H.O.M.E.
Habitat on Moon: Exodus

Final Report

DSE Group 28

Delft University of Technology

TU Delft

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by

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Executive Summary

It has been over 50 years since man first set foot on the Moon and proved that mankind could extend its borders beyond Earth, and yet no permanent lunar outpost has been set yet. Similar to how the construction of larger ports allowed for the exploration of the New World in the Renaissance, human exploration of space requires infrastructure beyond that required for spaceships and launchers. As our only natural satellite, the Moon is the best choice for a permanent base, one that would be capable of refuelling interplanetary missions, providing plentiful resources for the needs of people on Earth, and giving opportunities for fundamental research.

ESA set the goal to design the first crewed semi-permanent lunar outpost. The first challenge would be to design a mission resulting in a habitat capable of lasting 10 years, with four-person crews renewed on a yearly basis. Successful return of the first round of astronauts would prove long-term outposts far from Earth are possible and potentially profitable. The project encapsulates the mission as a whole, from the building of the habitat to its end-of-life, including the arrival and exit of the first round of astronauts. The mission was named 'Exodus', meaning mass departure, due its potential to be the first step towards human migration outside of Earth.

The habitat has to be able to fend off against the many dangers of the lunar environment, from the vast amounts of radiation to the complete lack of an atmosphere. It also is a space mission, meaning that keeping the habitat lightweight is necessary if the mission is to be affordable. Furthermore, sustainability is another key aspect, both on the Moon and on Earth, as this mission is meant to represent a renewal of human space exploration, and incorporating sustainability in such a feat would establish a powerful precedent.

The location for the habitat was chosen to be in the ridge near the Shackleton crater due to its near-constant illumination, which in turn gives both ample power and a manageable temperature range. Its location near the South Pole could also give an excellent source of water, although this needs to be confirmed.

It was decided that the optimal solution would be to have an inflatable structure, as it is considerably lighter and more compact than a rigid one. This would allow the entire habitat to be carried by a single launcher. The structure was chosen to be coupled with in-situ resources, as radiation and meteoroid impacts are both problems that are solved with great amounts of mass, which would be costly if it were to be brought from Earth.

The inflatable habitat is in the shape of a cylinder, and placed on its longitudinal side, up to 1 m deep in the lunar surface. It is connected to an airlock on one end for entering and leaving the habitat. The radius of the habitat is 6 m and the length is 27 m, excluding the airlock. It contains a tensairity structure for the flooring, with load bearing boxes underneath. Regolith bags will cover the top of the habitat with a thickness of 1 m for insulation, meteorite protection, and radiation shielding.

This multi-layer inflatable structure will be assembled remotely with robots. The nature of multi-layer structures allows them to be tailor made to the specific environment, with each layer fending off against a specific hazard. The structure incorporates a Teijin-conex lining, a polyethylene cosmic radiation protection layer, three saran bladders, a Dyneema SK99 restraint layer, and aluminised kapton and mylar for insulation.

The habitat will be manufactured by a variety of robots controlled from Earth in a period of about 15 months. The first step will be to set up the solar power farm, following this excavation of regolith will begin to form a thermal and radiation protection layer on top of the habitat inflatable. The regolith bags will be placed on top of the structure by 2 cranes.

The lunar base will be powered by solar power when sunlight is available. During initial construction of the solar field and during lunar night time, the base will receive its power from a regenerative liquid hydrogen fuel cell system. Due to the choice of location, the solar panels will be deployed vertically and are spaced in such a way that they do not intercept each others light

To communicate with Earth, a lunar relay satellite will be used. Communication between the habitat and the satellite is achievable with a high signal to noise ratio. However, the communication requires a data rate of 1200 Mbps, which in comparison to the ISS is quite high. Current systems are capable of reaching 622 Mbps, so developments will have to be made to comply with the requirements.

The safehouse was designed to be a compact living space capable of hosting the astronauts for 21 days in case of solar particle events or main habitat failure. As such, it was designed to locally offer very high solar radiation protection and to generally have fewer potential points of failure than the inflatable. The final

safehouse design is a rigid structure with varying materials and thicknesses throughout, and one that uses its water storage to protected against radiation as well as to keep the astronauts hydrated.

A total number of 3 launches have been scheduled for this mission. The first 2 launches will carry the payload needed for the habitat construction, the habitat itself and the safe house. The fully reusable Starship from SpaceX and its ”Super Heavy” rocket will be use to bring such a large amount of mass to the lunar surface. The third launch will bring the astronauts and their lunar rover to the Moon, once the habitat is fully ready. The travel time of each journey (excluding refilling) will be 6 days. The big steps of the journey are the following; starting in LEO the spacecraft will go into 2 Hohmann transfers (Earth and Moon) to finally arrive in LLO and finally perform lunar landing.

The reliability of the mission was seen to be an absolute minimum of 0.89, but one that is prone to increase with improvements in life support systems, orbit insertion manoeuvres, and launcher reliability. To compensate for an initially disappointing reliability estimate, a maintenance schedule was put in place that incorporates daily inspections of the systems most prone to failure. Some safety recommendations are also given.

The costs and profits of the mission were estimated using literature and calculated with Monte Carlo sampling in order to capture the risk inherent in future investment and the team’s uncertainty concerning its assumptions. Adding a preliminary lunar propellant production facility to the habitat yielded a 95% probability that no more than 5 billion euros be spent in total over 10 years, and a 40% chance that the operation would turn a net profit of at least 3 billion euros.

Overall, space missions are inherently environmentally unsustainable, but a number of steps have been taken for our mission’s impact on the environment to be minimised. Starting with the launch plan, it has been decided to use Starship, a fully reusable launcher, and thereby use a minimal number of launches to bring the cargo to the Moon. This is directly related with the habitat design in which a number of sustainable design choices have been made. To meet this first step, the up-mass had to be reduce so the use of natural lunar resources, regolith, was chosen. This leads to less polluting material, less propellant and rocket needed. Another design option adding to the sustainability of our design is the use of solar resources to create power. Finally, for the return mission, it has been decided to bring back to Earth any waste left behind to reduce lunar pollution and the creation of space debris.

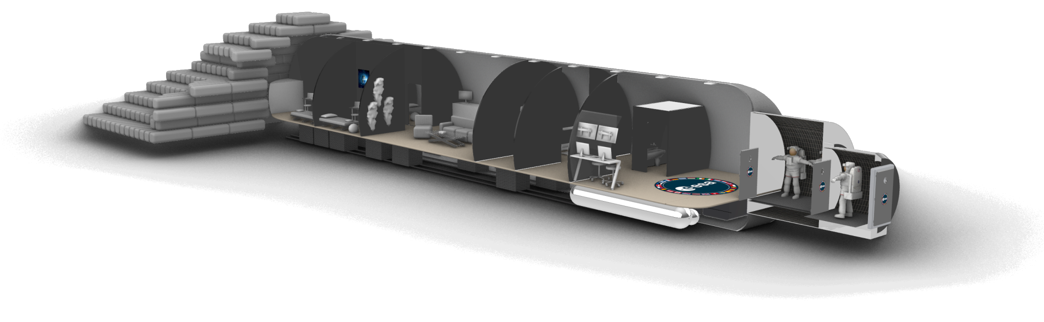

Figure 1: Habitat Concept

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[1] MARSHA project URL: https://www.aispacefactory.com/marsha [Cited on June 22, 2021]

Contents

Part I
Mission Overview and Conceptual Design

1. Introduction

With increasing interest in space development, as showcased by the sustained impressive growth in the space sector [1] [38], there comes a need to build infrastructure capable of sustainably exploiting space resources. To this end, building an outpost on Earth's only natural satellite becomes a very appealing idea. Indeed, it has been officially envisioned since before man stepped on the Moon in the first place. However, lack of technology readiness and a sharp decline in space activity after the Cold War made progress in this regard difficult. Lack of knowledge of in-situ resource utilisation (ISRU) made these projects very expensive, which led to a favouring of space-station development [82].

The International Space Station (ISS) is without a doubt one of the great milestones in recent space development history. It delivered considerable profits while providing unique research opportunities which, in turn, helped with scientific breakthroughs that might have never happened without it [2] [3]. In short, it proved that there is value in space development, and that the space environment can be useful beyond the expansion of human-kind.

Furthermore, extensive research on the space environment with the goal of building habitats has been conducted since the ideas were first proposed. Extensive guides for ISRU have been published [82], along with papers specifically focused on the design of lunar structures [47]. Together with planetary science findings [40], the possibility to set an outpost is more conceivable than ever.

It is in an effort to capitalise on these breakthroughs that multiple space agencies have started to spend resources on the development of lunar outposts. The Artemis program of NASA, the Moon village association by ESA, and the Russian-Chinese International Lunar Research Station all prove that there is a need and a will to set a permanent base on the Moon.

However, setting an outpost on the Moon is still a daunting task. The lunar environment is extremely hostile to human-life, with extreme temperature ranges, multiple large sources of unprotected radiation, sharp soil, no breathable atmosphere, and common meteoroid impacts. And yet, the step is necessary if we are to continue the exploration and development of space.

Hence, group 28 of the Summer 2021 DSE have set themselves the goal of designing the construction of a semi-permanent habitat on the Moon. Said lunar base would be built before the arrival of a group of 4 astronauts, who will remain there for a period of one year, before returning home safely. While on the Moon, they will be able to communicate with Earth, go on field trips, and conduct research using hypo-gravity currently unique to the Moon. After their return, the habitat will endure for at least another nine years, and will serve as the main piece of infrastructure to expand upon to eventually create a fully self-sufficient lunar outpost. The success of the mission would have tremendous impact on the future of space development, and can be seen as the first step towards becoming a multi-planetary species.

The report is divided into three parts:

Part I is meant to give an overview of the mission as a whole, so as to introduce the design concept the team has chosen to accomplish the mission. The market surrounding the mission is presented in Chapter 2. This chapter analyses the economic potential of the mission along with the factors the mission should incorporate to be economically sustainable. Having an overview and an economic justification of the mission, the risks thereof should be analysed and understood. These considerations should then be developed into a set of workable requirements. Requirements help establish goals and needs for the mission to be successful, and the process through which they were established is found in Chapter 3, which also describes the different

[1] URL: https://www.statista.com/statistics/946358/space¬economy¬global¬revenue¬segment¬2040/ [cited April 25, 2021]
[2] URL: [https://www.statista.com/statistics/330190/income-of-iss-world-worldwide]
[3] URL: [https://www.nasa.gov/mission_pages/station/research/news/iss-20-years-20-breakthroughs]

phases of the mission and its operational flow. Chapter 4 details how the lunar environment can negatively affect astronauts health, and gives an initial assessment of how each of these hazards can be addressed. Chapter 5 then creates an inventory of the main risks being considered throughout the mission as well as their corresponding mitigation strategies. Part I concludes with a presentation of the chosen design concept in Chapter 6 along with the preliminary decisions that constrained the design space into one that is manageable for a group of engineers.

Part II delves into the detailed design, and is therefore divided into the main functions of the habitat. Each chapter tackles its sustainability strategy individually, the goal being that each sub-system is optimised along a set of rules that are specific enough to be applied (rather than having large scale rules that are difficult to implement for small scale decisions). Chapters were all developed simultaneously, and therefore information is presented in an order that is convenient rather than chronological. Chapter 7 details the configuration of the habitat, the materials used for each of the main components, as well as the workings of the airlock. Chapter 8 then describes how the structure should be assembled and offers a detailed chronology of the logistics involved in the mission. The construction equipment and the habitat require an intricate power generation and distribution system, presented in Chapter 9. Chapter 10 lays out the different systems in place to keep the astronauts alive and healthy during their stay, as well as during trips outside of the habitat itself. Chapter 13 describes the launch, orbital, and return components of the mission, including how the habitat can be fit into the chosen launcher. The part concludes with the design of the safehouse, which is the structure to be used in case the main habitat fails.

Part III lays out the considerations for the mission that are not part of the subsystem designs. Such as the testing of the habitat before use, as described in Chapter 15 and the reliability, availability, maintainability, and safety details of the mission in Chapter 16. The verification and validation process is presented in Chapter 17, which details the decisions made during the design are known to solve the problem correctly. At this point, the mission as a whole is sufficiently detailed to estimate the expected return on investment, as seen in Chapter 18. The report ends with an overall conclusion in Chapter 19, followed by the teams recommendations for future research, design iterations, and routines to keep the astronauts safe.

2. Market Analysis

2.1. Introduction

The difference between the resources available in outer space and those found on Earth is that between entire supermarket's worth of food versus a single, solitary crumb laying on the ground. The process of tapping into the full economic potential of the cosmos has a timeline that transcends the lifetimes of anyone reading this report, but already within the next few decades lies the potential to create significant prosperity by looking no further than our orbital backyard: cislunar space. In 2018, United Launch Alliance (ULA) outlined a vision for 1000 humans to be living and working in cislunar space by the year 2047, and estimated that the yearly revenue of this economic zone would approach 3 trillion USD. For comparison, Earth's global energy market is currently valued at 7 trillion USD a year [63].

This monumental growth of the Moon's economic importance over the next 30 years is an opportunity that should not be ignored by actors which command the necessary capital to breach the high barrier of entry that space development presents. Space is capital-intensive on all fronts: financially, intellectually, technologically, and socially, among others. To a large extent, the institutions able to meaningfully participate at the edge of space development are mostly governmental space agencies, who lead the way by investing (often at a loss) in key technologies and infrastructure that enable the organic growth of private markets. A historical parallel is the public development of railroads or highways that allow small towns to sprout up alongside it, driving trade and development.

