable to perform mission for personal reasons;
   (b) Astronauts become sick in the first few days of mission;
   (c) Physical health issues during mission;
   (d) Mental health problems of astronauts during mission;
   (e) Human errors resulting in subsystem failure;
   (f) Astronaut becomes pregnant during mission;
   (g) Astronaut violates a law;
   (h) Decompression sickness.
8. Subsystem risks (pre-operational / operational phase) :
   (a) Pressure sensor failure;
   (b) Pressure regulation failure;
   (c) Atmospheric composition regulation failure;
   (d) Nuclear power system failure;
   (e) Power storage failure;
   (f) Power distribution failure;
   (g) Temperature sensor failure;
   (h) Temperature regulation failure;
   (i) Communication failure to Earth;
   (j) Communication failure between astronauts;
   (k) Ground support failure;
   (l) Fire detection system failure;
   (m) Insufficient / malfunctioning firefighting supplies available;
   (n) Debris detection failure;
   (o) CDH System failure;
   (p) **Radiation detection system failure**.

12.2.2. Risk Map

The likelihood of occurrence and the mission impact of each risk can be found in table 12.2 and 12.3. In the midterm report a mitigation plan was set up for the most critical risks. During the design of the final concept, this mitigation plan is implemented reducing the severity of the risks. In the risk map, all risks in italic moved places due to the implementation of the mitigation plan. All risks in bold are newly identified risks. These will remain in their original position and a mitigation plan will be set of for these risks in subsection 12.2.3.

Table 12.2: Risk map overview for non-technical failures

Probability	**Effect**			
	Negligible	**Marginal**	**Critical**	**Catastrophic**
Very likely				
Likely	1b			
Somewhat likely		7g, 1h, *4j*	4c, 7a, 7c, 7d,	1i, **2K**
Unlikely		1a, 2a, 5a, *7h,4a*	1c, 1n, 5b, 6a, 7b, 4f, 4h, 4l, *4e,4i*, **2d**	
Very unlikely		**6d**	1d, 1f, 1l, 2c, 2f, 6b, 7e, 4n,4o,*4g,4m,1d*	*2h*, 7f, 4k

Table 12.3: Risk map overview for technical failures

Probability	**Effect**			
	Negligible	**Marginal**	**Critical**	**Catastrophic**
Feasible in theory		**2i,2j**		
Working in laboratory model				
Based on existing non flight				4d
Extrapolated from existing design			1g, 1m, 1k, 2b, 2e, 3b, 3d, 3e, *4b*	3c, *1j*, *6c*
Proven flight design			8k, *8a, 8b, 8g, 8h, 8i, 8j*,**8p**	1e, 8n, *8d, 8e, 8f 8l, 8m, 8o, 8c, 2g*

As can be seen in the tables, all subsystem risks are moved to 'proven flight design' because of the V&V plans implemented in chapter 8. Also, the impact on the mission of a meteoroid strike and a fire has become less, since these are designed for now. The same holds for the likelihood of too high radiation levels, a structural failure and impassible terrain has dropped since more research has been done at this stage.

12.2.3. Risk Mitigation

There are three newly identified risks which need a mitigation plan. All earlier identified risks have already been mitigated, or a mitigation plan is already developed in earlier stages of the design process.

Risk 2i and 2j

The risks for the rovers used for installing the habitat are partly mitigated since these are fully redunant. Therefore the impact on the mission of one failing is marginal. Since these rovers do not exist yet, they are only feasible in theory. In order to decrease the likelihood of occurrence the rovers need to be build and tested thoroughly for performance and structural integrity. This process will shift these risks to 'working in laboratory model'.

Risk 2k

When the paved regolith does not meet the required properties, this can be catastrophic for the mission. In order to mitigate this risk, the likelihood can be mitigated by paving and testing a lunar regolith simulant, or actual lunar regolith. When the actual properties are known, the risk will move to 'unlikely' or even 'very unlikely'.

12.3. Concluding Remarks

Clearly, the project requires a high level of reliability to deal with the many risks present in a mission to the Moon. Especially the logistics of installation present a critical case that can only be dealt with by performing extensive design work on the conceptual rovers and the idea of regolith sintering. Given that these topics are adequately dealt with, the project risk is determined to be of such a level to justify project continuation.

13
Future Development

This chapter discusses the future development required for the Lunar habitat to become a reality. This chapter will highlight all those steps required to get the design to the next design phase. Section 13.1 describes the future development of systems, whereas section 13.2 describes future development related to the habitat itself and element integration. section 13.3 elaborates upon the sustainability approach of the project. Section 13.4 describes the continuation of the project in the next phases, in this section the sustainable development strategy, including the end of life procedures are elaborated upon. All future development will be visually represented in a Gantt chart, presented in section 13.5.

13.1. Future Development in System Design

13.1.1. Partnership

The nuclear power system, the alternative water processor, the ACLS, the environmental monitoring system and the relative navigation technique are currently being developed by potential partners. Collaborations can be set up with the respective partners, given the results of the development are as expected. If this is not the case, other systems have to be chosen and developed. Furthermore, when development is done, the systems need to be integrated with other systems and within the habitat. Integration tests need to be performed to ensure all systems work as conjoint as expected.

13.1.2. Food

As explained in subsection 8.3.4, two Advanced Plant Habitats will be installed in the Lunar habitat. In the first year of habitation, tests will be performed on the functioning of the APH. If the APH performs as well as expected, the possibility of expanding the habitat with a growth chamber shall be considered. As some research is being done on the subject by potential partners, it is expected that a collaboration could be set up, which would allow for much quicker development. However, thorough calculations have to be performed to determine if the growing chamber would prove to be beneficial in saving costs. Furthermore, a collaboration could be set up where the growing chamber of the Lunar habitat would serve as a test bed for planned manned missions to Mars, making it a marketing opportunity as well.

13.1.3. Medical Equipment

As not all required medical equipment that is needed is currently developed, part of the budget need to be set aside to allow further development. It is found to be especially important to invest in the design of the clean room, as this will decrease the chance of getting an infection during surgery.

13.1.4. Rovers and Return Vehicle

The ATV, the Ant and the Regolith Rover are all still very conceptual. For each of these rover types, a complete development project is to be set up. The astronaut transfer vehicle can be inspired on Apollo's Lunar Rover Vehicle, but is required to be an upgrade in terms of performance. It should be able to drive further and at higher speeds, while carrying more scientific equipment. Based on the type of research that needs to be performed, this scientific equipment has to be designed accordingly. Therefore, the expected mission outcomes need to be refined first. Once all requirements and goals are set up, research can be done into potential partners to develop the ATV and a development plan can be set up.

As the Ant and the regolith rover currently have a TRL level of 1, much more development is needed. Studies need to be done on the feasibility of the current designs, other, more extensive, designs have to be investigated as well. The Ant also needs to be able to carry the nuclear power plant for example, as for the current design this would not be feasible. The power supply for both the Ant as well as the RR has to be designed completely as no concrete information, as of yet, is available on power usage for both systems. Together with the power supply and the algorithms needed for co-operation, the regolith sintering mechanism is expected to be the largest research and development project of the regolith rover, being of vital importance for the success of the Lunar habitat. All these systems have to be tested intensively on Earth, to make sure the systems have the highest possible reliability. Lastly, the return vehicle needs further development as is explained in section 9.3. A lot of partnership possibilities arise from these rovers as expertise and funds are needed to make sure the rovers function as desired.

13.2. Future Development in Habitat Design

13.2.1. Safety Module

Currently, the structural design of the safety module is the same as any other module. This means, it is not better protected in case of severe solar flares or meteoroid impacts, whilst the team is convinced it is important to do so. Therefore it would be advised to do this simultaneously with the iteration on structural design, as will be performed in the next phase of the project.

13.2.2. Docking Node, Docking Mechanism and Airlock

The designs of the docking node, the docking/interconnection mechanism and the airlock are all based on existing and tested designs. The adjustments made to the docking node and docking mechanism are mainly in terms of sizes, meaning most feasibility research is already performed and the research and development will mainly consist of designing and system testing. Although the airlock is based on existing designs, a Lunar dust removing system needs to be developed as well as the suitports. As it was decided to use either an advanced version of the Z-2 spacesuit or a newly developed Z-3 spacesuit, a collaboration with NASA has to be set up [41]. Designing the Lunar dust shield will be outsourced to ensure the right level of expertise and reliability.

13.2.3. Expandability

Before the final concept was chosen, several concept designs were designed and considered, each having their own strengths and weaknesses. An attractive option was using inflatable domes. In the future it should be investigated if the inclusion of domes would be beneficial to the current design. Ideas could be found by looking at BEAM, an inflatable dome currently being tested on the ISS. To allow for the adding of a dome-like element, every module includes one extra door from which could be expanded. Moreover, a garage is to be designed and added where the rovers can be stored. This would highly decrease the wear due to the abrasive nature of the Lunar regolith.

13.2.4. KRUSTY Information Campaign

As explained in subsection 2.4.2 and subsection 8.6.1, involving a nuclear reactor is a choice that was not easily made. In order to ensure sufficient public and corporate support, an extensive information campaign will be set up [16]. This campaign will be focused on elaborating upon the importance of the mission itself as well as the inclusion of the KRUSTY system. The importance of the mission will be clarified by stating that it will push the boundaries of human technology and advancement, practising the survival of human beings in space, more specifically on celestial bodies and performing advanced scientific research on a numerous amount of subjects. Furthermore, space missions are known for pushing technological development which may also be used on Earth. These new technologies will result in economic growth, and will open up a lot of collaboration possibilities with international partners [7]. It is expected that this will not only help Europe keep its global position in technological advancement, but will also make sure Europe becomes a leader in space exploration. As for KRUSTY, it will be thoroughly explained to the public that it is a newly developed system powering the next generation of long duration, high power space missions. This power system is vital in successfully performing the mission, as solar power is not a solution due to the suncycle. On top of that, KRUSTY is safer to launch than the currently used RTG's. An exploding rocket will spread less nuclear material from a dormant nuclear fission reactor as U^{238} is not a fissile material on its own opposed to Pu^{238}, used in RTG's. The campaign for KRUSTY will be called: "KRUSTY: The Little Reactor that Could" [172] a tangible, positive name slogan. A similar amount of information will be given on other updated or newly developed systems, such as certain parts of the ECLSS. The campaign will be accompanied by various press releases as are standard for this type of mission. Furthermore, by means of social media the importance of the Lunar habitat mission will be further explained, and regular updates will be given on facts which are interesting to the public. As the public outcry against nuclear power has significantly decreased over the past decades [16] it is expected that this information campaign will be sufficient.

Should this not be the case, a more comprehensive information campaign will be set up. This campaign will include a more detailed description of the importance of the project itself and the critical role KRUSTY plays in it. Additionally an elaborate discussion will be included on the nuclear power systems that are used on Earth with a special emphasis on why accidents happen and why it differs so much with KRUSTY.

According to Downey et al. [15], to convince the risk adverse public, the reasons to use nuclear technology must be compelling. To ensure a proper transfer and wording of the explanation on the KRUSTY to public, experts on marketing need to be employed. Since the cost of the information campaign is assumed to be significant, a budget of €19 million is allocated. Simultaneously, the market analysis needs scheduled re-evaluations to check if the information campaign has the desired results on the public opinion. Furthermore, the market in spaceflight seems to be changing more rapidly than historically has been the case, due to the private companies entering, which is opening up the market.

13.3. Sustainability

The mission goal to make a contribution to the long term survival of humankind shall continue to be the focus point throughout the whole mission. This is the first and most important step in achieving social sustainability. To aid this goal, it is important that information, derived from the mission, will be shared with the scientific community, but only upon agreement not to use it for commercial purposes. Furthermore, for the duration of the mission a complete and thorough update on the mission will be given every 30 days to everyone interested. As the mission is government funded and uses resources that belong to humankind, it is important to give back to the society. Therefore as part of future development, an information and educational campaign will be set up. Part of which will be already included in the KRUSTY information campaign. According to the ESA sustainability guidelines [173], the last main point to consider is environmental sustainability. The environmental sustainability of mission is difficult to asses at this point in time. It is however clear that environmental sustainability has been a minor consideration giving the complexity of the mission. Finally, the end of life sustainability is discussed in section 13.4.

13.4. Future Project Phases

Based on the European cooperation of space standardisation's project management guideline [174], the Lunar habitat is currently at the end of phase A development and already covered some aspects from phase B. Where phase 0 includes the mission analysis and need identification based on the customer requirements. Phase A is mainly focused on the feasibility of the project, where a detailed project planning is made and the requirement list is refined. Also the feasibility of technical elements, cost estimation, risk and interfaces of all elements are evaluated. The continuation of the project will consist of phases B till F, which are discussed below:

Phase B: Preliminary definition of the project and product
The product and project will be further defined. Technical solutions will be presented for the selected concepts (subsystems), accompanying an explanation on how requirements will be met. A start will be made on the disposal plan and operational handbook, after which the first business agreements are set. The development as described in section 13.1 and section 13.2 shall mainly be performed in this phase. During phase B, the System Requirements review will be held, concluding phase B will be the Preliminary design.

Phase C: Detailed definition of the product
The design will be designed into full detail, including the production, testing set up, and product verification and validation. Every subsystem shall be designed to the greatest extend, in a way that every subsystem is ready for production. Furthermore, the design integration shall also be performed. At the end of phase C, the critical design review will take place. Among other things, the design justification file together with the verification plans shall be reviewed.

Phase D: Production and ground qualification testing
All parts, systems and subsystem needed for the Lunar habitat will be produced and all ground assembly procedures will be followed. To ensure the product is properly assembled, ground qualification and verification shall be performed. The ground qualification will consist out of assembling a prototype system somewhere in a harsh environment on Earth [175, 176]. Building up the system shall be done sequentially, integrating one subsystem at a time. This allows for finding systems (or combinations of subsystems) that are not performing nominally. Careful thought should be applied when the inclusion of KRUSTY is taken into consideration as this could lead to protest.
During phase D, qualification review will take place to ensure all delivered products are of the previously discussed level of quality and the ground elements function as expected. At the end of phase D. the acceptance review will take place.

Phase E: Utilisation
In the first part of phase E, the operations of the habitat will be validated during a 1 year mock mission on Earth, as well as validating EOL considerations. During the mock mission stress tests will be conducted on the habitat and the astronauts to determine the operational envelope and psychological impacts. The

second part of the utilisation phase consists of the utilisation itself. During this phase, request for improvement can be made, which can be implemented before the actual mission starts. Furthermore, the operation and maintenance plan shall be implemented. Moreover, the disposal plan shall be finalised. During phase E the flight, operational and the launch readiness review shall be performed before each launch. Also a flight qualification review and the in-flight acceptance review are planned to take place.

Phase F: Disposal and end of life sustainability
This phase comprises the end of life and final disposal. After ten years of operation, the disposal plan of the Lunar habitat shall be updated, after which it shall be executed. During this phase end of life sustainability is of vital importance. The main points to focus on are the removal of organic substances to avoid contamination of the Moon and the habitat, and a safe disposal of the nuclear core used in KRUSTY. To ensure the habitat is completely clean at end of life, all surfaces in the habitat will be cleaned, all air filters replaced and the last measurements of all sensors will be checked thoroughly. As explained in figure 8.3.4, these devices can not directly measure biological growth. To scan the surfaces of the habitat for the presence of biological substances, the same scanner as is used in hospitals will be utilised [93]. The process of measuring, scanning and cleaning will be iterated until all measurements are within acceptable bounds. Biological material, including human waste and plant remains will be brought back to Earth. As the customer deemed it acceptable to leave the habitat on the Moon, the habitat including all subsystems will be left on the Moon. All the systems and subsystems will be put in hibernation mode and decoupled, all the moveable systems will be placed at their respective designated spot. The rovers shall be parked in the then available garage.

The end of life procedures regarding the KRUSTY are not set up yet by NASA, as more detailed research needs to be done. However to even be able to perform the end of life procedures, much more development is needed on the rovers to asses their capability on handling the nuclear core. During the next design phase, a detailed end of life procedure for the KRUSTY will be set up.

13.5. Project Gantt Chart

The project gantt chart can be seen in figure 13.1. According to Larson and Wertz [121], a typical mission design for a complex mission is expected to take between 10 and 15 years. This is the time from the feasibility study onwards, till the point where the habitat is considered operational. This estimate seems to be reasonable as MIR, skylab and the ISS took between 10 and 16 years to become fully operational [177–179]. Furthermore, a more detailed description on the installation of the habitat on the Moon can be found in chapter 11.