This was the case with LEO (Low Earth Orbit) and GEO (Geostationary Orbit): in the last decade, the commercial LEO market experienced an average CAGR (Compound Annual Growth Rate) of 17% through, partially, the growth of the processing of remote sensing and communication data by software companies in order to add value downstream in the value chain (e.g. by processing rural remote sensing data into useful agricultural insights) [63]. If these companies had been forced to invest in satellites, launchers, and expensive hardware testing facilities, they would likely not have been able to enter the market and survive. The barrier of entry was lowered through years of sustained innovation and launch cost reductions both by corporations and governments, and this allowed the creativity of the market to bloom into bottom-up value creation.

Human development in cislunar space is going to undergo a similar process, but now, at the beginning of the 2020's, we are at the same point LEO was at in the 1990's: still no existing infrastructure (the ISS had not yet been built), but with a clear recognition of the need for it. Plans for cislunar infrastructure have surfaced independently in different public blocs of space capability: NASA, ESA, and ROSCOSMOS-CNSA. Since the customer of the architecture proposed in this report is ESA, the analysis of opportunities and policy recommendations will be conducted from this agency's perspective in this new space race aligned along different axes than the last one.

This chapter will treat the role of the Moon and its orbital points of interest in the cislunar economy, the potential markets in cislunar space, the importance of lunar crewed surface infrastructure within this system and how our architecture can capture some of the aforementioned markets. All of the costs of our proposal will be discussed in Section 18.1, while its profitability can be found in Section 18.2. Further policy recommendations for ESA are then treated in Chapter 20.

2.2. The Cislunar Key to Space

Developing cislunar space will potentially be the single most significant phase of human space development due to its effect on Earth and our access to the surrounding solar system. As can be seen in Figure 2.1, the consequences of making propellant, construction material, and volatiles available in orbit has consequences on the rest of the system which are difficult to explain linearly because of the many feedback loops of the network. This section will focus on some of the turnkey aspects of this system, how they benefit humanity from an existential perspective, and what role the Moon specifically plays.

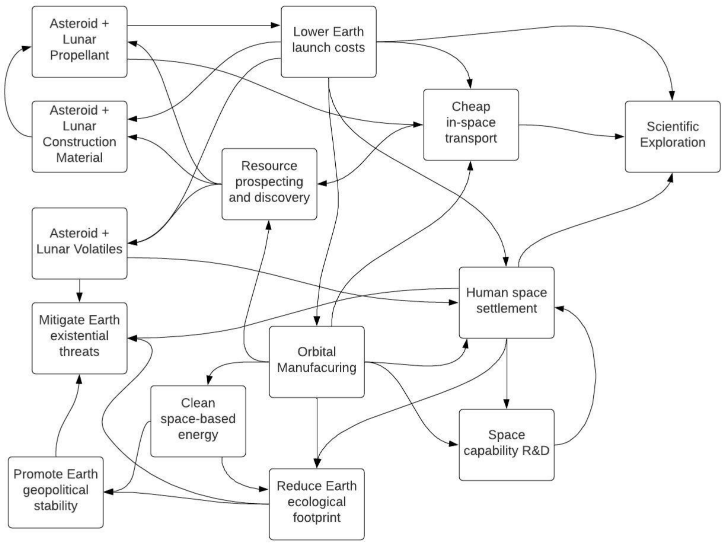


Figure 2.1: Conceptual graph of some effects of cislunar development on Earth and the system itself [own work]

2.2.1. Benefits for Humanity

Looking at Figure 2.1, in the bottom left corner are some high-level effects of space-based resources on our condition on Earth. All of them point to mitigating existential threats to the human species through several different mechanisms. In the short term, the exponential growth of energy consumed by humanity for computing, heavy industry and transportation will at some point exceed our capability to burn fossil fuels or produce renewable energy generation infrastructure quickly enough. This demand can be met through space-based solar power where PV (photovoltaic) satellites beam collected solar energy down from LEO directly to terrestrial consumers - note that the implementation scale needed for a substantial impact on our collective energy consumption budget only becomes possible if the construction materials and manufacturing capabilities are located outside of Earth's gravity well. These two, in turn, are enabled by the lower launch costs made possible by lunar and asteroid ISRU propellant [7].

In the mid-term future, similar dynamics apply to the supply of space-based volatiles (such as fresh water) and manufactured products, with both inputs serving as pressure valves for the increasing resource scarcity on Earth. This scarcity not only drives humanity to ever more destructive extractivism here on Earth, but also fuels geopolitical tensions. In a similar vein, progressively moving heavy industry (specially the construction of space habitats and vehicles) off of Earth and into orbit releases pressure on the Earth's biosphere, buying us time to sort out our climate-related problems. One last existential risk mitigation effect that is not mentioned in Figure 2.1 is the development of much more advanced Space Situational Awareness technology due to increasing challenge of performing space traffic control duties for the thousands of present and future satellites in orbit - this makes us much more capable of detecting asteroids and comets that could collide with our home planet and take action before disaster strikes.

Considering even longer time frames than in the aforementioned examples, the risk of overpopulation and subsequent resource depletion on Earth is one that will become increasingly possible to mitigate with the construction of human settlements in space and the emigration of large populations to orbit. The consequences of extinction-level catastrophes befalling our planet are also attenuated, through autarkic human space settlements, from total annihilation to a mere setback in development, due to the preservation of knowledge and technology on some other settled planet or moon. However, human space settlement lies far in the future because of the significant technical challenges that remain to be solved concerning sourcing the construction material for such structures, developing in-orbit manufacturing capabilities, lowering Earth launch costs enough to where the average person can afford the ticket, and establishing the safety of human habitation in reduced-gravity and irradiated environments. The last point is one that can only be solved empirically by actually settling low-gravity locations, and this capability is one of the main benefits the H.O.M.E. proposal offers on a long time frame.

In short, the benefits of cislunar development for Earth are numerous, exist both now and in the future, and are important to the very survival of our species in several different ways. Developing the capability of crewed space infrastructure brings us one step closer to becoming a multi-planetary species.

2.2.2. Unlocking Cislunar Industries

The main constrain on the cislunar economy, in terms of capital barriers of entry, is currently the prohibitively high cost of launching payloads into space from Earth. Eliminating this constraint opens up the rest of the system for growth.

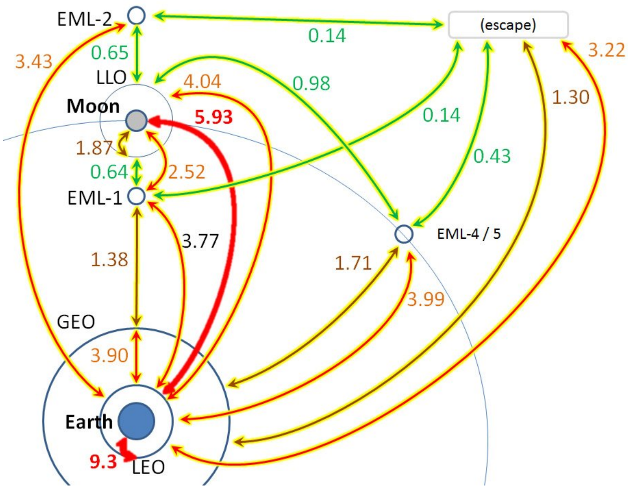


Figure 2.2: Map of delta v requirements between all cislunar destinations

As can be seen in Figure 2.2, the energy required to access LEO from the Earth's surface is larger than travel between any other two points on the graph, and this segment alone accounts for 60-75% of the ΔV (the energy needed for space travel expressed as a velocity difference) needed to reach most cislunar locations from Earth. This is due to the exponential growth of the required propellant needed to accelerate a certain payload by some ΔV, and one of the unfortunate effects of this is the drastic reduction of delivered payload in locations such as the Moon's surface compared to LEO along with a steep increase in launch prices. However, cutting this growth by regularly refuelling our vehicle along a given path allows us to linearize the propellant-ΔV curve, as can be seen in Figure 2.3 [1].

This linearization can cut the cost of reaching EML1, LLO and LS by up to 90% and initiates the cascade of effects described in Figure 2.1: enabling cheap in-space transport, allowing for orbital manufacturing to profitably scale and increasing the rate of space resource prospecting, among others [100]. This is the desired commercial growth of cislunar development that can sustainably drive the expansion of human influence without further destroying our home planet's ecosphere.

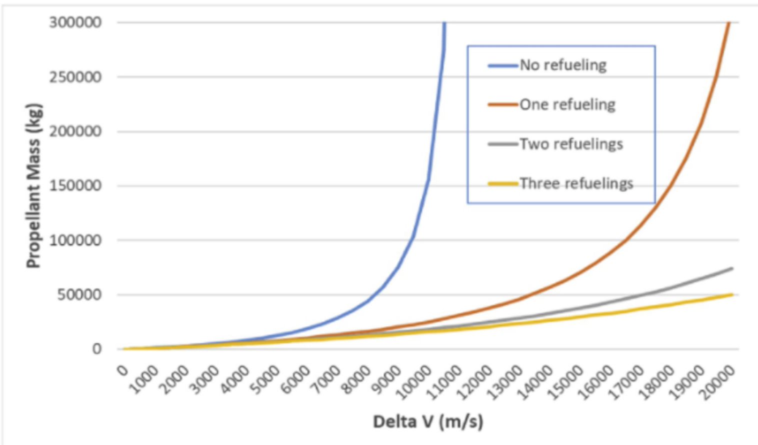


Figure 2.3: The effect of refuelling on the growth of required propellant with growing ΔV [100]

2.2.3. The Moon as a Central Node

The Moon, its orbital locations, and the Lagrangian points of the Earth-Moon system will be the most important destinations in space for the coming few centuries, and infrastructure built there will have the greatest impact on space development, as will be explained in this section. Another look at Figure 2.2 reveals how much less

[1] URL: https://pbs.twimg.com/media/CIlRuBzUYAQ2S9q.jpg

ΔV is needed to travel between cislunar orbits than to reach LEO from Earth - in the cases of the EML points, it is almost an entire order of magnitude. To complement the data in Figure 2.2, it should be added that the ΔV required to reach potentially valuable Near Earth Objects from EML1 is between 0.5-2 km/s, and reaching a Mars transfer orbit from there is less than 1 km/s (as opposed to 4.7 km/s from LEO). This means that supply chains and mission stagings done completely in space can be executed much more quickly and cheaply than interfacing with Earth [63].

To illustrate the effects of cislunar staging and refueling on interplanetary travel, consider the propellant growth curve of several refuelings in Figure 2.3: reaching Mars' surface, which is a total of 18.91 km/s from Earth, is already in the zone past the exponential growth's inflection point for a single refueling on Earth. If refuelings could happen in LEO and EML1, the propellant requirements would be reduced to be point of feasibility, enabling much larger payloads to be delivered with roughly the same amount of propellant as a single launch to LEO would normally require. The same is true for other planets, as well as the Main Asteroid Belt between Mars and Jupiter. What these propellant requirement decreases show is that cislunar staging and refuelling depots can make interplanetary exploration as cheap as cislunar activities - they will be the central transport hubs of our future as an interplanetary species.

However, it is not only in the long term that cislunar transport nodes are important. It can be argued that the space economy of the future will be concentrated around Earth, decreasing in population and trade volume as one travels further and further out. This is because markets do not exist without end consumers, and it is very likely that the vast majority of our descendants will continue to live on Earth, at most venturing out into settlements in Earth orbits or at EMLs. Irrespective of our attempts at settlement and terraforming of planets and moons, there will never be a more ideal environment for our species than Earth itself, and recreating it is impossible with the technology we currently possess. This forces the locus of space markets to return to cislunar space (where it has easy access to Earth markets) without regard for the origin of traded goods. Making these resource flows as cheap and frictionless as possible is paramount, and it all begins with lowering launch costs from Earth by making cheap propellant available throughout cislunar space.

2.3. Drivers of Demand

Now that it has been shown how important cislunar development is for our future on Earth and in space, the sources of demand for services in this economic zone can be analysed in order to yield the markets that can be explored by the European space sector through ESA incentives. Figure 2.4 shows how different actors and activities create demand for further goods and services. The two green block represent the main customer and architecture proposed in this report, namely ESA and crewed lunar infrastructure. The two blue blocks show the most central nodes in the system, a property which can be estimated preliminarily by counting the amount of connections to other nodes in the graph, as well as how many nodes are within a maximum of two connections away. The outcome of this analysis makes intuitive sense, as transport and propellant (which is gotten from asteroid and lunar ISRU) are needed for all space activities, meaning there will always be an inherent demand for them underlying space development.

Before treating the markets resulting from this self-reinforcing network of generated demand, the specific demand drivers for governments and private actors will be analysed and connected to the specific context of this project.