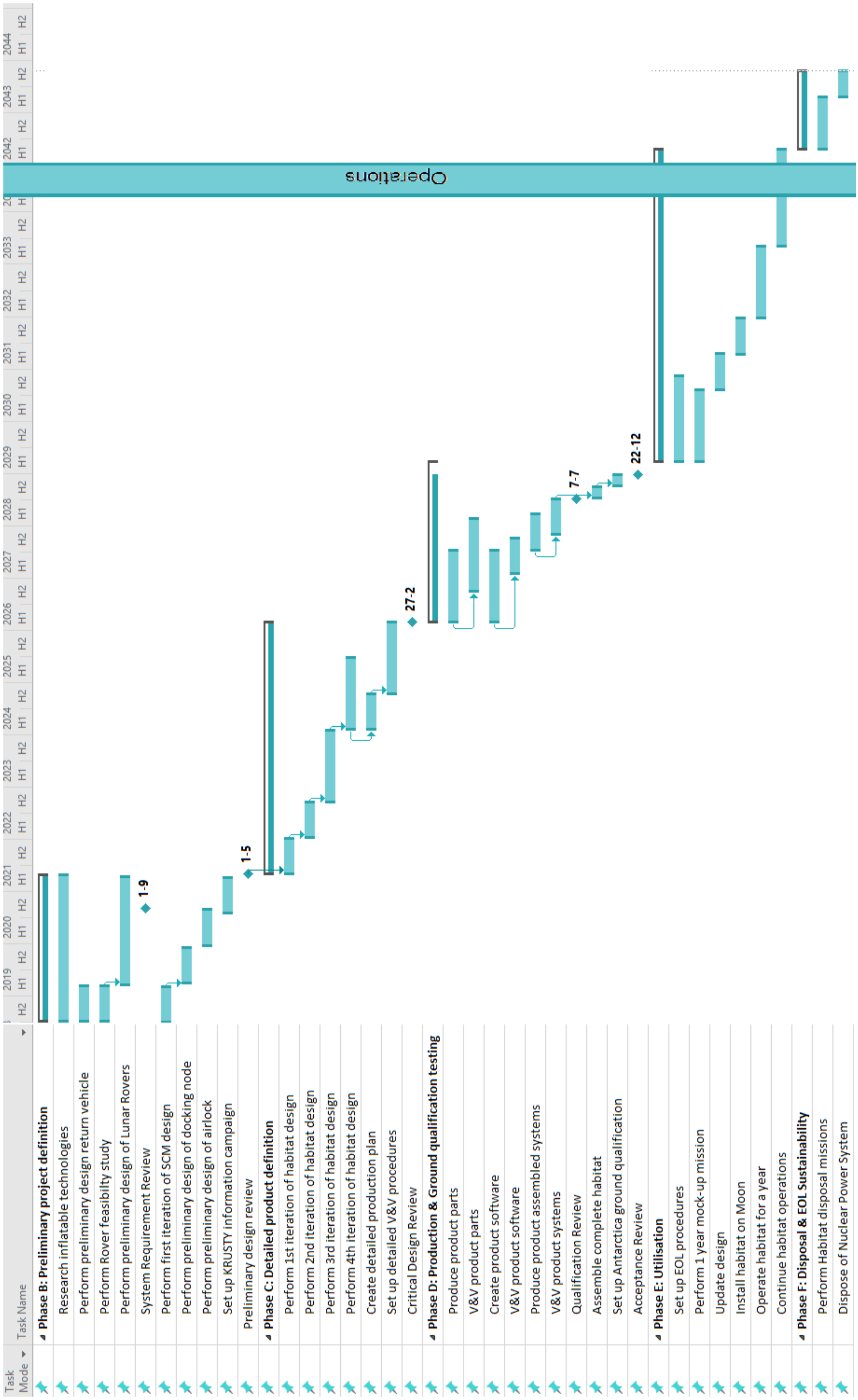


Figure 13.1: Project Gantt Chart

14
Complete Mission Cost

As described in NASA's Cost Estimation Handbook [180], there are three ways of estimating project costs. These cost estimation methods correspond to the design level of the product, with the first being the most conceptual estimation method and the last being the most detailed (but also accurate) cost estimation method. These methods are:

- Analogous cost estimation;
- Parametric cost estimation;
- Engineering Build-Up cost estimation.

In analogous cost estimation, a similar product or mission is taken, and minor adjustments in cost the estimation are made to accommodate the specific nature of the new product or mission. This kind of estimation method is only used to define a financial 'ball-park', in which the budget is supposed to be determined.

In parametric cost estimation, a large dataset of similar products is analysed for correlation between a certain parameter (usually mass) and cost. This kind of cost estimation is usually more accurate than analogous methods, but rather large contingency factors are still associated with parametric cost estimations.

The engineering build-up cost estimation method uses the WBS for the project. For this type of cost estimation, the WBS has to be defined in such detail, that the work packages contain only several man-hours' worth of work. The cost of all work packages is then estimated and summed to arrive at the total mission cost. The advantage of this type of cost estimation is that it is very accurate, but it requires a rather high progression level of the mission design.

14.1. Cost Estimation

The chosen method will be a combination of methods similar to the engineering build-up cost estimation and the analogous cost estimation methods. Within the project, enough detail is present to be able to estimate the costs of different subsystems. This will be done using the analogous cost estimation method by looking at similar systems or operations, and subsequently the total cost estimation will be a build-up of the costs of the separate subsystems and operations. The cost estimations of the different subsystems and other costly operations are shown below in table 14.1.

The Environment Control and Life Support, Communication, Meteoroid Protection, Radiation Protection, Power, Command Data Handling and Computer System costs are already explained in their respective chapters. The Rovers, Return Vehicle and Launch costs are based on the costs given in chapter 9 and multiplied by the amount needed for the mission. The costs of the structure is based on the assumption that the raw materials only account for 1% of the total costs, and it has to be multiplied by a factor ten to get to the space grade costs. The cost of one suitport suit was found to be € 4.4 million [181], and eight suits will be present in the habitat, one for each crew member, and one redundant one for each crew member. For both the airlock and the connection node, their ISS counterparts, namely the Quest Joint Airlock and the Tranquility node were used for the cost estimation. For PR and marketing, it was assumed that the equivalent of one year's worth of NASA's public relations and marketing budget will be spent over the course of the mission for promotion. As a full product validation test, a mock-up of the final habitat design will be built on Earth and tested in the extreme environment of Antarctica. The price is derived from the cost of similar research stations on Antarctica. Lastly, the transfer vehicle budget was estimated from the development costs of the Apollo transfer vehicle, divided by two to account for the research already done in the Apollo project. This number was also corrected for inflation.

Because of the design maturity combined with the scope and complexity of the project at this point, it is impossible to obtain a reliable cost estimate. Also the comparable programs for this mission, namely the Apollo Program and the International Space Station Program had expenditures which were much higher than the almost €20bn that is determined for this mission. Taking into account all these factors and the maximum cost requirement, a contingency of 50% of the current best estimate was determined by judgement [182] to find the expected costs at about €30bn, leaving a margin of €13.9bn of the maximum allowable cost.

Table 14.1: Total mission cost estimation

System		**Estimation**
Environmental control and life support system	€	5,100,000,000
Communications system	€	50,000
Meteoroid protection system	€	1,000,000
Radiation protection system	€	22,600,000
Power system	€	21,000,000
Command & data handling and computer system	€	849,500,000
Structures	€	14,000,000
Rovers and Return Vehicle	€	11,600,280,000
Launching	€	1,368,000,000
Suitports	€	35,200,000
Airlock	€	164,000,000
Connection node	€	409,000,000
Product validation	€	30,000,000
PR and marketing	€	19,000,000
Transfer Vehicle	€	254,000,000
Total	€	19,868,630,000
Contingency	%	50
Total with contingency	€	29,802,945,000
Cost/kg	€	341,350
Maximum allowable cost	€	43,654,500,000
Margin	€	13,851,555,000

15
Requirements Compliance

In this chapter, the compliance matrix of all the requirements is presented in table 15.1. If a requirement has not been fully complied to, the rationale why the design does not meet the requirement, and the required modifications are explained in the feasibility analysis section 15.2. The list of requirements is stated in B.

15.1. Compliance Matrix

This section describes the requirement compliance. The requirement compliance matrix can be seen in table 15.1.

Table 15.1: Compliance matrix for the requirements of the mission

Code	Compliance	Required value	Achieved value
		Users	
U-1	✓	4	4
U-2	✓	Yes	Yes
U-3	✓	Yes	Yes
U-4	✓	Yes	Yes
U-5	✓	Yes	Yes
U-6	✓	Yes	Yes
U-7	✓	Yes	Yes
U-8	✓	Yes	Yes
U-9	✓	Yes	Yes
U-10	≈	Yes	Partially
U-11	✓	Yes	Yes
U-12	✓	Yes	Yes
U-13	✓	Yes	Yes
U-14	✓	Yes	Yes
U-15	✓	Yes	Yes
U-16	✓	Yes	Yes
U-17	✓	< 500000 €$/kg$	341350 €$/kg$
U-18	✓	Yes	Yes
U-19	✓	Yes	Yes
		Pre operations - Transfer to the Moon	
SYS-PRE-TRN-1	✓	$<11.9\ t$	$<11.9\ t$
SYS-PRE-TRN-2	×	>4.5g	future development
SYS-PRE-TRN-3	×	>|2g|	future development
SYS-PRE-TRN-6	×	<4.5g	future development
SYS-PRE-TRN-10	✓	$> 3.3\ km/s$	$> 3.3\ km/s$
SYS-PRE-TRN-13	✓	$<50m$	$40m$
SYS-PRE-TRN-17	✓	Yes	Yes
SYS-PRE-TRN-18	✓	Yes	Yes
SYS-PRE-TRN-19	✓	Yes	Yes
		Pre operations - Installation	
SYS-PRE-INS-1	✓	Yes	Yes
SYS-PRE-INS-3	✓	Yes	Yes

	Safety - Meteoroids		
SYS-OP-SFT-M-1	✓	21300 J	Protects for 6.86 MJ
SYS-OP-SFT-M-2	✓	Yes	Yes
SYS-OP-SFT-M-3	✓	Yes	Yes
SYS-OP-SFT-M-4	≈	≥300 s	≈ 1.5 h
SYS-OP-SFT-M-6	✓	Yes	Yes
SYS-OP-SFT-M-7	✓	Yes	Yes
SYS-OP-SFT-M-8	✓	> 0.98	0.9993
	Safety - Radiation		
SYS-OP-SFT-R-1	✓	165 mGy	78.3 mGy
SYS-OP-SFT-R-2	✓	5 $mGy/year$	2.1 mGy
	Life support - Internal Atmosphere		
SYS-OP-LS-IA-1	✓	No	No
SYS-OP-LS-IA-2	✓	69 < kPa	101 kPa
SYS-OP-LS-IA-3	✓	< 102.4 kPa	101 kPa
SYS-OP-LS-IA-4	✓	40	881kg
SYS-OP-LS-IA-5	✓	Yes	Yes
SYS-OP-LS-IA-6	✓	> 20 % & < 40%	21 %
SYS-OP-LS-IA-7	✓	0	0
SYS-OP-LS-IA-8	✓	40 < & < 70	40 < & <70
SYS-OP-LS-IA-9	✓	Yes	Yes
SYS-OP-LS-IA-12	✓	Yes	Yes
SYS-OP-LS-IA-15	✓	Yes	Yes
	Life support - Water		
SYS-OP-LS-W-1	✓	668 < l	2684 l
SYS-OP-LS-W-3	✓	Yes	Yes
SYS-OP-LS-W-4	✓	34.1 < l	34.2 l
SYS-OP-LS-W-5	✓	Yes	Yes
SYS-OP-LS-W-6	✓	< 0.5 l	0.5 l
SYS-OP-LS-W-7	✓	< 1%	0.1%
SYS-OP-LS-W-9	✓	Yes	Yes
SYS-OP-LS-W-10	✓	Yes	Yes
SYS-OP-LS-W-11	✓	Yes	Yes
SYS-OP-LS-W-12	✓	Yes	Yes
	Life Support - Food		
SYS-OP-LS-F-1	✓	Yes	Yes
SYS-OP-LS-F-2	✓	Yes	Yes
SYS-OP-LS-F-3	✓	3000 < $kcal$	3000 $kcal$
SYS-OP-LS-F-4	✓	Yes	Yes
SYS-OP-LS-F-5	✓	Yes	Yes
SYS-OP-LS-F-6	✓	50%	50%
	Life Support - Waste Management		
SYS-OP-LF-WST-1	✓	No	No
SYS-OP-LF-WST-2	✓	No	No
SYS-OP-LF-WST-3	✓	Yes	Yes
SYS-OP-LF-WST-4	✓	Yes	Yes
	Life Support - Pressure		
SYS-OP-SFT-P-2	✓	Yes	Yes
SYS-OP-SFT-P-4	✓	10 < m^3	11 m^3

SYS-OP-SFT-P-5	✓	Yes	Yes
SYS-OP-SFT-P-6	✓	1 Pa	1 Pa
Life Support - Thermal Control			
SYS-OP-LS-T-1	✓	Yes	Yes
SYS-OP-LS-T-2	✓	Yes	Yes
SYS-OP-LS-T-3	✓	< 1 K	0.1 K
Life Support - Comfort			
SYS-OP-LS-CF-16	✓	< NC-50	< NC-50
SYS-OP-LS-CF-17	✓	< NC-40	< NC-40
Life Support - Health Care			
SYS-OP-LS-HC-1	✓	Yes	Yes
SYS-OP-LS-HC-2	✓	Yes	Yes
SYS-OP-LS-HC-3	✓	Yes	Yes
SYS-OP-LS-HC-4	✓	Yes	Yes
SYS-OP-LS-HC-5	✓	Yes	Yes
SYS-OP-LS-HC-6	✓	Yes	Yes
SYS-OP-LS-HC-7	✓	Yes	Yes
SYS-OP-LS-HC-8	✓	Yes	Yes
SYS-OP-LS-HC-9	✓	Yes	Yes
SYS-OP-LS-HC-10	✓	Yes	Yes
Communications			
SYS-OP-COM-1	✓	Yes	Yes
SYS-OP-COM-2	✓	Yes	Yes
SYS-OP-COM-3	✓	Yes	Yes
SYS-OP-COM-4	✓	5000 m	> 5000 m
SYS-OP-COM-5	✓	Yes	Yes
SYS-OP-COM-6	✓	Yes	Yes
SYS-OP-COM-7	✓	Yes	Yes
SYS-OP-COM-8	✓	Yes	Yes
SYS-OP-COM-9	✓	100 $Mbps$	564 $Mbps$
SYS-OP-COM-10	✓	100 $Mpbs$	564 $Mbps$
Command & Data Handling			
SYS-OP-CDH-1	✓	Yes	Yes
SYS-OP-CDH-2	✓	< 2 kW	0.55 kW
SYS-OP-CDH-3	✓	99	99
SYS-OP-CDH-4	✓	Yes	Yes
SYS-OP-CDH-5	✓	Yes	Yes
SYS-OP-CDH-6	✓	100 rad	1000 rad
Power			
SYS-OP-PWR-1	✓	Yes	Yes
SYS-OP-PWR-2	✓	Yes	Yes
SYS-OP-PWR-3	✓	7.5 kW in 12 days	7.5 kW in 12 days
SYS-OP-PWR-4	✓	18 kW	19 kW
SYS-OP-PWR-5	✓	23 kW for 1 hour	24 kW for 1 hour
SYS-OP-PWR-6	✓	≤160 W	< 160 W
SYS-OP-PWR-7	✓	≤200 W	160 W
SYS-OP-PWR-8	✓	≥500 W	500 W
SYS-OP-PWR-9	✓	≥4 kW	4 kW
SYS-OP-PWR-10	✓	≥1 kW	1 kW
SYS-OP-PWR-11	✓	≥2 kW	2 kW

	Astronauts Rover Vehicle		
SYS-OP-ARV-1	✓	2 people	2 people
SYS-OP-ARV-2	✓	left/right	left
SYS-OP-ARV-3	✓	Yes	Yes
SYS-OP-ARV-4	✓	Yes	Yes
SYS-OP-ARV-5	✓	Yes	Yes
SYS-OP-ARV-6	✓	Yes	Yes
SYS-OP-ARV-7	×	Yes	Future development
SYS-OP-ARV-8	×	Yes	Future development
SYS-OP-ARV-9	×	Yes	Future development
	Materials and Structures		
SYS-OP-MAT-1	✓	6 g	> 6 g
SYS-OP-MAT-2	✓	6 g	> 6 g
SYS-OP-MAT-3	✓	Yes	Yes
SYS-OP-MAT-4	×	Yes	Future development
SYS-OP-MAT-5	×	Yes	Future development
SYS-OP-MAT-6	✓	Yes	Yes
SYS-OP-MAT-7	×	Yes	Future development
SYS-OP-MAT-8	×	Yes	Future development
SYS-OP-MAT-9	≈	Yes	partially designed for
SYS-OP-MAT-10	≈	Yes	partially designed for
	Post operations - Expandability		
SYS-POS-EXP-1	✓	Yes	Yes
	Post operations - Sustainability		
SYS-POS-SUS-2	≈	Yes	Partially designed for
SYS-POS-SUS-4	✓	Yes	Yes
SYS-POS-SUS-6	✓	Yes	Yes
SYS-POS-SUS-7	✓	Yes	Yes
SYS-POS-SUS-10	✓	Yes	Yes
SYS-POS-SUS-11	✓	Yes	Yes
SYS-POS-SUS-12	✓	Yes	Yes
SYS-POS-SUS-14	✓	Yes	Yes
SYS-POS-SUS-15	✓	Yes	Yes
SYS-POS-SUS-16	✓	Yes	Yes
SYS-POS-SUS-19	✓	Yes	Yes

15.2. Feasibility Analysis

In this section, the feasibility analysis will be performed. The rationale for all requirements that have not been, or only partially complied to, will be provided. The requirements for which is mentioned 'future development' will not be evaluated, as there was not enough time for the team to develop the design to such a level that these requirements could be met with certainty.

U-10: The habitat shall comply with the international legislation regarding Moon exploration.
After a detailed study of the Moon Agreement [183], two articles are found to be mentioned. First, Article 7 states the following:

> 7.1 *In exploring and using the moon, States Parties shall take measures to prevent the disruption of the existing balance of its environment, whether by introducing adverse changes in that environment, by its harmful contamination through the introduction of extra-environmental matter or otherwise. States Parties shall also take measures to avoid harmfully affecting the environment of the earth through the introduction of extraterrestrial matter or otherwise.*

7.2 *States Parties shall inform the Secretary-General of the United Nations of the measures being adopted by them in accordance with paragraph 1 of this article and shall also, to the maximum extent feasible, notify him in advance of all placements by them of radio-active materials on the moon and of the purposes of such placements.*

This tells us measures to the greatest extend shall be made to not disrupt the Moon's environment in any way, and that the used power plant module is allowed on the Moon. However, the Secretary-General of the United Nations shall be well-informed, far in advance of the start of the mission. At the end of the mission, the power plant will be brought back to Earth. Then, Article 8 states:

8.1 *States Parties may pursue their activities in the exploration and use of the Moon anywhere on or below its surface, subject to the provisions of this Agreement.*

8.2 *For these purposes States Parties may, in particular:*

8.2 a) *Land their space objects on the Moon and launch them from the Moon;*

8.2 b) *Place their personnel, space vehicles, equipment, facilities, stations and installations anywhere on or below the surface of the Moon;*

8.3 *Personnel, space vehicles, equipment, facilities, stations and installations may move or be moved freely over or below the surface of the Moon.*

Article 8.1 states the Moon can be used for activities of States Parties. Article 8.2 elaborates on this, and states in 8.2 b) that States Parties are allowed to place facilities, stations and installations anywhere on or below the surface of the Moon. Therefore, it can be concluded that it is allowed to sinter the surface of the Moon to a certain extent. Only the minimum surface needed for the activity shall be sintered. As this surface area is chosen in compliance with a high safety for the astronauts and minimum lunar dust disruption during landing, it is believed that the plans do not contradict the Moon Agreement.