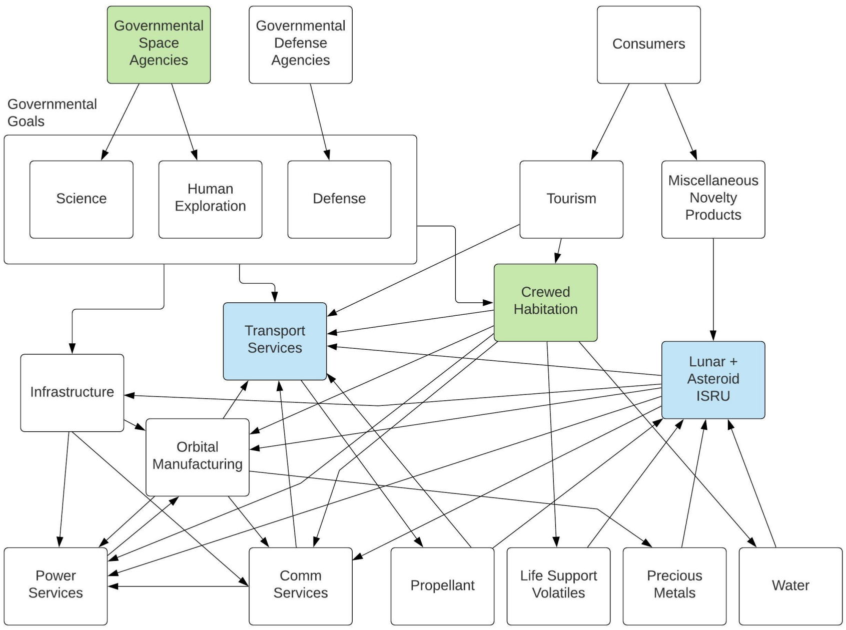


Figure 2.4: Conceptual graph of the drivers of service demand in cislunar space

2.3.1. Governmental Demand Drivers

The role of governments, aside from regulation and defense, is sometimes doing things that are not immediately profitable for ulterior reasons. Of course, they do not like to take sustained losses, but the liquidity needed to take such risks in novel ventures or vital infrastructure is restricted to supranational corporate conglomerates, ultra-rich billionaires, and governments.

The three main kinds of ”transcendent” activities which governments finance in space are science, human exploration, and defense. Of course, these three can yield great returns both financially and otherwise, but they are inherently risky and can feature a phase of sustained investment before any returns appear. [18]

Science

The demand for chemical, physical, and biological measurements from remote sensing orbiters and increasingly, surface samples, has been growing across the scientific community in all the main spacefaring powers. These serve the missions of the search for extraterrestrial life, furthering our understanding of the origin of planets and moons, and the prospecting of resources that could be exploited. Additionally, the astrophysical investigation of the cosmos is projected to be greatly enhanced by the radio silence, and protection from Earth’s electromagnetic interference, that can only be found on the Far Side of the Moon. Alongside this is the novelty of reduced gravity environments and the technological research and development, as well as purely scientific experiments, that can only be conducted in such conditions.

The results of these scientific investigations have the potential to blossom into new opportunities for the private sector as well as further our understanding of nature and our capabilities in the space environment. Governments invest in science because it pays off handsomely in one way or another.

Human Exploration

This driver of the demand is the most transcendent of the three due to its origin from, and impact on, the human psyche. The difference between having a probe visit some faraway world and personally setting foot there is probably so substantial in our collective consciousness due to the fact that personally exploring some hidden site on Earth was the only way of doing it until very recently. We have not adapted our impressions to the reality of robotic exploration, but an unintended positive outcome of this drive to go further is the necessary development of human-rated habitation for these extreme environments waiting to be set foot on. This means that crewed exploration, of the kind meant to build national prestige, blazes the trail for later, more pragmatic reasons to have people living off Earth.

2.3.2. Why Crewed Infrastructure?

Disconsidering the spiritual aspect of human exploration, are there any operational benefits to having crewed habitation? Yes, there are: fully robotic operations, whether they are autonomous or not, have a certain tolerance to failure and risk mitigation policy that the machinery has been designed for. However, all of these robotic architectures are inherently limited in their ability to resolve crises or critical failures that are found only on the tail-end of the probability distribution. Astronauts, on the other hand, are selected to be as knowledgeable and versatile as possible in the face of disaster - in effect, they reduce the risk of failure by covering the most extreme events through their ability to creatively respond to adversity. Alongside this coverage of "black swan" contingencies, a human presence also facilitates the maintenance, repair, and evaluation of critical systems.

This redundancy provided by a crewed infrastructure unfortunately comes at the cost of designing for human survival, which costs mass and complexity in the form of life support systems, emergency escape options, consumables, etc. However, these drawbacks are more than made up for by the knowledge gained through our continued physical presence in space - every astronaut we send to the ISS is another small lesson learned in human spaceflight. One example of this is Scott Kelly's entire year spent in the ISS and our resulting understanding of the effects of extended microgravity on the human body [2]. This is one of the main motivations behind the H.O.M.E. proposal: seeing if humans can even survive on the Moon for at least a year.

Defense
The essential motivation for a government's defense of its resources or vantage points can be summarized as Security of Supply, the supply here being of some vital good or tactical access. A good example of this is LEO, where a quiet arms race in satellite offensive and defensive technologies is happening between the USA and Russia [3]. The supply in this scenario is communication capabilities as well as a vantage point over the entire planet.

In the case of the Moon, a likely reason for military installations is to secure the supply of cheap propellant for internal operations (which are sold to opponents for a premium, of course). In later phases, the extraction of lunar minerals will have direct effects on orbital manufacturing, space-only vehicles, and a large potential industry of orbital ballistic weapons - all of which point to several key supplies which must be secured.

Even though military spending on cislunar space is small now, it is likely to grow significantly over the coming decades as the new Cold War spills into orbit [4].

2.3.3. Private Actor Demand Drivers

The two categories of private actors are consumers and businesses. As can be seen in Figure 2.4, consumers drive demand for tourism and miscellaneous novelty products resulting from cislunar activities in general. These will be discussed in the following chapter. Businesses were not included as a separate block in Figure 2.4 because they both create and fill the demand for all the activity nodes. These markets will have to be analysed to discover their dependence on prerequisite services and technologies, as well as their likely emergence period being within H.O.M.E.'s mission timeframe of 2030-2040.

A final note on private actors are that a limited number of them, both people and corporations, command capital that is comparable to the budgets of governments e.g. Jeff Bezos investing 1 billion USD every year into his rocketry company Blue Origin. These should not be ignored, but in the analysis of markets and their profitability, the average consumer/space company is considered rather than the outliers.

2.4. Cislunar Markets

Figure 2.5 shows some of the main markets that currently exist and can be incentivized at each location in cislunar space. These markets will be combined with the results of Figure 2.4 and filtered with a number of criteria in order to leave only the ones that are directly captured by crewed surface infrastructure, are viable in our mission timeframe, have returns significant enough to offset the capital expenditure our proposal requires, and has enough data on them to allow a more detailed profitability analysis. All markets approved in this section will be discussed in further detail in Chapter 18.

[2] URL: https://www.sciencenewsforstudents.org/article/how-year-space-affected-scott-kellys-health
[3] URL: https://www.thespacereview.com/article/4180/1
[4] URL: https://www.thespacereview.com/article/4176/1

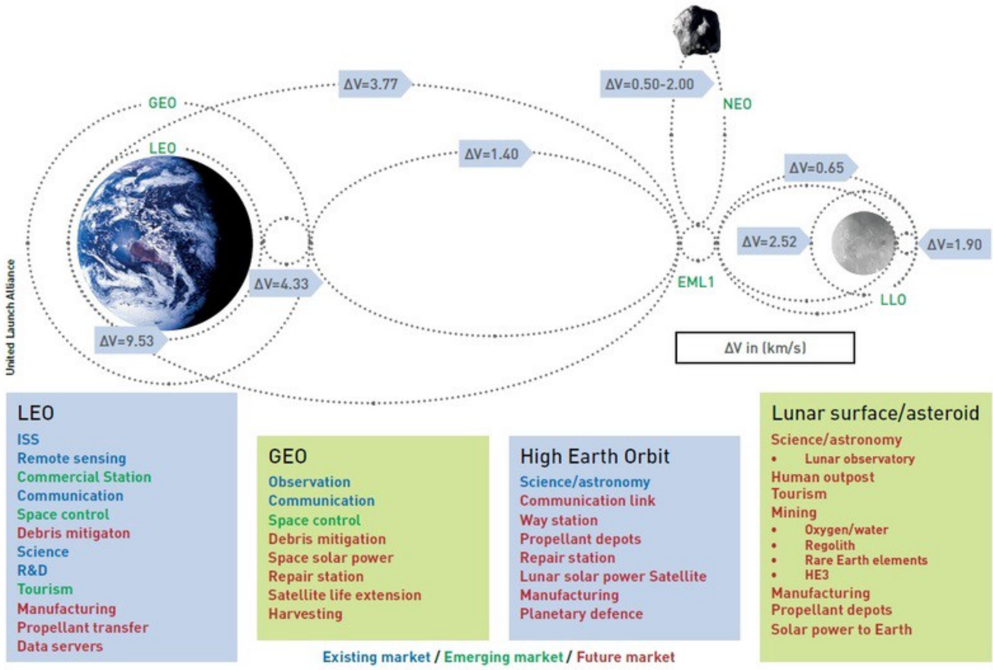


Figure 2.5: Map of cislunar space and the markets associated with each destination [5]

2.4.1. Potentially Captured Markets

Due to the limitations of this report, markets which are derivatives of crewed lunar surface infrastructure will not analysed - only flow in the cislunar economy which directly pass through the lunar surface and are affected by the habitat are considered [18].

The **human outpost** market is the main activity of H.O.M.E., but it is not meant for commercial purposes in its initial phase - only ESA astronauts are planned to inhabit it over its currently envisioned 10-year lifespan.

Scientific experiments, which have reserved floor space in our habitat design, are already a considerable opportunity that is being capitalized on by the ISS, which is conducting several dozen experiments at any one point for a combination of public and private clients. However, finding the financial reports of these payments for access to microgravity was unsuccessful, and no further methods for estimating the expected returns from scientific hypogravity experiments was discovered. It is also probable that the returns from the volume and power budget H.O.M.E. allocated for experiments will not make a substantial dent in the mission's financial statements. This market will therefore remain mentioned without further investigation.

The beaming of **solar power** to Earth markets is not viable in the coming few decades due to the significantly higher price that will have to be charged for lunar energy in order to regain the high initial investment [18]. Lunar power is most useful to lunar operations, and its generation infrastructure is mostly built on a per-mission basis. Short-to-mid term revenue in this market could only result from selling the excess power generated by H.O.M.E's solar panels, which has the double problem of few customers (which need to be in the immediate vicinity of the habitat) and the low probability of having sufficient excess power to sell for significant returns.

Tourism is a possible contender for a secondary purpose for the habitat, but since the structure is being designed for a maximum of four astronauts and is planned to be continuously occupied over its ten-year lifespan, this makes accepting guests more difficult. This can be mitigated through some rearrangement of the internal layout of the habitat and restricting the stays of tourists to a few days at most, but it is ultimately easier to add an extra structure to the lunar compound and reserve that for paying customers.

However, it would be morally irresponsible to do so before establishing whether extended lunar habitation is even possible or safe, which is one of the main goals of H.O.M.E. - the astronauts being sent to inhabit it will be well aware of the inherent risks which are still present despite the engineers' attempt to mitigate them. Exposing paying customers to these risks before solidly understanding of how to minimize them will almost certainly invite disaster, loss of confidence from the public, and the consequent preclusion from engaging with this market in future.

[5] URL:https://medium.com/life-on-the-other-planets-whats-new/creating-a-viable-cislunar-economy-eb37482f88fc

The **mining of metals, Rare Earth Elements, and He3** is one of the most sought-after phases of lunar development due to the systemic impact this activity will have [18]. Since part of the construction of H.O.ME. involves excavating and moving large quantities of regolith, there is a possibility of repurposing the already planned machinery for mineral extraction. The metals present on the Moon are useful for construction, power infrastructure, and many other industrial uses [6], while REE's (Rare Earth Elements) are extensively used in delicate technology and power storage systems (the demand for these is growing steadily on Earth due to depletion of reserves and a Chinese monopoly on supply). Helium-3, potentially a fuel for fusion reactors, will become incredibly valuable if aneutronic fusion is ever solved [95], but exists on the Moon in minuscule concentrations.

The problem with these three markets is twofold: firstly, there are no current or projected (over 2040) customers in space for any of these products. This could, however, indicate a chicken-and-egg situation of reciprocally generated demand which only needs an initial spark i.e. if you provide supply, demand will rise even though there was no initial collective awareness of the reason for that demand [7].

The second issue is partly illustrated in Figure 2.6, which shows how the technology to mine, separate and process lunar products grows from simpler to more complex regolith products (e.g. from raw regolith to purified metals). Water is the easiest to extract, while metals and relatively scarce elements (He3 concentrations are measured in parts per billion) each need their own refinement process and pipeline, many variants of which are low-TRL (Technological Readiness Level), expensive, and resource-intensive in operation. For these reasons, any regolith products more complicated than water and raw regolith will not be considered in the market analysis.

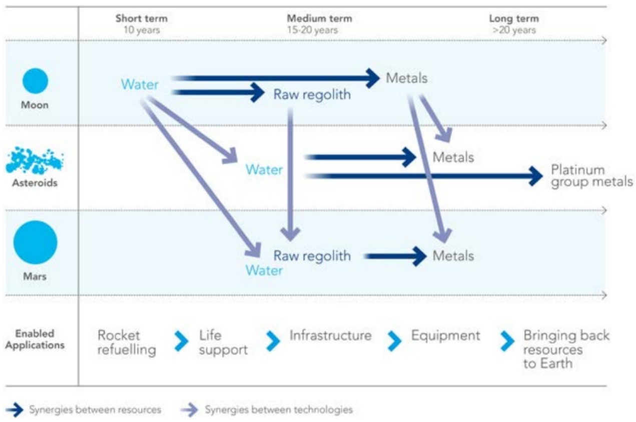


Figure 2.6: Graph of the evolution of ISRU technologies between the Moon, asteroids, and Mars [13]

Manufacturing on the lunar surface is a capability that will need to be built up as cislunar development proceeds, but in the case of H.O.M.E., it is unsure what products could be manufactured out of the locally available (and sufficiently refined) materials, and for which customers. The manufacturing capability inherent to the architecture proposed in this report is limited to bags of packed regolith, since all other products, consumables, and structures are brought from Earth.