SYS-OP-SFT-M-4: The habitat shall have early warning systems for meteoroids with a kinetic energy of more than 50 MJ at least 300 s before impact.
As was explained in subsection 8.1.7, with today's technology there isn't a reliable way of predicting most of the meteoroids that will impact the Lunar surface. As mentioned, meteoroids are only detectable once they emit some sort of energy. In this case, this is either when it passes through the Earth's atmosphere and leaves an ionising trail or at the moment of impact. From the meteoroids that hit the Earth-facing side of the Moon, only a small part is detectable. These are the meteoroids that slightly pass through the Earth's atmosphere before heading for the Moon. As mentioned in subsection 8.1.7, the US Space Surveillance Network's collaboration is needed to detect and communicate possibly threatful meteoroids, and meteoroids storms heading for the Moon. The approximated time of detection is estimated to be 1.5 h, which assumed a speed of 70km/s and the distance from Earth to the Moon. Nevertheless, there shall always be meteoroids that have a possibility of striking the habitat and that can not be detected in advance with today's technology.

SYS-POS-SUS-2: No harmful contamination of space and celestial bodies shall take place by the introduction of extraterrestrial matter.
The same procedure as is executed during any space mission will be executed during the building of all parts. All parts will be build in clean rooms, and decontaminated after completion. This way, the amount of extraterrestrial matter brought into space via contamination of systems is minimised. Using suitports, the outside of the spacesuits will not enter the habitat, having no possible way of contaminate outer space. Only when the airlock is opened, the left-over air will be put into outer space. As most of the air is already sucked away during decompression of the airlock, this contamination is rather small.

SYS-POS-MAT-9 & SYS-POS-MAT-1
Both these requirements state that the habitat's materials must be able to withstand the temperature range and cycling of the Lunar environment. The analysis and design has been satisfied for regolith, which is the material in direct contact with the Lunar environment. The exits and the materials of the rovers, however, haven't been checked to meet these requirements and shall require more verification for future development.

16 Conclusion

Building human presence on the Moon and beyond has been a goal for humanity since the beginning of space exploration, with conceptual studies having been performed on the subject for over sixty years. In this document, a detailed elaboration upon a concept for a 10-year Lunar habitat, chosen by a thorough trade-off procedure is documented.

Starting with the conceptual trade-off, a concept featuring horizontally placed and docked cylindrical modules was chosen. The location of the mission was determined to be close to the Apollo 11 site, in the Mare Tranquillitatis area, where relatively large, smooth and flat areas are present. These areas typically have a rather thick layer of Lunar regolith. This is important because it will be used to shield the habitat from radiation and meteoroids.

By means of thorough thermal, structural and radiation models it was determined that 1.2 m of Lunar regolith was necessary to protect against the harmful cosmic background radiation. The analyses showed that this was also abundant to protect against meteoroid impacts and keep a constant temperature inside the habitat, which lowers thermal stresses.

The habitat itself will be consisting of four horizontally placed cylindrical structures, connected by a central node and featuring one airlock module on an outward face of one of the cylinders. The node and cylinders will provide 163 m^3 of habitable volume, stretched over 67 m^2 of floor surface area. The remaining internal volume is used to house the subsystems.

To comply with the requirements regarding the functions the systems needs to perform, the main subsystems were sized and investigated. A thorough literature study on the current developments of Environmental Control and Life Support Systems (ECLSS) was performed to set a baseline for the design of the system. The ECLSS and other subsystems will be managed by the Command and Data Handling (CDH) system, for which an architecture ensuring almost perfect reliability is constructed, given the importance and sensitivity of the system. To provide the power over the 14-day eclipses the Moon faces, a trade-off between nuclear and photovoltaic power generation and their implications was conducted. Ultimately, two KRUSTY nuclear fission reactors will be used to provide a continuous power supply of 20 kW to the habitat at all times. Two antenna dishes will provide communications to Earth's ground stations at a data rate of over 500 $Mbps$ with a round-trip time of about 2.71 s.

At this stage, most subsystems had been designed in sufficient detail to initiate an investigation of the logistics and mission profile. To this end, a comprehensive table detailing the planning of the 19 weeks that stretch from the launch of the first module to the arrival of the first crew was created. This phase in the mission will be critical and proves to be a logistic challenge. Rovers will first be landed on the Lunar surface to prepare it by sintering for the arrival of the habitation modules, and they will assemble the habitat before the crew arrives to finalise the system for use, starting their one-year mission on the Moon.

Concurrently, all subsystems were integrated into a coherent system, and all subsystems were fitted geometrically in the habitat to create the definitive internal design. The final budgets could be updated from parametric estimations to estimations based on a build-up from the components of the system, which yields more accurate estimates. The total mass will be 87,309 kg, the total pressurised volume will be 466 m^3, and the system will require an average power of 16,463 W. The final cost investigation concluded a cost of €29,8 billion.

Proper system design, especially in a harmful environment such as that of the Moon, requires all risks to be investigated and accounted for. To make sure the team had not neglected to include any risks to this mission, a systematic approach was used to identify mission weak points and mitigate them where necessary.

Conclusively, it was determined that there is still a lot of work to be done before a manned station on an extraterrestrial body will be realised. However, as the efforts presented in this document show, it is certainly not impossible for humans to evolve beyond Earth.