Nearly nothing can be manufactured out of purely unprocessed regolith (except for filled bags and other powdery structures), but even simple hardware additions, like a smaller solar sinterer and additive manufacturing unit, could enable simple structures, parts, and shapes to be produced. However, the material properties of sintered or otherwise additively manufactured regolith products are at current insufficient for most of their potential uses. The low TRL, system complexity, and resource demand are limiting factors for this operation that also eliminated this technology from being used in the construction of the main habitat.

Additionally, manufactured products should be optimized for value per weight, e.g. gold and electronics have high specific costs. Early lunar products will be bulky and not very dense, e.g. cast basalt or glass fiber reinforced concrete, making it difficult to deliver products to the consumer market mainly located on Earth at competitive prices. Since there will initially be a local lunar scarcity of precursors needed for high-tech manufacturing and the mining of precious metals, it is unlikely that manufactured products will be competitive until higher levels of development are reached.

One market that is exempt from the value-per-weight limitation is **small luxury and novelty lunar products**

[6]URL: https://sites.wustl.edu/meteoritesite/items/the-chemical-composition-of-lunar-soil/
[7]URL: https://rationalstandard.com/demand-and-supply-is-no-chicken-or-egg-problem/

which have inherent worth due to their scarcity on Earth. The issue with the source of their perceived market value is the inherent limitation it places on yearly sales and revenue before the supply is inflated (relative to the small demand for such niche products) and prices fall accordingly. In other words, there are only so many consumers willing to pay for luxury lunar products like jewelry, trinkets, rocks, and regolith - when enough products have been shipped back to Earth to fulfill all the demand, any additional shipments will begin to dilute prices and decrease profits in future. Any revenue made without incurring these effects will likely be in the order of magnitude of millions of euros, not the billions necessary to offset mission costs [18].

A historical parallel is the Spanish discovery and large-scale mining of silver and gold in South America: since these metals were literally used as money at the time, they thought to have secured a source of inexhaustible wealth. Upon bringing their pile of precious metals back to Europe, the relative scarcity thereof plummeted, causing a major inflationary crisis across the continent, and ruined the long-term economic prospects of the Spanish [8]. Space developers would be wise to avoid similar mistakes in the economic exploitation of the new ”8th continent” that the Moon presents.

2.4.2. Markets with Analysis Potential

The only significant and potentially captured market not yet mentioned is the **mining of water from regolith** and electrolysing it for the **production of propellant** in the form of LH2/LOx. As can be seen in Figure 2.4, the two blue blocks, Transport and Lunar ISRU, are central to the cislunar system of supply and demand. All other activities in space inherently require propellant, and the demand for it already exists at significant levels in several different orbital locations. There is also a considerable body of data and estimations on this topic, allowing for models to be created for the estimation of potential profits therefrom [63], [100].

One example of such projections can be seen in Table 2.1, which shows ULA's forecasts for the demand for water and propellant at several cislunar locations over the next few decades. These values attempt to capture the demand driven not only by planned space agency missions and the supporting infrastructure e.g. NASA's Moon-to-Mars and Lunar Gateway, but also the organic growth of commercial demand due to the nonlinear market effects shown in Figure 2.4.

Table 2.1: Propellant and water projections in cislunar space for the next few decades [63]

Propellant & Life Support Water per Year (MT)	2010	2025	2040	2055	2070
LEO Depot	2	433	2385	3096	23320
EML1 Depot	0	425	3133	5534	43158
Moon Surface Depot	0	13	482	771	4665

The selection of lunar propellant production for revenue generation in the short term aligns with ESA's Strategy for Space Resource Utilisation [13], in which one of the assumptions underlying the strategy is:

> ”*The primary use case for space resources in the foreseeable future will be propellant with life support consumables as a secondary use case that alone will not justify the systems or investments needed for space resources. Materials production may follow later if a use case emerges, most likely related to lunar surface infrastructure.*”

Furthermore, it can be assumed that by 2030, the first planned year of operations for H.O.M.E., sufficient progress will have been made in the development of the needed ISRU technology and in the prospecting of lunar resources. This assumption stems from some of ESA's stated goals for 2030 [13]:

- ”*Identification and characterisation of at least one deposit of polar ice.*
- *End to end demonstration of the production of water or oxygen at the lunar surface from locally sourced materials.*
- *Space resources integration planned for in reference international exploration architecture with early demonstrations defined.*”

A more detailed analysis of the profitability of a lunar propellant production facility and its effects on ESA's spending on H.O.M.E. is conducted in Section 18.2.

[8]URL: https://www.worldhistory.org/video/2537/the-spanish-empire-silver--runaway-inflation-crash/

3. Functional Breakdown and Mission Requirements

3.1. Mission Requirements

To start with the generation of the requirements, a brainstorm was held from which a requirement discovery tree was established. This can be seen in Figure 3.1. The tree was branched into different parts to divide the mission into smaller parts. The first division encompasses the difference between performing the mission technically and performing it within the constraints. The former was defined as what is necessary to get the habitat on the moon with all working equipment to facilitate the astronauts survival. This was then further divided into the habitat itself and the mission of getting the habitat on the moon. The constraints were seen as what endangers this survival but also social, political and economic aspects which need to be considered in the design. [38]

These options then flow down into the various mission segments for which requirements were made to ensure a complete and safe mission. A complete list of requirements per section can be found in each relevant chapter, for clarity. In addition to that, a compliance matrix will show whether the requirements are met or not together with a justification for this.

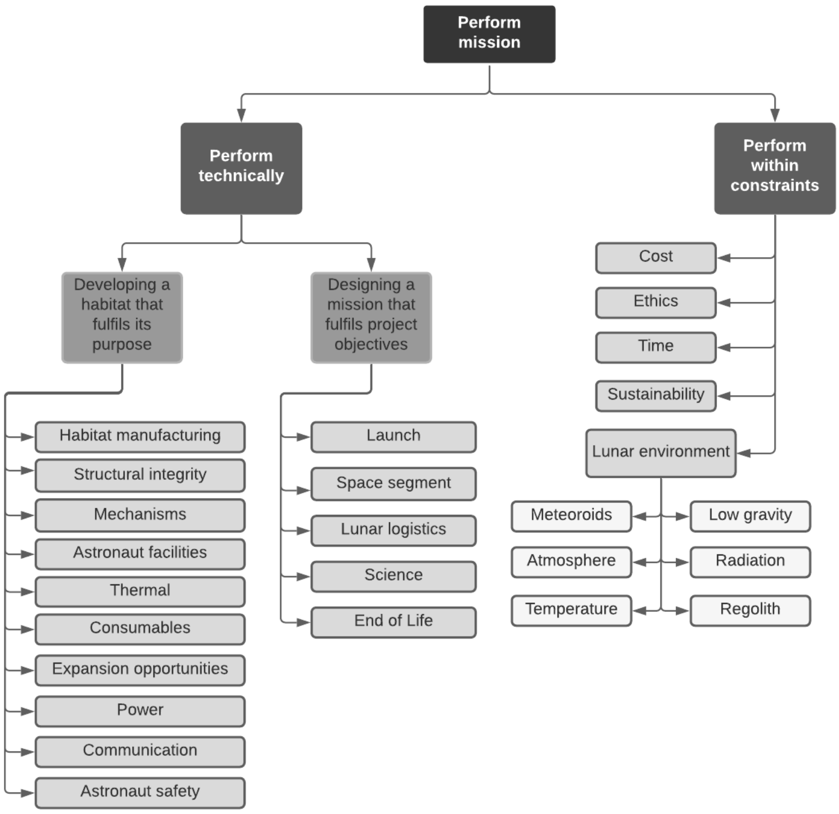


Figure 3.1: Requirement discovery tree

3.2. Functional Diagrams

3.2.1. Functional Flow Diagrams

The mission's Functional Flow Diagram (FFD), illustrated in Figure A.1 and Figure A.2 in Appendix A, outlines the different system and subsystem functions in chronological order and is divided into distinct top level mission phases. The chosen phases following the order of the mission itself are:

1. Finish the production plan
2. Production of the required machines, habitat parts and sup-parts, acquisition of the material
3. Astronauts selection and training
4. Launch 1: Safe house, manufacturing machines, solar farm etc...
5. Transfer orbit of launch 1
6. Landing of launch 1
7. Cargo 1 transportation
8. Start of the manufacturing (setting up solar farm, excavation...)
9. Launch 2: Main habitat, internal transporter, internal subsystems and life support
10. Transfer orbit of launch 2
11. Landing of launch 2
12. Cargo 2 transportation
13. Setting up of the main habitat, stacking of the regolith bags, setting up of the subsystems....
14. Operations and testing
15. Launch 3: Astronauts and their rover
16. Transfer orbit of launch 3
17. Landing of launch 3
18. EOL procedures

3.2.2. Functional Breakdown Structure

The mission's Functional Breakdown Structure (FBS) encompasses all the different functions the mission should perform by giving a detailed overview of the different levels. Such detailed analysis assists in the process of meeting the set requirements. Starting with the same top level phases as the FFD and by going down to a lower level, more details about the previous level can be found. The numbering flows from the FFD and since the latter has significantly less detail than the FBS, extra decimals in the numbering were added where needed. [38] The mission FBS can be visualised in Figure A.3, in Appendix A

3.3. Operational Flow

For the mission to be completed successfully there are many operations that should be performed. An overview of the operational flow for this mission was first introduced in the midterm report [39] and can be found again in Figure 3.2. It highlights the operations that should take place at the different mission phases: transport, operations and End of Mission.

Since multiple launches will occur the boxes specifically for the final launch with the astronauts are highlighted in grey in the pre-operational phase. For this final launch the habitat construction section is also no longer required as the habitat life-support shall be up and running following from requirement.

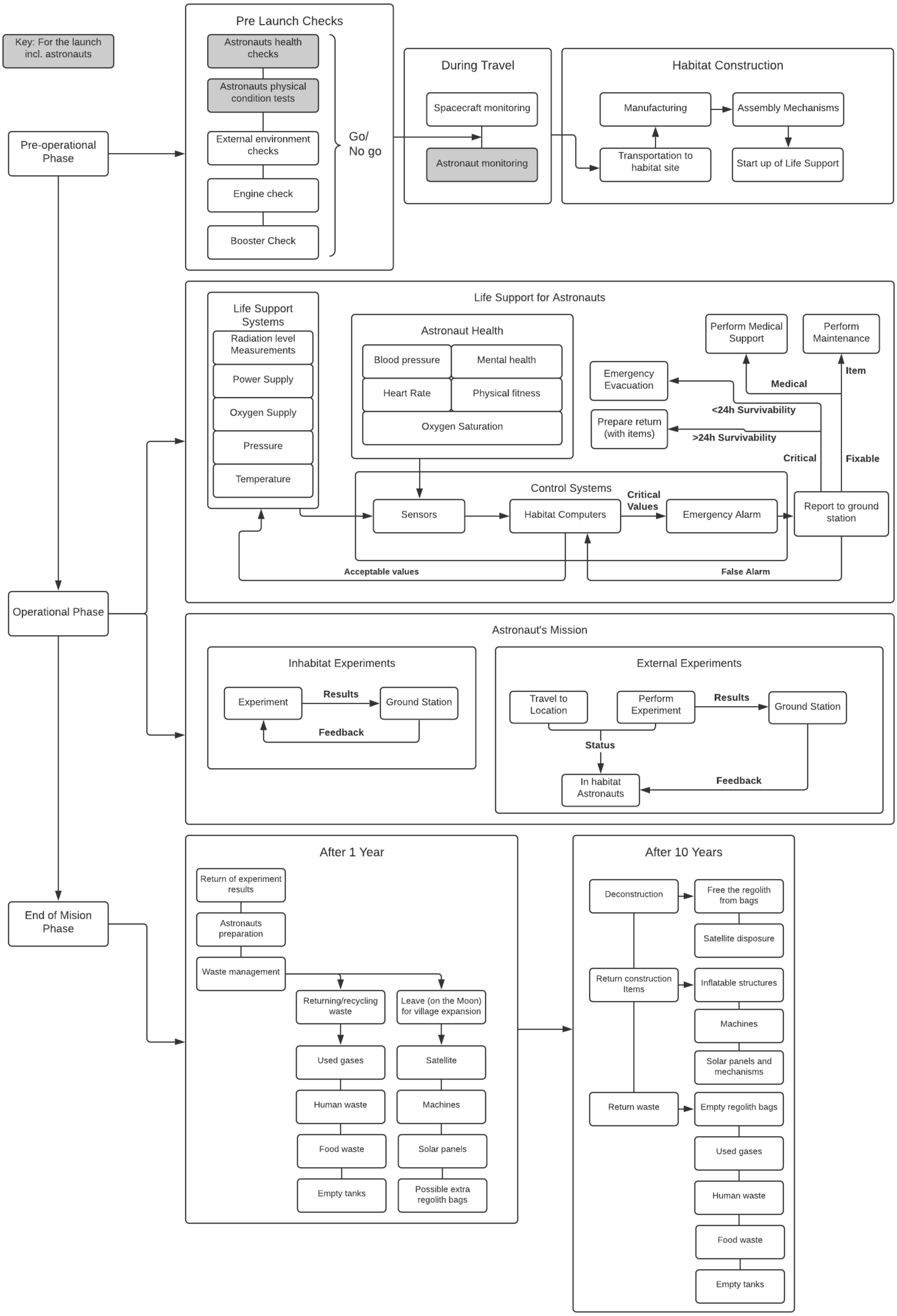


Figure 3.2: Flow diagram visualising operational flow of the system

4. Astronauts’ Health in Space

When performing such mission to space, it is inevitable that astronauts’ health will be at stake due of a large number of factors influencing both their physical and mental health; an example of such would be the risk of cancer and degenerative diseases when exposed to radiation[1]. Studies on outer space radiation, isolation and lack of gravity are being done so that future missions be designed in such a way that these effects be minimised.