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A
Functional Analysis

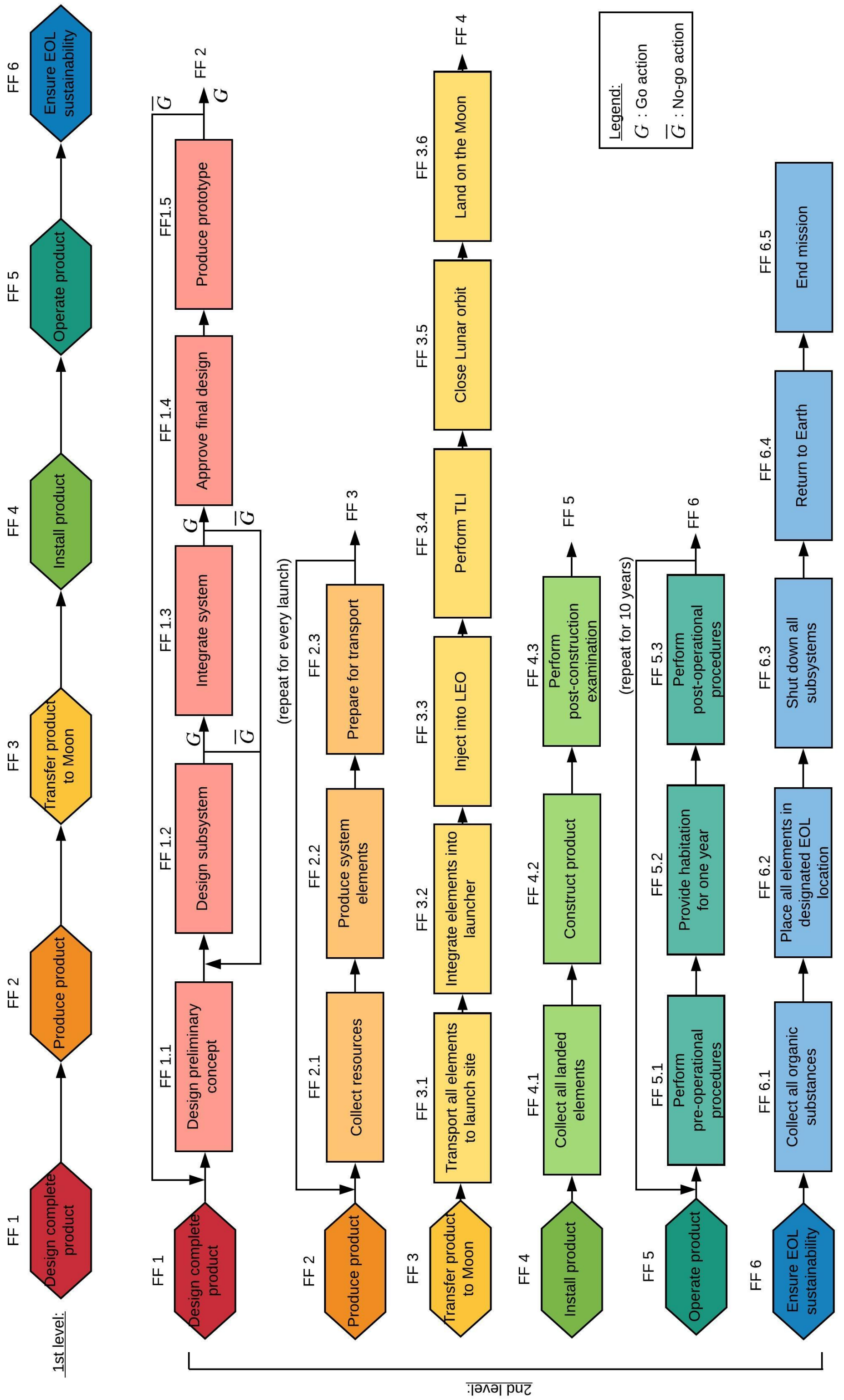

Figure A.1: Functional flow diagram of the product: first and second levels

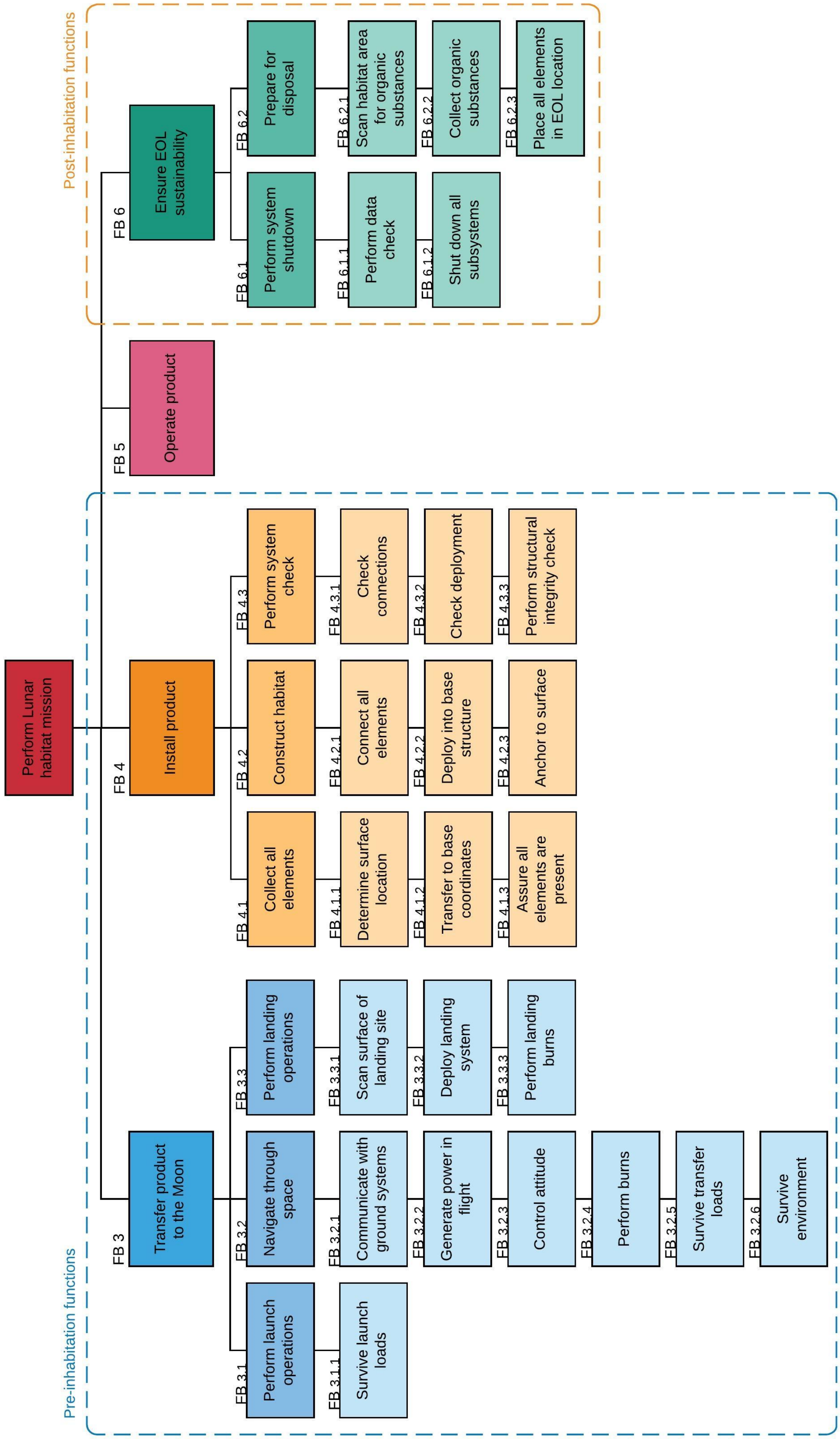

Figure A.2: Functional breakdown diagram of FB 3, 4 and 6: second and third levels

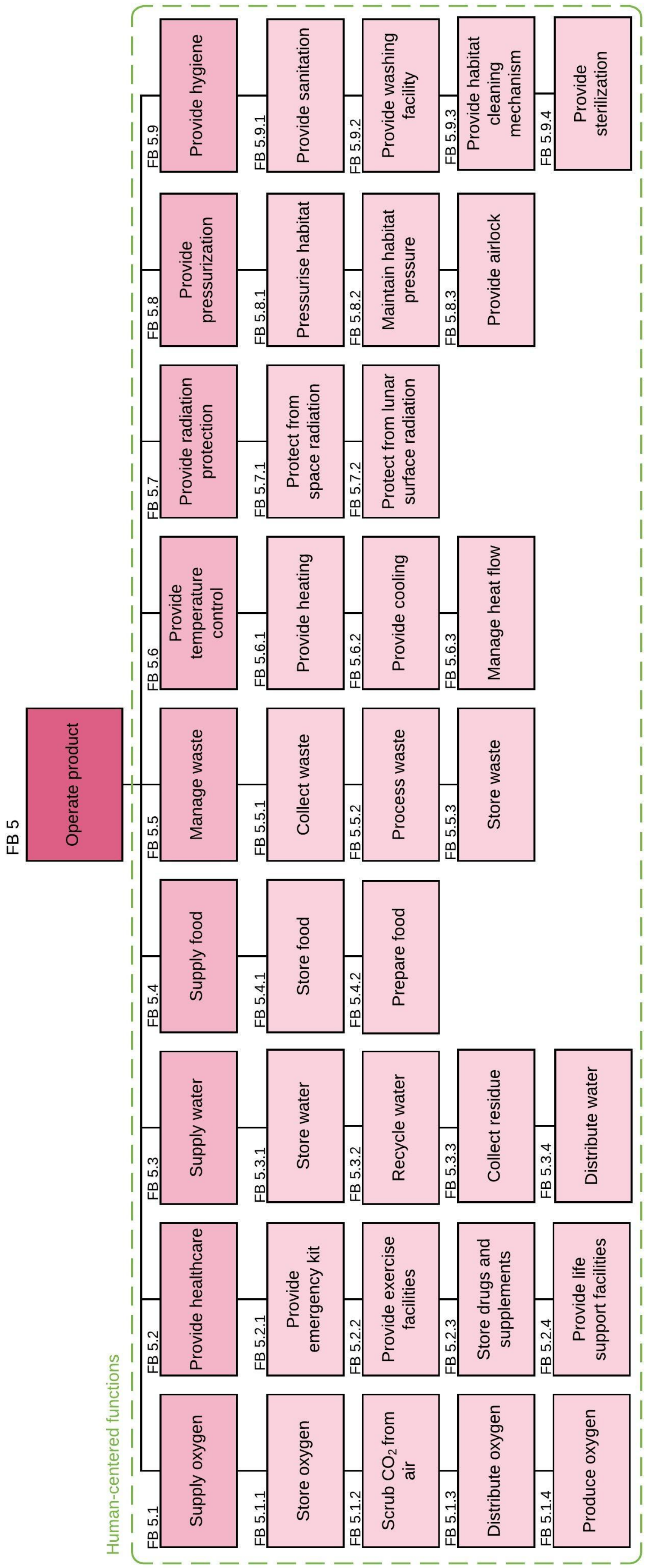

Figure A.3: Functional breakdown diagram of FB 5: second and third levels FB 5.1-5.9