For this mission, the different factors that could potentially harm the astronauts are the following:

- Galactic cosmic ray (GCR)
- Solar particle events (SPEs)
- Meteoroids
- Isolation
- Lack of gravity
- Living in an extreme environment

The effect of these factors will be minimised by making sure the astronauts are trained properly, are physically and mentally ready to go into space, and by directly protecting them from such events. To account for major malfunctions or fatal events, a safe house has been designed as well. This safe house design will be further elaborated on in Chapter 12. The way such factors have been taken into account during our mission design and the possible effects they have on the astronauts’ health will be discussed further in the upcoming sections. To start with, the effects of radiation hazards will be discussed in Section 4.1, together with the mitigation taken into account during the mission design. After this, the same will be done for micro-meteoroids in Section 4.2. Then, the effect of hypogravity on the astronauts’ health will be analysed in Section 4.3. After that, the impact of Moon’s extreme environment will be analysed in Section 4.4 and to finish with, the effect of isolation, when in space, on the astronauts mental health will be discussed in Section 4.5.

4.1. Radiation Environment and Protection

Radiation is one of the, if not the, biggest hazard for interplanetary travel. There are many factors in play when going to space but radiation is unavoidable. It causes health risks to the astronauts but also has an influence on sensitive electrical instruments which can go haywire. In this section, the different types of radiation and their dangers will be discussed and how they will be mitigated in the design. After that, a discussion will follow on the use of SPENVIS (Space Environment Information System) which is operational software from ESA and maintained by the Royal Belgian Institute for Space Aeronomy[2]. This programme was used to verify the designed protection of the habitat regarding radiation.

GCR’s and SPE’s

To start off, a clear distinction needs to be made in the different types of radiation present in space. The two main types are Solar Particle Events (SPE) and Galactic Cosmic Rays (GCR). The trapped particles in the Van Allen radiation belt around the Earth and gamma ray bursts are not considered due to the planned fast traverse through that belt and the small possibility of such occurrence, respectively. The SPE’s are caused by events or disturbances of the Sun as the name suggests. Think of Solar flares and Coronal Mass Ejections (CME) [96]. They cause a high flux of mainly low energy protons to be sent everywhere in the solar system with a defined source, the Sun. The energy of these protons is only in the range of below a few hundred MeV but due to their high fluxes they can cause serious dangers to the astronauts [81]. Not only increasing their chance of getting cancer but it can also have immediate effects like Acute Radiation Syndrome (ARS) or in extreme cases even death. Luckily, due to their low energy, SPE’s are rather easy to shield against[3]. The duration of such an event can take from a few hours to multiple days. They occur almost at random and are very hard to predict [67]. However, the intensity of Solar activity, and thus also the SPE’s, depends on the period in the solar cycle. This activity reaches a maximum every 11 year in which additional and more intense solar flares and CME’s occur. Currently, cycle 25 has just begun and will reach a maximum in 2025 and a minimum somewhere in the beginning of the 2030’s as seen in Figure 4.1.

[1]NASA: Space radiation risks https://www.nasa.gov/hrp/elements/radiation/risks

[2]https://www.spenvis.oma.be/ [cited on 17/06/2021]

[3]https://www.nasa.gov/audience/foreducators/topnav/materials/listbytype/SF_Radiation_Challenge_HS_Mod1.html[Cited on 17/06/2021]

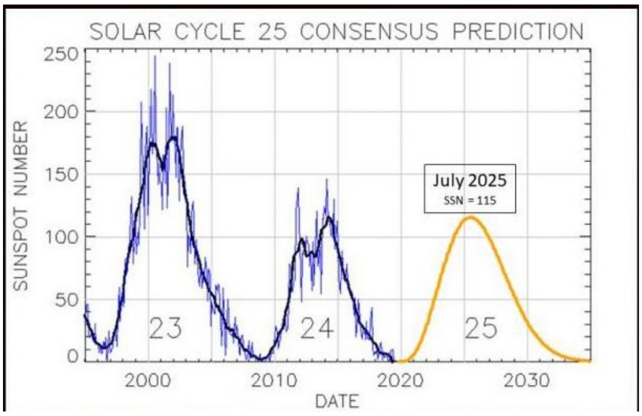


Figure 4.1: Prediction of solar cycle 25 by NOAA [4]

Solar cycle 20 was a remarkable one. In the year 1972, there was a solar storm which took place in August. Not only were there large disturbances in the electrical grid of North America and also the satellites circling the Earth, but during that same year, both Apollo 16 and Apollo 17 had taken place. It was only by shear luck that there were no astronauts outside the magnetic field and the atmosphere of the Earth. If this had occurred during one of the missions, an emergency return mission would have been the best possible outcome if the astronauts even survived in the first place [71].

Galactic Cosmic Rays are different from SPEs in quite a few perspectives. First of all, GCRs come from outside the solar system and most likely originates from explosions of supernovae. This background radiation is isotropic as it comes from all directions and in equal proportions while being fairly constant over time. Their energy is a lot higher with peaks up to 20 GeV[107]. They do not only contain hydrogen ions (85%) and helium ions (14%) but also high-energy nuclei with charges ranging from Z=3 (Lithium) to Nickel (Z=28) (1%-2%) [96]. This highly charged transition material like iron (Z=26) are very hard to shield against and can penetrate up to several centimetres of tissue. Another difficulty about this type of radiation is the occurrence of secondary radiation when it collides with the walls or shielding material. Especially heavier elements produce this secondary type which lead to high penetrable neutrons[81]. This makes it difficult to shield against and in some cases, it turns out better not to shield against them than only having a light shield[67]. In contrary to SPE's, this background radiation, although being fairly constant in intensity, is inversely related to the solar activity. During high solar activity, the heliospheric magnetic field strengthens and deflects incoming GCR's. Although of less immediate danger due to their lower flux compared to SPE's, GCR's become the main concern during longer missions. Their extreme difficulty to shield against is currently one of the biggest hurdles for human interplanetary travel[15].

Health Issues and Use of Units

Although it is well known that radiation causes health issues, it is often unpredictable how it will affect a specific individual. When ionising radiation permeates an atom, it can change the atom's charge[5]. When such an ion does this to a cell, there are multiple options. If the flux is high enough, it can directly kill the cell. In other scenarios, it changes the DNA stand breaks [17]. There are immediate (acute) effects or long term ones due to more and more mutated DNA. This can cause increased cancer risks and cardiovascular disease. While it is difficult to predict its precise effects, more exposure always means a higher chance of these effects. As a result, much is dependent on statistics due to the different interactions of various ions. Some cells repair and no effect takes place while others do not repair or even misrepair which leads to mutations[6]. Therefore, a difference can be made between the measurable and physical dose an object absorbs and the biological equivalent dose on tissue. The former is expressed in the SI unit Gray [Gy] and the latter in Sievert [Sv]. Although, often the radiation absorbed dose [rad] is also used which has the relation of 1 Gy = 100 rad [17]. In the following parts, these three units will be worked with. The radiation weighting (W_R) and tissue weighting (W_t) factors are typically used to perform the converion. These have been set by the International Commission on Radiological Protection (ICRP) and can be found in Figure 4.2 and Figure 4.3. These convert the physical dose to a biological one. The use and relation of these factors and units can be seen in Figure 4.4.

[4] https://www.swpc.noaa.gov/news/solar-cycle-25-forecast-update [Cited on 18/06/2021]
[5] https://www.nasa.gov/sites/default/files/atoms/files/space_radiation_ebook.pdf [Cited on 18/06/2021]
[6] https://www.nasa.gov/feature/space-radiation-is-risky-business-for-the-human-body [cited on 17/06/2021]

Radiation type	Radiation weighting factor, w_R
Photons	1
Electrons[a] and muons	1
Protons and charged pions	2
Alpha particles, fission fragments, heavy ions	20
Neutrons	A continuous function of neutron energy (see Fig. 1 and Eq. 4.3)

Figure 4.2: Radiation weighting factor (W_R) determined by ICRP in 2007 [16]

Tissue	w_T	$\sum w_T$
Bone-marrow (red), Colon, Lung, Stomach, Breast, Remainder tissues*	0.12	0.72
Gonads	0.08	0.08
Bladder, Oesophagus, Liver, Thyroid	0.04	0.16
Bone surface, Brain, Salivary glands, Skin	0.01	0.04
	Total	1.00

* Remainder tissues: Adrenals, Extrathoracic (ET) region, Gall bladder, Heart, Kidneys, Lymphatic nodes, Muscle, Oral mucosa, Pancreas, Prostate (♂), Small intestine, Spleen, Thymus, Uterus/cervix (♀).

Figure 4.3: Tissue weighting factor (W_t) determined by ICRP in 2007 [16]

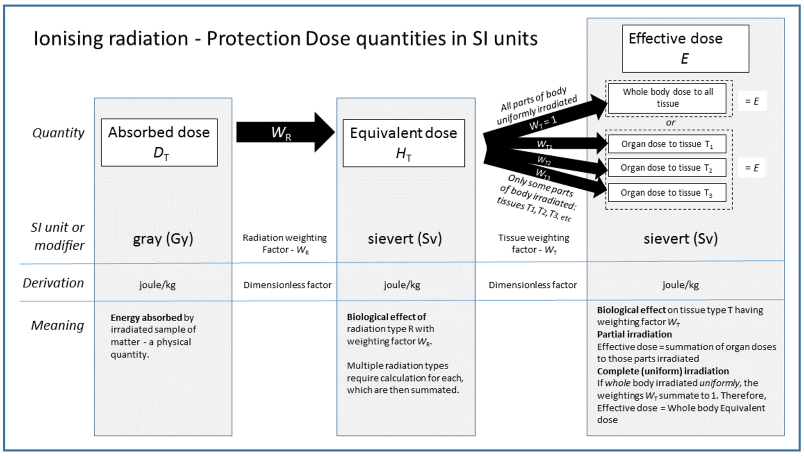


Figure 4.4: Relationship between different radidation doses and their SI unit according to ICRP [73]

Dose Limits

As discussed above, there are two important types of radiation to be shielded against. Both Galactic Cosmic Rays and Solar Particle Events are stopped on Earth by the atmosphere which acts as a mass barrier and the magnetic field which alters the path of the ionising radiation. People on Earth still receive some radiation, on average about 3 mSv/year but, essentially, it is without danger to their health. Measurements from the Moon, performed by the Chinese Chang'E 4, show that on average a GCR equivalent dose of 1369 μSv per day is received on the Lunar surface. This would come down to about 500 mSv received over a year. Their worst case scenario is somewhere around 600 mSv per year. On the ISS, the GCR contribution accounts for only 190 mSv/year[113]. Although ESA career limits for astronauts are a lot higher than this amount of radiation dose, as can be seen in Figure 4.5, ESA also sets warnings when their astronauts receive more than 400 mSv [19]. There is uncertainty about how much effective dose will be received by an SPE on the lunar surface. Unfortunately, the Chang'E 4 did not encounter such an event during its measurements. However, some sources say it can go from 1 Sv [87] to a worst case of even 150 Sv [81] when an extreme SPE hits. Although immediately detrimental to the mission and the dose career limits, it can easily be shielded against if enough material is present.

Protection in the Habitat

The initial idea for the protection of these two types of radiation rests on the difference of the main habitat and the safe house. The main habitat will be of prime importance in shielding against the constant GCR's which, when accumulated over a year of exposure, can decrease the average lifetime expectancy of the astronauts by ten or more years. It is assumed that at least one, if not more, SPE's will take place during the first year of this mission but undoubtedly there will be many if the ten years of service of the habitat is taken into account. For now, the safe house will be the place where the astronauts can hide and be shielded against such events. To do this properly, the first problem arises. These SPE's need to be predicted. However, in practice this can be quite difficult. Only since a few years are scientists able to predict solar events. A new study from 2020 shows they could predict up to seven out of nine mayor solar flares from the last solar cycle [66]. The use of forecasting will play a big role if this method can keep developing in the coming years. Another method which will run in parallel is the use of nowcasting. Using data on Earth and the sensors inside the habitat to measure

TABLE III. Age- and gender-dependent career effective dose limits (in Sv) as recommended by different space agencies (Cucinotta, Hu et al., 2010; Straube et al., 2010). NASA limits are always based on 3% risk of exposure-induced death, and the values refer to a 1 yr mission. Different values will be calculated for different mission durations (Cucinotta, Hu et al., 2010).

		Age at first exposure, (yr)			
Space agency	Gender	30	35	45	55
NASA (USA)	Female	0.47	0.55	0.75	1.1
	Male	0.62	0.72	0.95	1.5
JAXA (Japan)	Female	0.6	0.8	0.9	1.1
	Male	0.6	0.9	1.0	1.2
ESA		1.0	1.0	1.0	1.0
FSA (Russia)		1.0	1.0	1.0	1.0
CSA (Canada)		1.0	1.0	1.0	1.0

Figure 4.5: The annual career effective dose limits set by the different space agencies. This table was taken directly from [26]

radiation and give a warning when a peak occurs. When this happens, the astronauts will have to grab their suit and flee to the safe house as quickly as possible. The importance of having the safe house close by is therefore greatly beneficial and will be discussed in Section 6.2.

As mentioned before, GCR's are difficult to shield against due to their high energy, penetrating up to even several hundred centimetres of aluminium [88]. More than once, scientists have vouched for active magnetic shielding on a lunar habitat or spacecraft. In this way, trying to reenact the magnetic field of the Earth which leads the dangerous particles around the habitat [67]. However, with the current technology, this method is way too energy-intensive. Definitely when other solutions exist. Adding multiple metres of a material coming from Earth is unfeasible due to the high upmass. This makes the decision of In-Situ-Resource-Utilisation (ISRU) an easy one. Making use of the regolith available on the Moon to cover the habitat will provide a first form of protection against GCR's. Careful consideration needs to be taken into account for the scattering of secondary radiation. However, research shows that using material with low atomic mass performs really well against protecting this type of radiation and the lower the atomic number, the less secondary radiation occurs. Therefore, hydrogen-rich materials are very suitable options [17]. As it is preferred to have as little secondary radiation as possible inside the habitat, the hydrogen-rich material will be one of the last layers through which the ionising particles will travel. Research showed that Polyethylene ($(C_2H_4)_n$) is a good candidate because it shows great protection against Fe ions [79]. Other layers include water bags above the sleeping areas, air and the structural layers. To ensure the safety of the astronauts, the ALARA (As Low As Reasonably Achievable) [22] principle will always be regarded in the further design of the habitat.

SPENVIS

To verify the protection of the proposed design and the total ionising dose the astronauts will receive, the Multi-Layered Shielding Simulation (MULASSIS) tool was used. This is a Geant4 based tool which runs on Monte Carlo simulations . This Geant4 tool is very useful for radiation research as it simulates the passage of particles through material [7]. Using MULASSIS in the SPENVIS environment which is ESA's SPace ENVironment Information System [8] a simulation can be set up where the different protection layers are entered and from which the total ionising dose can be found. Below, the settings for the current project have been established.

- Due to the Moon not being in the package, the orbit type was set to Near Earth interplanetary. Except for the Moon having half of the incoming radiation due to the mass protection of the Moon itself on the surface.
- Departure was set in 2030 with a mission length of 1 year. This is close to solar minimum so GCR's will have more influence.
- For the solar particle fluences, the SAPPHIRE (1 in n year event fluence) was chosen with n=1000 to make sure the worst event is protected against.
- For the galactic cosmic ray fluxes, everything was set to default except solar activity data which was set to mission epoch.
- During testing, 10000 particles were used to shorten simulation time. When the final results were close, this changed to 100000 particles.

[6] https://essr.esa.int/project/mulassis [Cited on 19/06/2021]
[7] https://geant4.web.cern.ch/node/1 [Cited on 19/06/2021]
[8] https://www.spenvis.oma.be/[Cited on 19/06/2021]

- Initially, a sphere was worked with in the geometry part trying to reenact the cylindrical shape of the habitat. However, there seemed to be a bug so this was changed to a planar slab again.

In the beginning, different parameter tests were done to understand SPENVIS and MULASSIS better. During this period, the use and navigation through the programme became a lot smoother. However, some bugs came along which slowed down the progress. For some reason, when adding extra layers of vacuum, the radiation dose either increased (with thin layers) or decreased (with thick layers >1 m). Another oddity was the fact that when adding a layer of 2 mm gave different results compared to two layers of 1 mm. Also the protection of certain materials (like hydrogen) which were of high performance in literature, showed poor results and also increased the radiation dose. The reason is still unknown but, fortunately, these effects disappeared when changing back to a planar slab.

Assumptions

SPENVIS and MULASSIS offer a wide variety of options and settings. It can simulate cases that are far beyond the scope of this project. Therefore, a few assumptions had to be made. Sometimes these assumptions may seem far-fetched and could lead to false results but keep in mind that the use of this programme was intended to verify the design proposal and play around with the layers to see the effects and importance of correctly placing those layers.

- The first assumption to be made was that the GCR's were only simulated for the protons they contain. It is possible to simulate from hydrogen to iron or nickel but this would all need to be accumulated. As GCR's consist out of 85% hydrogen protons, this was assumed good enough.
- As done in literature before this report, a human phantom, represented by 30 cm of water was added to the layers. This way, it is easier to check what the actual received dose is when a human would have stood there [5].
- Another assumption and probably the most uncertain one is the representation of regolith by SiO_2. Although the surface exists mostly out of this silicum oxide (about 45%) [105] it is presumptuous to assume this. However, no better simulant for lunar regolith was found which could be easily implemented in MULASSIS. Adding all the different oxides as various layers also does not represent it well as this forces the ionising particles to go through a layer of minerals it would otherwise just miss due to its low abundance. Both methods have its downsides, but using only one layer will give the most trusthworthy results in this case as the order of the layers could already change the result drastically.

With these assumptions in mind, the results of three simulations with different parameters can be investigated better. For all three, first a layer of vacuum of 1m is added to have a datum to compare against. Then a layer of 1 m SiO_2 as regolith is used, a layer of Polyethylene or PE ($(C_2H_4)_n$), 1 m of air and 30 cm of water which represents a human phantom as discussed above.

The change in parameters is fairly compact and only summarises previous work. First the standard layout will be used: Vacuum, regolith, PE, air and water. The results can be seen in Table 4.1. The second one switches the layer of regolith and PE to show the effect of changing the order of the layers of which the results are found in Table 4.2. The third layout can be found in Table 4.3 involves making the PE layer from 2 cm to 10 cm to check its protection potential.

Table 4.1: Layer change testing on SPENVIS, version 1

Layer	Thickness	Density	Dose	Error
Vacuum	1 m	1e-22 kg/m3	6.6038E+00	2.4510E-01
Regolith	1 m	2.65 g/cm3	3.3818E+00	7.4115E-02
PE	2 cm	940 mg/cm3	1.6293E+00	7.1907E-02
Air	1 m	1.20479 mg/c	1.2963E+00	6.9544E-02
Water	30 cm	1 g/cm3	1.1769E+00	4.6489E-02

Table 4.2: Layer change testing on SPENVIS, version 2

Layer	Thickness	Density	Dose	Error
Vacuum	1 m	1e-22 kg/m3	6.4622E+00	2.1073E-01
PE	2 cm	940 mg/cm3	5.3055E+00	1.2519E-01
Regolith	1 m	2.65 g/cm3	3.4614E+00	7.7041E-02
Air	1 m	1.20479 mg/c	1.4822E+00	9.0788E-02
Water	30 cm	1 g/cm3	1.3779E+00	6.4790E-02

Table 4.3: Layer change testing on SPENVIS, version 3

Layer	Thickness	Density	Dose [rad]	Error
Vacuum	1 m	1e-22 kg/m3	6.2541e+00	2.5304e-01
Regolith	1 m	2.65 g/cm3	3.4635e+00	7.7475e-02
PE	2 cm	940 mg/cm3	1.1961e+00	5.3447e-02
Air	1 m	1.20479 mg/c	7.3110e-01	6.3134e-02
Water	30 cm	1 g/cm3	7.0344e-01	4.2757e-02

First, using the normal layout leads to a good protection against GCR's. Having a total absorbed dose of 1.1769 rad of protons over the course of a year is about equal to an equivalent dose of 23 mSv following the conversion in Figure 4.2. This seemed unreasonably low but a study was found which also used 1 m of regolith and a human phantom made out of water. They came down to a value of around 50 mSv and they used the old values of ICRP 60 (1991) where the W_R of protons ranged between two and five instead of ICRP 103 (2007) which sets the value at two [5]. The method used in this report is preliminary but already gives a good estimation of the order of magnitude. Second, PE aids in the reduction in layout 2 but the portion is lower than when this layer is put after the regolith layer. When considering the third layout, the increased thickness shows that PE does not reduce the dose more. This could be explained by the fact that PE is good against shielding of HZE particles but not necessarily protons. This makes the equation of finding an ideal layout a bit more tedious. As stated before, only hydrogen particles will be discussed in this report.

As the protection was better than expected, another simulation was run to check what would happen in case of an SPE. As already discussed, the SAPPHIRE (1 in n years event) simulation with the setting of a 1000 years was used for the standard configuration (layout 1). The result can be seen below in Figure 4.6:

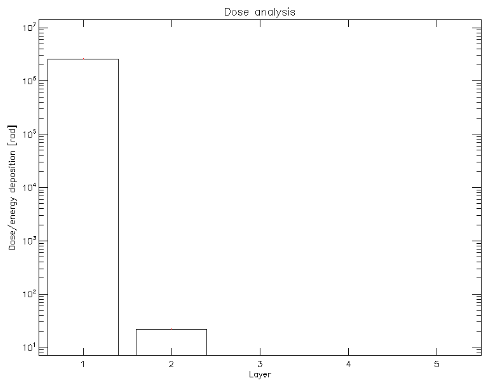


Figure 4.6: Simulation for an extreme SPE with SAPPHIRE (1 in 1000 year event) with the layers in configuration 1

Surprisingly, this also bodes very well for the protection of the habitat. It was known that SPE's could easily be stopped by using a lot of mass. It was unexpected to see the dose going down to zero. However, research shows that a layer of 50-100 centimetres already reduces the dose significantly [69]. Although other papers suggest using a layer of at least 5.4 metres [47]. It is recommended more thorough research needs to be done on this design to verify and validate this result. One thing is sure and that is that 1 m of regolith in combination with a hydrogen-rich material like PE protects very well against radiation encountered on the Moon.

4.1.1. Conclusion

Radiation on the Moon is a big hazard for the health of the astronauts. There are two important radiation types for missions to the Moon called Solar Particle Events (SPE) and Galactic Cosmic Rays (GCR). The former is incidental but with immediate peril due to high fluxes of particles while the latter is fairly constant over time with lower flux but can lead to health risks due to the accumulated dose. SPE's are more easily stopped due to their low energy by adding mass inbetween the astronauts and the incoming radiation. GCR's are more tedious to stop as they do have high energy and can easily penetrate multiple centimetres of material. This can be mitigated by choosing material with low atomic mass or hydrogen rich compounds like Polyethylene (PE). The method of using 1 m of regolith and 2 cm of PE show promising results for the protection of the astronaut. Preliminary results give values of accumulated equivalent dose of 23 mSv over a year. This is far below the dose of 400 mSv after which ESA sets a warning. It is possible to use a layer that is less than 1 m of regolith. However, following the ALARA principle, the extra protection for the health of the astronauts is by far more advantageous than the increased level of difficulty of adding more regolith on top. Further and more in-depth research needs to be done for the exact amount of radiation dose received during the mission.

4.2. Micro-meteoroid Hazards

With the lack of an atmosphere, meteoroids cannot burn up before hitting the surface of the Moon. If one looks at the Moon, it is obvious that the landscape is formed mostly by meteoroid impact. Especially with the velocities of these space rocks which can range from 20 km/s up to an eyeballing 72 km/s.[9] This leads to even a small particle of a few millimetres already causing severe damage. Protection against this is therefore of importance to ensure the safety of the habitat and thus the astronauts.

In Figure 4.7, the flux distribution of meteoroids hitting the lunar surface per square metre per year is plotted against the size (in cm). A study on lunar habitats shows that a layer of 45.9 cm regolith can protect against meteoroids up to 7 cm in diameter.[47]

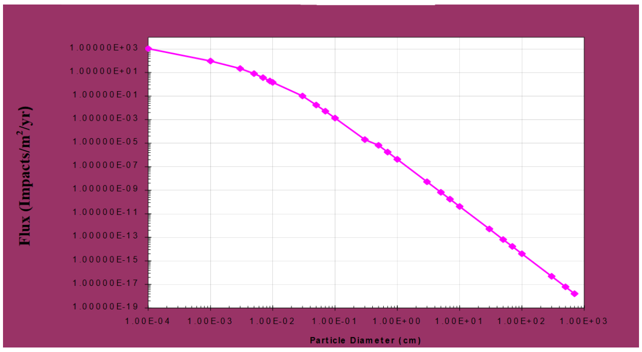


Figure 4.7: Meteoroid flux on the Moon in size (cm) and per square metre per year. Taken from [69]

Looking at Figure 4.7, the chance of such rock is about 1.000E-10/m^2/yr. If a conservative estimate is made for the area solar farm, the habitat and the safe house it comes down to about 10000 m^2 and with ten years of operation, this would lead to a flux of 0.00001 for that surface area over ten years. This is definitely a rough estimate but taking into account that the layer of regolith will be more than 45.9 cm and that the flux of bigger meteroids hitting is even smaller it is assumed that the habitat, when using 1 m of regolith, should ensure the safety of the astronauts against micro-meteoroid impacts.

4.3. Hypogravity

Another factor influencing the astronauts health and more importantly their physical condition is the lack of gravity. The gravity on the Moon is 6 times lower than on Earth; 1.62 m/s^2. Even if gravity isn't null, it will not be big enough to prevent muscle atrophy. However, by performing similar exercises than the ones performed on the ISS such phenomenon can be minimised. It is for this reason that an exercise area will be included in the main habitat (including Moon-adapted ISS workout machines). In the safe house, the astronauts will also have to workout, since between 5 to 11 days without working out, the astronauts experience up to a 20% loss of muscle mass in space [10]. For this reason, small apparatus such as resistance elastic-bands will be used to workout for 21 days in the safe house.

On top of muscle reduction, an important factor to take into account is the fact that human body is not built to function in space, we are used to live in a gravity environment and changing such routine requires important training pre-launch. Hypogravity can have a real impact on the physiological phenomena which are impacted by the body fluids redistribution [89]. The astronauts will thus be highly-trained and closely supervised by flight surgeons to survive and work efficiently in such low-gravity environments.

4.4. Extreme Environment

In addition to the low-gravity environment experienced on the Moon by the astronauts, a large number of extreme events, altering astronauts' actions are present on the Moon. These are; hypobaric environment, extreme temperature and Moon dust.