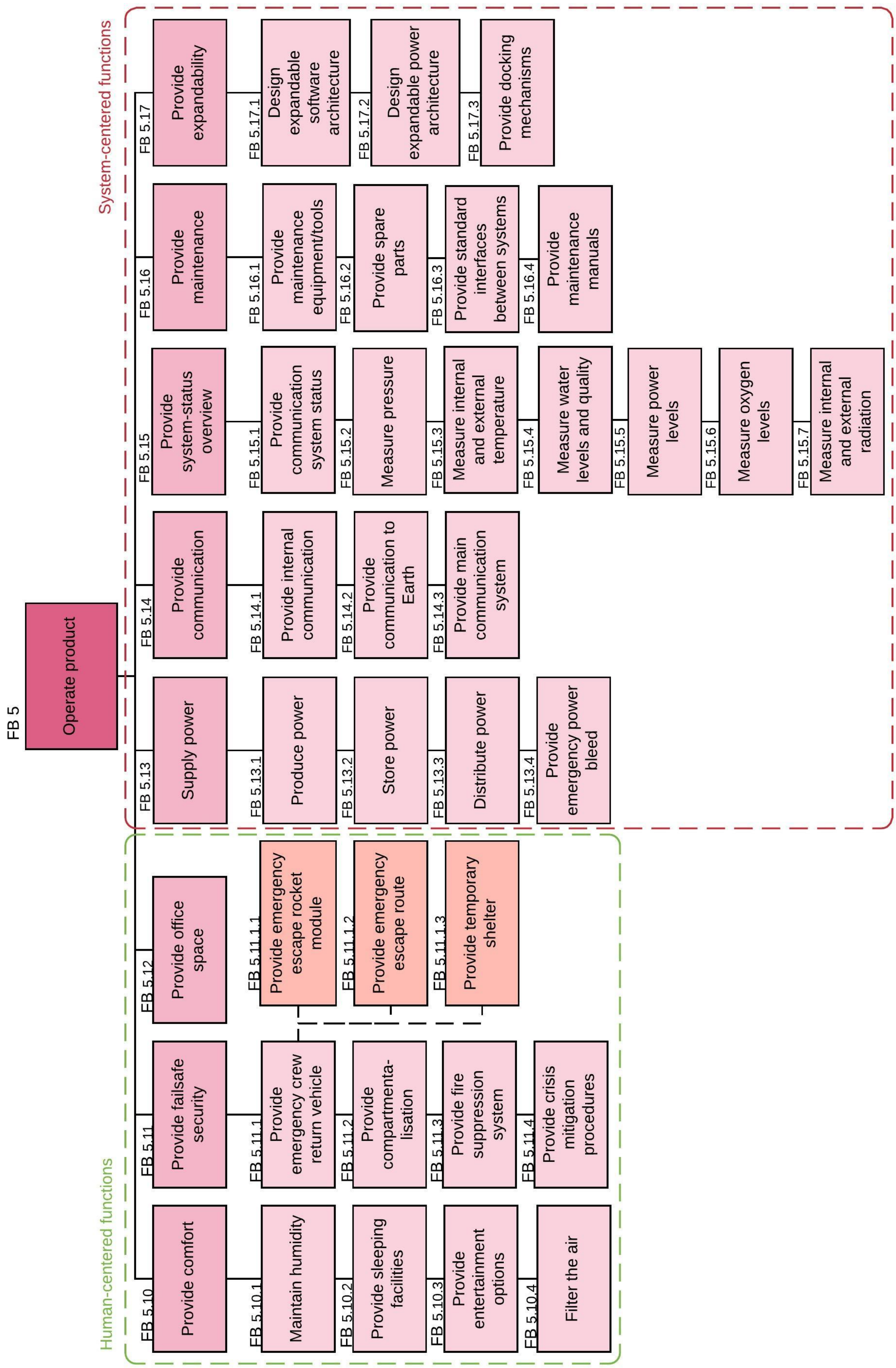

Figure A.4: Functional breakdown diagram of FB 5: second and third levels FB 5.10-5.17

B
List of Requirements

User

U-1 The habitat shall provide a living and working space for four astronauts;
U-2 The habitat shall be able to host the crew for at least one year at a time;
U-3 The habitat shall remain in the same location it was constructed for 10 years;
U-4 The habitat shall have such environmental conditions that the inhabitants can use the same outfits compared to those worn in the ISS;
U-5 The habitat shall provide a safe and relatively relaxed space to live, sleep, and work;
U-6 The habitat shall be fully operational for a duration of at least 10 years;
U-7 The habitat shall be situated on either the Apollo 11 or 17 landing site;
U-8 The habitat shall remain on the Moon at the end of its life as a derelict entity.
U-9 The constraints imposed by transport to the Moon and its assembly shall be considered;
U-10 The habitat shall comply with the international legislation regarding Moon exploration;
U-11 The habitat shall provide a reliable and safe environment by protecting the astronauts against all hazards that may occur on the Moon surface, such as strongly fluctuating temperature, micrometeoroid impact, high radiation levels, and lack of oxygen.
U-12 The supply installation for energy, air, food, communication, and all other subsystems shall be defined as to requested capacity;
U-13 The transportation of habitat construction material to the Moon shall be done by launchers that are commercially available no later than 2018;
U-14 The habitat construction elements and any payload transported to the Moon shall be able to withstand the loads introduced by the launching system;
U-15 The system shall have adequate backup facilities to allow the safe return of the astronauts to Earth under all conditions;
U-16 The design or transportation of the additional equipment to be used at a later stage for the actual Moon exploration shall be ignored;
U-17 The all-inclusive cost of research, development, manufacturing, transport, deployment, usage, and disposal shall not exceed 500000 $€/kg$ of material delivered to the Moon;
U-18 The habitat system shall be designed to be operated in a gravitational acceleration of 1.6 m/s^2;
U-19 The habitat system design shall allow for future expansions.

Pre operations - Transfer to the Moon

SYS-PRE-TRN-1 Each payload shall have a dry mass lower than 11.9 t;
SYS-PRE-TRN-2 The transfer vehicle structure shall be able to withstand an axial load of 4.5 g during launch without sustaining any damage;
SYS-PRE-TRN-3 The transfer vehicle structure shall be able to withstand a lateral load of 2 g during launch without sustaining any damage;
SYS-PRE-TRN-6 The maximum acceleration during transfer shall not exceed 4.5 g;
SYS-PRE-TRN-10 The system shall provide a minimum of 3.3 km/s delta-V;
SYS-PRE-TRN-13 The landing system shall ensure a landing accuracy of at least 50 m;
SYS-PRE-TRN-17 The launcher shall not use PBAN propellant;
SYS-PRE-TRN-18 The launcher shall not use APCB propellant;
SYS-PRE-TRN-19 The launcher shall not be fully expendable;

Pre operations - Installation

SYS-PRE-INS-1 The construction of the habitat shall comply with the ESA sustainability strategy;
SYS-PRE-INS-3 The system has to be able to transfer all landed elements to the designated habitat location;

Safety - Meteoroids

SYS-OP-SFT-M-1 The habitat shall provide protection against meteoroids with a maximum kinetic energy release of 21300 J;
SYS-OP-SFT-M-2 The habitat shall include an emergency alert in case of meteoroid impact;
SYS-OP-SFT-M-3 The habitat shall have measures in place to ensure mission continuity in case of an impact of meteoroid with a total kinetic energy release of 50 MJ;
SYS-OP-SFT-M-4 The habitat shall have early warning systems for meteoroids with a kinetic energy of more than 50 MJ at least 300 s before impact;
SYS-OP-SFT-M-6 Procedures manuals in case of a critical meteoroid impact shall be present in the habitat;
SYS-OP-SFT-M-7 The outer layer of the habitat shall not be damaged by meteoroids;
SYS-OP-SFT-M-8 The habitat shall be protected for a PNP of at least 0.98.

Safety - Radiation

SYS-OP-SFT-R-1 The interior radiation levels shall not exceed 165 mGy/yr;

SYS-OP-SFT-R-2 The habitat shall include a system that measures internal and external radiation with an accuracy of at least 5 mGy/yr.

Life support - Internal Atmosphere

SYS-OP-LS-IA-1 The habitat shall contain no gases in toxic quantities;

SYS-OP-LS-IA-2 The internal atmosphere of the habitat shall be at least 69 kPa;

SYS-OP-LS-IA-3 The internal atmosphere of the habitat shall be at most 102.4 kPa;

SYS-OP-LS-IA-4 An oxygen supply of at least 40 kg shall be present in the habitat;

SYS-OP-LS-IA-5 Oxygen shall be provided to every interior area of the habitat;

SYS-OP-LS-IA-6 The internal air composition of the habitat shall contain between 20% and 40% of oxygen;

SYS-OP-LS-IA-7 The internal air of the habitat shall not contain hazardous particles that have a NFPA rating in any category larger than 0 [184];

SYS-OP-LS-IA-8 The internal humidity of the air in the habitat shall be in the range of 40% to 70%;

SYS-OP-LS-IA-9 The carbon dioxide content of the air inside the habitat shall be below 0.4%;

SYS-OP-LS-IA-12 The dewpoint temperature shall be within the range of 278 K and 289 K during normal operations;

SYS-OP-LS-IA-15 The habitat shall include a systems that can detect BMD levels from 1% as defined by The National Academies [185].

Life support - Water

SYS-OP-LS-W-1 The habitat shall have a water supply storage of at least 668 l;

SYS-OP-LS-W-3 A redundant water recycling system shall be present;

SYS-OP-LS-W-4 The water system shall provide at least 34.1 l drinkable water per day;

SYS-OP-LS-W-5 The redundant water supply system shall sustain the crew for at least 12 days;

SYS-OP-LS-W-6 The habitat shall include systems that can measure water levels with an accuracy of 0.5 l;

SYS-OP-LS-W-7 The habitat shall include systems that can measure water purity with an accuracy of 1% of the toxic quantities as defined by The National Academies [185];

SYS-OP-LS-W-9 The water system shall provide cold water at 277 ±3 K;

SYS-OP-LS-W-10 The water system shall provide ambient water at 294 ±5 K;

SYS-OP-LS-W-11 The water system shall provide hot water up to 339 K;

SYS-OP-LS-W-12 The temperature of the crew hygiene water shall be adjustable from 294 to 318 K.