Hypobaric Environment

Decompressing sickness caused by the hypobaric lunar environment, or more precisely caused when switching from a higher to a lower pressurised environment (or inversely) can be seen as a simply uncomfortable

[9] https://www.nasa.gov/centers/marshall/news/lunar/overview.html [Cited on 21/06/2021]

[10] NASA information on muscle atrophy, URL: https://www.nasa.gov/pdf/64249main_ffs_factsheets_hbp_atrophy.pdf [Cited June 9, 2021]

state for the astronauts but it can really lead to a number of health complications such as arterial gas embolism [83]. In order to minimise these effects and for the astronauts to perform EVAs, pressurised suits will be used by the astronauts for the pressure difference between the two environments to be at a minimum. On top of that, training on Earth pre-launch will be done by the astronauts to get somewhat used to such extreme change in environments.

Except when performing EVAs, astronauts won't be subjected to hypobaric environment since the habitat atmospheric pressure will be at atmospheric sea level pressure (namely 1 atm) at all time. In order to maintain such internal pressure, an airlock will be used as a transition area to prevent the inflatable from deflating and the astronauts to live in an unhealthy environment. Sensors will also be used to track pressure fluctuations inside the habitat and adjust the pressure accordingly, when needed.

Extreme Temperature

The habitat, located on the south pole of the Moon as elaborated on in Section 6.2, won't experience as extreme temperature ranges as it can in other parts (as low as -173°C and as high as 127°C)[11]. Between the Shackleton crater and the de Gerlache crater, where our habitat will be located, the temperature range varies from (-60°C and +40°C) [39]. The location choice already mitigates these extreme environments, however a temperature of -60°C can still be quite critical for the human body, especially when exposed to it for long periods. The entire habitat will contain thermal controls and sensors in order for the atmosphere to stay at a constant comfortable temperature for the astronauts, averaging 22°C. When performing EVAs the astronauts will wear special suits including thermal protection by making use of technology like micro-encapsulated phase-change materials [76], that adjust the temperature in accordance with the outside temperature.

Moon Dust

The Moon surface is covered in regolith dust, ultra-fine particles. Such micro-particle can highly harm the astronauts if inhaled due to their sharp nature and their large surface area that is chemically reactive [48]. Lunar dust shouldn't enter the habitat with the risk of interfering with the electronics, other subsystems and causing respiratory health risks to the astronauts. In order to prevent such events, the astronauts will be equipped with protective suits when performing EVAs and airlocks will be used at the main habitat and safe house entrance to provide the astronauts with a shower to remove fine-dust and enable to enter the habitat dust-free.

These extreme environmental conditions also have major effects on the habitat subsystems, this is the reason why they will all be adapted to lunar conditions.

4.5. Social Interaction

Astronaut's mental health is extremely important and is carefully taken into account throughout this mission, starting with in the mission design. The impact isolation has on astronauts' health are actively being studied in remote areas such as Antarctica in order to improve the astronauts' mental health when confronted to a year long mission confined in space with 3 other astronauts.

The effects of such confinement in space are diverse; neurocognitive and immune modulatory changes, fatigue, sleeping disorder [85] which can highly harm their mental health and physical health. In order to minimise these negative effects, and for the mission to be successful the astronauts are highly trained pre-launch and selected according to specific criteria.

In addition to astronauts' training and screening a number of arrangement and considerations have been made when designing the habitat. A total amount of 247 m^3 of habitable space (excluding airlock and corridors) will be available for the astronauts, they each will have separate rooms of 6.6 m^2 or 15.84 m^3. The rooms will be arranged in such a way to make the space look bigger; arranging the table, wardrobe and bed for maximum walking space.

Since no windows are allowed in the habitat, a screen will display a number of nature movies, pictures and posters will be placed inside the habitat and inside the personal units, this will enable the astronauts to feel a bit more like in a Earth-like environment.

Another factor that might affect the astronaut's psychological health, taste buds and olfactory system is the food they eat [12]. Astronauts' food will be an add-water type of food, will taste similarly to Earth food and will be highly analysed for the astronaut's not to run out of nutrients, which could alter their health.

[11] Moon Temperature, URL: https://www.space.com/18175-moon-temperature.html [Cited June 10, 2021]
[12] Space food technology, URL: https://www.foodunfolded.com/article/space-food-technology-why-we-havent-been-to-mars-yet [Cited June 13, 2021]

5. Risk Assessment

In order to ensure that the mission will be successful it is important to assess potential risks that might occur throughout the mission. This chapter outlines the risks that were identified during the mission and assess the likelihood of these risks occurring as well as the severity this would have for the mission.

5.1. Overview of Determined Risks

The risks have constantly been addressed throughout the project in both the baseline report [38] as well as the midterm report [39]. Four categories of risks were determined, risks occurring during production and testing (PD), during travel (T), during habitat assembly (HA) and those during the operational phase (OP). An overview of these previously analysed risks is provided in Table 5.1. A more detailed explanation for each of these risks can be found in previous reports.

Table 5.1: Overview of previously introduced risks [39] [38]

Label	Risk	Label	Risk
[PD01]	Delays during production	[OP01]	Solar Particle Events
[PD02]	Production issues	[OP02]	Asteroids
[PD03]	Manufacturing errors	[OP03]	Moon quakes
[PD04]	Failing the testing stage	[OP04]	Medical emergency
[PD05]	Damage during testing	[OP05]	Life support equipment failure
[PD06]	Material deficiencies	[OP06]	Loss of communication
[PD07]	Production exceeds budget	[OP07]	Maintenance need
[T01]	Launching/landing failure	[OP08]	Supply shortage
[T02]	Collision with space debris	[OP09]	Loss of Oxygen
[T03]	Spacecraft failure	[OP10]	Temperature malfunction
[T04]	Damage during habitat transportation	[OP11]	Pressure Loss
[T05]	Landing inaccuracy	[OP12]	Galactic Cosmic Rays
[T06]	Lunar dust formation	[OP13]	Perforation of the inflatable structure
[T07]	Launch delays	[OP14]	Fire Hazard
[T08]	ADCS failure	[OP15]	Microbial Overgrowth
[HA01]	Habitat is damaged	[OP16]	Malfunctioning of sensors
[HA02]	Unexpected obstacles at the location	[OP17]	Presence of harmful gases
[HA03]	Problems with the automatic deployment		

For each of these risks the likelihood of it occurring as well as the severity of this risk were determined. Then a mitigation strategy was drawn up in order to try to decrease either this likelihood, severity or both and thus potentially lower the impact of this risk.

5.2. Newly Identified Risks

As mentioned before the identification of these risks is a continuous process. As more details of the mission become clear, new risks can be identified. Additionally, some risks were omitted at an earlier stage and have been added now. An overview of these new risks, as well as their corresponding likelihood, severity and mitigation strategy is provided in Table 5.2.

- **[OP18] Power Loss**: a power outage could occur during the mission and possibly endanger the astronauts as the life support systems need constant power.
- **[OP19] Lunar Dust Interference**: the risk of lunar dust formation during landing was already identified in risk [T06], however the risk of lunar dust interfering is present throughout the entire mission. The largest dust formation after landing will most likely occur during the assembly of the habitat when the lunar soil will need to be excavated as described in Chapter 8. Lunar dust can interfere with electronics as well as pose a health hazard to the astronauts as it sticks to airways when it is inhaled [104].
- **[OP20] Astronaut mental health deteriorates**: while the astronauts are well prepared for both the physical and mental challenges that are present during spaceflight issues might still occur. The relatively confined space the astronauts should share together might cause tension to occur as have happened before during missions. Additionally, possible anxiety and depression reports have been made in the

past. However, only about 2% of reported medical issues by astronauts have been psychologically related [1], it is important that steps are taken to improve astronaut well being.

- **[HA04] Manufacturing machines malfunction:** One or more machines used for manufacturing and habitat assembly fail to function properly which lead to the habitat not being assembled properly or not at all depending on the malfunction severity.
- **[HA05] More time or energy needed to set up the solar farm:** Setting up the solar farm is an essential step in the manufacturing process since it enables to generate power. Such step, being autonomous, can be very complex and might require more time or energy than initially planned for.
- **[T09] Refueling in LEO fails to work:** Starship, being a reusable spacecraft, should be refueled in LEO by another refuel-specific Starship spacecraft. If refuelling fails to happen the spacecraft won't have enough propellant to land on the Moon
- **[PD08] SpaceX doesn't meet its 2024 goal:** SpaceX is really optimistic about its 2024 goal to bring back people on the Moon using Starship. But it is possible that Starship will face some delays and fails to meet this deadline.
- **[PD09] Starship propellant system not improved to be refuelled using lunar resources:** The current Starship model is using CH4 and 02 as propellant, which can't be created using lunar resources. In order to bring back the spacecrafts to Earth at the end of the mission, propellant is needed in large quantity and should be manufactured on the Moon. If such process isn't possible, the Starship used will have to be abandoned on the Moon which defeats the purpose of using a reusable spacecraft and pollutes the lunar environment.
- **[OP21] Safe house fails at the same time as the main habitat:** The safe house if being used in case of a catastrophic event was to happen to the main habitat, for the astronauts to be safe for a total number of 21 days. If its structure fails the same way as the main habitat, it defeats the purpose of having such shelter and would lead to the astronauts being unsafe.
- **[OP22] Effects of long-term (>6months) exposure of GCR underestimated:** The amout of GCR per astronauts per days might have been underestimated which can have direct consequences on the astronauts health Section 4.1.
- **[OP23] Electrical equipment malfunction due to environmental reasons:** The electrical equipment might disfunction when subjected to SPE or too much lunar regolith.
- **[HA06] Habitat location not suitable for the mission:** The habitat location isn't suitable for the future experiment that will be held on the Moon or the lunar soil at this location is too dense and digging becomes harder than necessary.
- **[OP24] Leak in the Oxygen or Hydrogen tank:** The tanks containing the Oxygen or Hydrogen might not be 100 % hermetically sealed and could lead to potential gas loses
- **[OP25] Failure in the fuel cells:** Some fuel cells might fail in the lunar environment, during transportation or due to a fabrication flaw.
- **[OP26] Electrolysis device failure:** The electrolysis devise is essential in our mission but a system failure can always happen.

5.3. Risk Mitigation

In order to mitigate a risk, its risk rate should first be known in order to know how critical the risk is and how much attention should be payed to reduce its rate. The risk rates can be easily calculated using Equation 5.1, and be classified in 5 different categories: Trivial: 1-3, tolerable: 4-6, moderate: 8-10, substantial: 12,14,15,16, and intolerable: 20, 25.

$$\text{Risk rate} = \text{Likelihood} \times \text{Impact} \tag{5.1}$$

In order to determine the risk rate both the likelihood and impact rating should first be assigned to each risk. These values have been chosen using common sense, literature and group brainstorms. Then a mitigation strategy should be picked to reduce the risk rate when necessary. An explanation for the different scale used (1= Remote, 2= Unlikeky, 3= Possible, 4= Likely, 5= Almost certain), for the impact rating (1= Trivial, 2= Minor, 3= Moderate, 4= Major, 5= Severe) and for the different mitigation strategies (Avoid, Reduce, Accept and Transfer) can be found in the baseline report [38].

[1]https://cmsw.mit.edu/angles/2019/headspace-how-space-travel-affects-astronaut-mental-health/

Table 5.2: Overview of newly introduced risks and their mitigation strategy

Risk ID	Likeli-hood	Impact rating	Risk rate (pre-mitigation)	Planned response	Risk rate (post-mitigation)
OP18	3	5	15	REDUCE - while (partial) power outages have occurred on for example the ISS [2] this risk can mostly be mitigated by having redundant power sources. As will be described in Chapter 9 the habitat will have access to both hydrogen fuel cells as well as solar power thereby adding redundancy into the design.	6
OP19	5	4	15	REDUCE – The formation of lunar dust is inevitable, equipment will be protected by transporting cargo in shielded environments as described in Chapter 8 and to further prevent dust from entering the habitat the airlock will contain dust showers to reduce the amount of dust carried into the habitat	9
OP20	2	2	4	REDUCE - several measures will be taken to improve the astronaut well being. Firstly, a crew should be chosen that gets along well. Additionally, improvements in morale can be made with with the design of the habitat and mission as was detailed in Section 4.5 such as variation in the food provided, plenty of light simulating the day/night cycle, private spaces, etc.	2
OP21	2	3	6	REDUCE – The safe house will be made out of a completely different structure, including extra SPE and CGR protection and using material having different performance. It will also contain a number of redundant subsystem in order to account for possible failure.	3
OP22	2	3	6	REDUCE – In order to reduce such risk, one mitigation strategy would be to make further research about this specific phenomenon and its characteristics at this specific location on the Moon.	3
OP23	3	3	9	REDUCE – In order to reduce such risk, a number of mechanism can cover sensitive devices and protect them further against SPE (water bags or something with high mass) and regarding regolith dust, bagging them in sealed environment would be a great protection option	3
OP24	2	2	4	REDUCE — To reduce this risk a mitigation strategy would be to simply use multi-layer insulation for the tanks together with a large safety factor (4) when designing its structure.	2
OP25	3	3	9	REDUCE — A mitigation strategy would be to account for this risk and bring back-up fuel cells. In case of failure, the fuel cell can simply be changed for a new one	3
OP26	2	4	8	REDUCE — I order to mitigate this risk, a solution would be to back two electrolysis systems for redundancy.	4
HA04	2	4	8	REDUCE – Extensive testing will be done on Earth using these machines in order to make sure they are autonomous and accurate enough to build the habitat on the lunar surface. In addition to this a number of redundant system will be used to make sure that in case of failure, such machine is still able to operate. For most of the machines used, there will be at least a copy (except the transporter) so a solution can always be to only use one of them even though the work will be done slower	4
HA05	3	4	12	REDUCE - A solution would be to anticipate for such risk and bring more Hydrogen and Oxygen	4
T09	2	5	10	TRANSFER – SpaceX is conducting and will continue to conduct a large number of tests for such manoeuvre to become close to risk-free. By 2030, a large amount of test would have been conducted and refuelling in LEO would have become somewhat 'easy' and possibly new technologies might have become available by then.	6
PD08	2	3	6	ACCEPT – New technology for lunar exploration might have become available by then so it isn't crucial for the mission. Going to the Moon is a goal for a number of countries and research on Moon travel is being done every day. If not Starship, a new spacecraft might have become available by then	6
PD09	2	3	6	TRANSFER – After all, the main reason why the Moon wants to be used would be as a 'refuelling' area in order to more easily reach Mars and to bring more mass there. This means that creating a Starship that isn't refillable on the Moon doesn't serve any purpose and thus using lunar resources as propellant should really be investigated further by SpaceX and Starship propellant system should be adapted.	3
HA06	2	4	8	REDUCE — This part of the Moon, more particularly the ridge next to the Shackleton Crater has already been investigated by NASA's Lunar Reconnaissance Orbiter (LRO) and the ground composition is know. In order to reduce further this risk, a ground/atmosphere scan can be performed before the lunar manufacturing starts in order for the ground composition and overall atmosphere to be further detailed.	4

[2] https://www.nasaspaceflight.com/2014/05/iss-power-loss-suspected-electrical-short-component/

5.4. Risk Maps

A risk map is used to visualised how critical a risk is on the mission. On the risk map the likelihood rating is plotted against the severity to easily assess which risks will pose the greatest threat to the overall success of the mission. The risk map before the mitigation strategy can be found in Table 5.3 and the updated values after the mitigation strategy is applied are presented in Table 5.4.