Life support - Food

SYS-OP-LS-F-1 The redundant food supply shall sustain the crew for at least 12 days;

SYS-OP-LS-F-2 The quality of the food present in the habitat shall comply with Regulation (EC) No 178/2002 of the European Union [186];

SYS-OP-LS-F-3 At least 3000 $Kcal$ shall be available per person per day;

SYS-OP-LS-F-4 Maximum 35% of the $Kcal$ consumed by astronauts shall come from lipids;

SYS-OP-LS-F-5 The food supplied to astronauts shall have a below average sodium content;

SYS-OP-LS-F-6 At least 50% of the food provided to astronauts shall be fresh.

Life support - Waste

SYS-OP-LF-WST-1 The disposal of waste shall not disrupt the Moon's environment;

SYS-OP-LF-WST-2 No waste shall be left on the Moon at the EOL;

SYS-OP-LF-WST-3 All living spaces shall be scanned for waste and contamination at EOL;

SYS-OP-LF-WST-4 All waste shall be disposed at end of operation.

Life support - Pressure

SYS-OP-SFT-P-2 Single habitat modules shall be able to be depressurised;

SYS-OP-SFT-P-4 The EVA airlocks shall have a minimum volume of 10 m^3;

SYS-OP-SFT-P-5 The pressure inside the airlock shall not exceed the internal habitat pressure;

SYS-OP-SFT-P-6 The habitat shall include systems that can measure cabin pressure with a 1 Pa accuracy;

Life support - Thermal control

SYS-OP-LS-T-1 The internal air temperature of the habitat shall be between 292 and 300 K during normal operations;

SYS-OP-LS-T-2 The habitat shall contain a redundant temperature control system;

SYS-OP-LS-T-3 The habitat shall include systems that can measure internal and external temperature with an accuracy of 1 K.

Life support - Comfort

SYS-OP-LS-CF-16 The level of noise shall be kept below NC-50;

SYS-OP-LS-CF-17 The level of noise shall be kept below NC-40 in both the sleeping cabin and the quiet room.

Life support - Health care

SYS-OP-LS-HC-1 The habitat shall include standard first aid medical equipment;

SYS-OP-LS-HC-2 The habitat shall have a health monitoring system;

SYS-OP-LS-HC-3 The astronauts shall be trained to operate all first aid and medical equipment present in the habitat;

SYS-OP-LS-HC-4 The habitat shall include exercise facilities for the astronauts;

SYS-OP-LS-HC-5 Astronauts performing the mission will need to obtain a JAR-FCL 3, Class 2 medical examination certificate or equivalent such as those from the Federal Aviation Authority (FAA);

SYS-OP-LS-HC-6 The astronauts performing the mission shall be medically examined and comply with the requirements from [187];

SYS-OP-LS-HC-7 There shall be contact available with a psychiatrist in case of mental health problems;

SYS-OP-LS-HC-8 There shall be contact available with a medic in case of physical health problems;

SYS-OP-LS-HC-9 Emergency health protocols shall be present in case an astronaut experiences health issues;

SYS-OP-LS-HC-10 The exercise facilities shall include a water dispenser.

Communication

SYS-OP-COM-1 The habitat shall facilitate continuous communication to Earth by video to the ground station on Earth;

SYS-OP-COM-2 The habitat shall facilitate continuous communication to Earth by audio to the ground station on Earth;

SYS-OP-COM-3 The habitat shall facilitate continuous communication to Earth by text to the ground station on Earth;

SYS-OP-COM-4 There shall be audio communication available between astronauts within a range of at least 5000 m from each other at all times;

SYS-OP-COM-5 The communication system shall be fully redundant;

SYS-OP-COM-6 The habitat shall be able to perform communication system checks;

SYS-OP-COM-7 Private communication to Earth shall be available for every astronaut;

SYS-OP-COM-8 The private communication shall be available 24/7;

SYS-OP-COM-9 The downlink of the communication system shall be at least 100 $Mbps$;

SYS-OP-COM-10 The uplink of the communication system shall be at least 100 $Mbps$.

Command & data handling

SYS-OP-CDH-1 The CDH system shall provide an overview of the status of all electronic systems in the habitat;

SYS-OP-CDH-2 The CDH system shall consume less than 2 kW;

SYS-OP-CDH-3 The CDH shall have a total system reliability of 99 %;

SYS-OP-CDH-4 The CDH system shall be fully redundant;

SYS-OP-CDH-5 The CDH system shall be able to check that it is running nominally;

SYS-OP-CDH-6 The CDH system shall operate nominally after a 100 rad dose.

Power

SYS-OP-PWR-1 The habitat shall have a power system that does not require resupply from Earth to complete a one year mission;

SYS-OP-PWR-2 There shall be a central power management unit in the Lunar habitat;

SYS-OP-PWR-3 The redundant power system shall provide average power of at least 7.5 kW for at least 12 days;

SYS-OP-PWR-4 The power system shall provide an average power of at least 20 kW;

SYS-OP-PWR-5 The power system shall provide a minimum peak power of 25 kW for 1 hour;

SYS-OP-PWR-6 The communication system shall not have an average power exceeding 160 W;

SYS-OP-PWR-7 The communication system shall have a maximum peak power of 200 W;

SYS-OP-PWR-8 In case of emergency, minimum 50 W shall be available for the back-up communication systems;

SYS-OP-PWR-9 In case of emergency, minimum 4 kW shall be available for the ECLSS;

SYS-OP-PWR-10 In case of emergency, minimum 1 kW shall be available for the CDH system;

SYS-OP-PWR-11 In case of emergency, minimum 2 kW shall be available for the maintenance facilities systems.

Astronaut rover vehicle

SYS-OP-ARV-1 The ARV shall accommodate two seated astronauts wearing spacesuits;

SYS-OP-ARV-2 The ARV shall be operated by either the left or the right astronaut;

SYS-OP-ARV-3 The ARV shall carry payload, including astronauts, tools, science equipment, cameras, etc.;

SYS-OP-ARV-4 The ARV shall be designed in such way that it can be folded inside the lander;

SYS-OP-ARV-5 The ARV shall operate in the harsh vacuum environment of the Moon;

SYS-OP-ARV-6 The ARV shall be regolith-resistant;

SYS-OP-ARV-7 The ARV shall withstand temperatures between -160 to +125 $^\circ C$;

SYS-OP-ARV-8 The ARV shall be designed for maximum static and dynamic stability on slopes up to 45°;

SYS-OP-ARV-9 The tires of the ARV shall be able to withstand the rough surface of the Lunar soil for its operational lifetime.

Materials

SYS-OP-MAT-1 No materials shall buckle under launch/landing forces of a maximum of 6 g;

SYS-OP-MAT-2 No materials shall plastically deform under the launch/landing forces of a maximum of 6 g;

SYS-OP-MAT-3 All Lunar dust shall be removed when entering the living area of the habitat;

SYS-OP-MAT-4	All Lunar dust shall be removed when entering the lunar orbital lander;
SYS-OP-MAT-5	All Lunar dust shall be removed when entering the TV cabin;
SYS-OP-MAT-6	Every system in direct contact with Lunar environment shall maintain its function for its operating lifetime, despite the abrasive nature of the Lunar regolith;
SYS-OP-MAT-7	The space suits shall have a lifetime of at least one year;
SYS-OP-MAT-8	The space suits shall have a continuous operating time of at least 8 hours;
SYS-OP-MAT-9	Habitat material in direct contact with the Lunar environment shall be able to withstand a temperature range of 99 to 380 K without transition;
SYS-OP-MAT-10	Each habitat material in direct contact with the Lunar environment shall be able to withstand a temperature cycling range of 281 K for a minimum of 120 cycles;
	Expandability
SYS-POS-EXP-1	The building site shall have attachment facilities;
	Sustainability
SYS-POS-SUS2	No harmful contamination of space and celestial bodies shall take place by the introduction of extraterrestrial matter;
SYS-POS-SUS4	During the operational phase of the mission, a complete and thorough update on the mission shall be given at least every 30 days;
SYS-POS-SUS6	Any information derived from the mission shall only be used for scientific purposes;
SYS-POS-SUS7	Detailed mission information shall only be shared with parties upon agreement of not using it for commercial reasons;
SYS-POS-SUS10	No waste shall be left on the Moon at the EOL;
SYS-POS-SUS11	All living spaces shall be scanned for waste and contamination at EOL;
SYS-POS-SUS12	All subsystems shall be shut down at EOL;
SYS-POS-SUS14	All supplies shall be emptied or stored at end of operation;
SYS-POS-SUS15	All living spaces shall be sterilised at end of operation;
SYS-POS-SUS16	All living spaces shall be sterilised at EOL;
SYS-POS-SUS19	The system shall be able to scan the internal area for organic substances;

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