Table 5.3: Risk map displaying the risks before implementing mitigation strategy

		Severity				
		1	2	3	4	5
Likelihood	5			T06	OP12, OP19	
	4		PD01, PD03	T07, HA05	PD06, OP06	OP01
	3		OP07, OP15 PD05	PD07, OP23, OP25	PD02	OP18, OP14, OP16
	2		OP20	T04, T08, HA02, OP03 OP21, OP22 OP24, PD08, PD09	OP02, OP26, PD04, HA04, HA06, OP13	T02, T03, HA01, HA03, OP05, OP09, OP10, OP11, OP17, T09
	1			OP04	T05, OP08	T01

Table 5.4: Risk map displaying likelihood and severity of the determined risks, after mitigation

		Severity				
		1	2	3	4	5
Likelihood	5	T06, OP14, OP16,	OP12			
	4	PD03, T07		OP01		
	3	OP07	OP02, OP06, OP18	OP19		
	2	PD01, PD07, T08, OP04	PD02, HA02, OP03, OP13	T09	PD06	PD08
	1	T04	PD05, OP15 OP24, OP20	T05, OP10 OP21, OP22 OP23, OP25 PD09	PD04, OP08 HA04, HA05 HA06, OP26	T01, T02, T03, HA01, HA03, OP05, OP09, OP11 OP17

As can be seen from the table the most critical risk will be OP01, solar particle events, followed by OP12, galactic cosmic rays. These 2 radiation related risks carry important consequences for the astronauts and their potential health after the mission. While a lot is done to protect them as much as possible, the risks can not be completely erased. The implications such risks have on the astronauts and how they are being mitigated for our mission is discussed in Chapter 4 and throughout the habitat design chapters.

6. Conceptual Design

This chapter concludes the first part of the report by presenting the design concept believed to be the most suitable for the mission. Many of these decisions were finalised in the midterm report [39]. This report concluded that the design was feasible. Below, the habitat location, initial structural and assembly choices, and the first plan for the launcher segment are all presented. This chapter lays the groundwork for the detailed design in Part II.

6.1. Habitat location

Being roughly the size of Asia, or about $38 \cdot 10^6$ km^2, the Moon offers very plenty of possible locations [1]. Given that different areas vary considerably in terms of availability of resources and expected temperature ranges, a set of criteria was chosen to narrow down the search for the ideal spot.

Firstly, the availability of sunlight is critical for the solar power to be used. Days on the Moon are equal to 29.53 days on Earth [47], leading to a long period for solar energy to cover. To avoid using large amounts of bulky batteries, a location with more sunlight and fewer shadows has to be found. Due to the tilt of the Moon only being 1.54 ° compared to the 23.5° on Earth, summers and winters do not differ greatly. The Moon also has areas with (almost) permanent light and permanent darkness due to the peaks of the craters.

Inside the craters, there can be so-called cold traps of permanent darkness where the temperature has not exceeded 40 K in over a billion years. These cold traps can contain water in the form of ice. Moreover, research on the origins of lunar water is of great interest. The presence of water is especially located within 20° latitude of both poles [68], hence the location is further restricted to this area.

The next criteria is that of the temperature range. Extreme temperatures would warrant additional weight in the form of active or passive temperature control systems, which can be avoided. Due to the long nights and days, the Moon heats up and cools down significantly with temperatures ranging from -233 °C to 123 °C. However, the range is smaller in the poles. Due to the almost constant illumination and the Sun being close to the horizon, the average surface temperatures range between -23 °C and -3 °C. [103]. Other sources mention an average of -53 °C with the same range of $\pm$ 10 °C[47]. Hence, a location near the poles is preferred.

The final criteria is that of terrain. The terrain near the poles is full of craters and slopes. However, the Perseverance Rover had an interesting feature: it made use of Terrain Relative Navigation (TRN), which uses both a map of the location as well as on-board equipment and sensors to seek out a safe landing spot[2]. Currently, they are looking at precisions of about 100 m.[28] This makes it possible to land in harsh terrains which were previously deemed too difficult, such as Jezero on Mars. Because it has already proven its capability, it is believed that TRN will enable landing on one of the poles without too much difficulty.

The ridge near the Shackleton crater satisfies all of the criteria above. Named after the infamous Antarctic explorer Ernest Shackleton, the Shackleton crater has been the interest of scientists for over a decade now. The crater coincides with the rotational axis of the Moon and has a diameter of about 19-21 km and a depth of 4.1$\pm$0.05 km [114]. To compare the size of this with something that is more well known, the distance of the Hague to Rotterdam with Delft in between is superimposed on a map of the Shackleton crater as seen in Figure 6.1. At the rims of the crater, near permanent illumination can be found. However, the inside of the crater -being a cold trap- has not seen light in a few billion years. The average temperature is between 50 and 70 K. Research also shows that it is strongly believed the bottom contains water ice [60]. That there is water on the Moon has been proven and proven again. In 2020, NASA's Stratospheric Observatory for Infrared Astronomy (SOFIA) even discovered water on the sunlit side of the Moon.[3] The actual presence of water ice on the moon has been previously confirmed for PSR's at the South pole.[4] A different study showed that near the crater, seven out of eight prioritized lunar science concepts can be investigated (e.g. bombardment history, polar volatiles and regolith processes) [40].

Hence, the connecting ridge between the Shackleton and the de Gerlache crater was found to be the best suitable location, found at 89°28′ South, 136°40′ West, and 1947 m altitude [60]. On average, it receives

[1] URL: https://www.space.com/18135-how-big-is-the-moon.html [cited 17 May 2021]

[2] URL: https://science.nasa.gov/technology/technology-highlights/terrain-relative-navigation-landing-between-the-hazards [cited 17 May 2021]

[3] https://www.nasa.gov/press-release/nasa-s-sofia-discovers-water-on-sunlit-surface-of-moon[Cited on 21/06/2021]

[4] https://www.nasa.gov/feature/ames/ice-confirmed-at-the-moon-s-poles[Cited on 21/06/2021

sunlight 85.7 % of the time with periods of darkness of 4.5 days and a continuous period of illumination of 230 days. When increasing the height of the measurement to 10 m above the surface, this illumination percentage can increase to a bit more than 90% [35]. Due to these long periods of illumination, short periods of shadow, and the Sun being close to the horizontal, the temperature range is moderate with an average of -10 $\pm$50 °C. [60] Pictures of the chosen location can be found in Figure 6.2 and Figure 6.3.

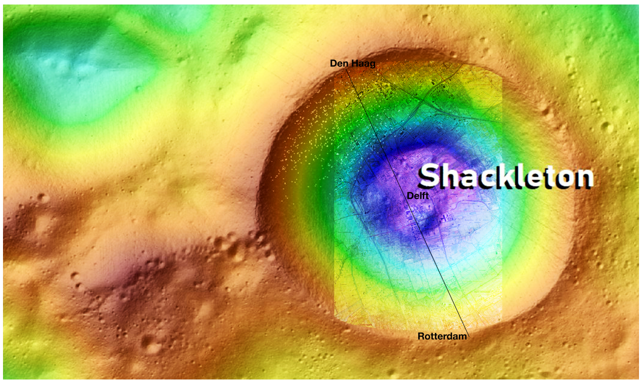


Figure 6.1: Map of the Hague, Rotterdam and Delft superimposed on a topography map of the Shackleton crater

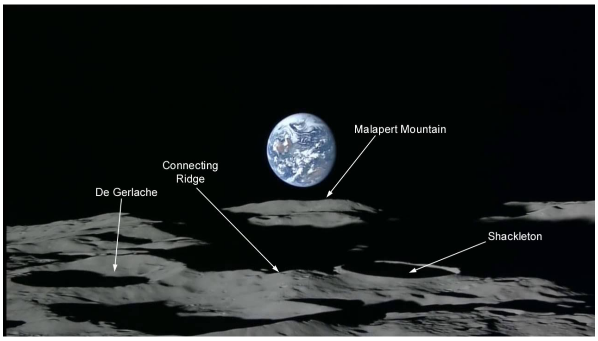


Figure 6.2: Isometric view of the de Gerlache and Shackleton crater with the Connecting Ridge shown as well taken by Japan's KAGUYA spacecraft [5]

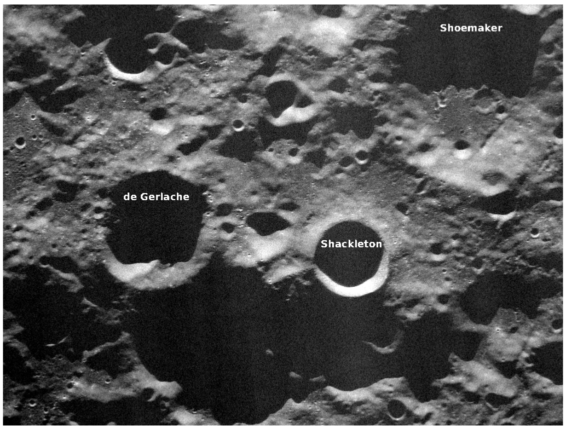


Figure 6.3: Radar view of the de Gerlache and Shackleton crater with the illuminated connecting ridge in between [72]

6.2. Location sites

When the location was chosen for the conceptual design, there was still an area of about 20 km^2 to be skimmed down on. In this section, the precise location will be elaborated on, together with the different sites for the main habitat, the safe house, the excavation area, the landing area and the solar farm. This will be done with the use of various maps on topography, slopes and illumination which can be seen in Figure 6.4[6], Figure 6.5[7] and Figure 6.6[8]. All three figures contain the same region of interest as decided upon in the preliminary design of which the discussion can be found in Section 6.1.

For the five areas, different criteria apply. These are listed below in the same order as the numbering in Figure 6.7:

1. Main habitat: There are no specific requirement for the main habitat except that it needs to be central and close to all the other sites for easier logistics. It shall not be in a PSR to prevent the habitat from going to very low temperatures. It shall be a flat area for simple manufacturing.
2. Safe house: Requirement SH06 says the safe house shall be within 60 minutes reach from the main

[5]URL: http://www.developspace.info/international/14-ridges.html [cited 24/05/2020]

[6]https://www.lpi.usra.edu/lunar/lunar-south-pole-atlas/maps/SPole_SRidgemap_LOLA_v20190515.pdf [Cited on 20/06/2021]

[7]https://www.lpi.usra.edu/lunar/lunar-south-pole-atlas/maps/SPole_SRidgemap_LOLA-Slope-Illum_v20190515.pdf [Cited on 20/06/2021]

[8]https://www.lpi.usra.edu/lunar/lunar-south-pole-atlas/maps/Lunar_South_Pole_Ridge_Slope_V9_300dpi_Flat.pdf [Cited on 20/06/2021]

habitat. However, as discussed in Section 4.1, it is preferable to have it as close as possible without being too close it can get destroyed too when disaster strikes.

3. Solar farm: For the solar farm, the location had to be picked more carefully. It had to be close to the habitat while being in a highly illuminated area without throwing shadows over the other solar towers. Fortunately, in Section 9.3.1 an optimisation was made in which the solar farm uses about 10 solar towers over an area of 500x100 m.
4. Excavation area: For filling the bags and stacking them on the habitat, enough regolith needs to be excavated first. Next to the area being excavated underneath the habitat and the safe house, an excavation area of about 600 m^2 is still necessary. This cannot be too close to the other structures to make sure regolith dust formation is minimised so it does not interfere with the mission.
5. Landing area: As mentioned in Section 6.1, with the use of Terrain Relative Navigation (TRN), it is expected the spacecraft can land with a precision of 100x100 m. The area appointed to it will be larger to make sure no rubble is in the way and TRN can work optimally. Same for the excavation site, the location is chosen to be further from the habitat and other structures to minimise dust formation there.

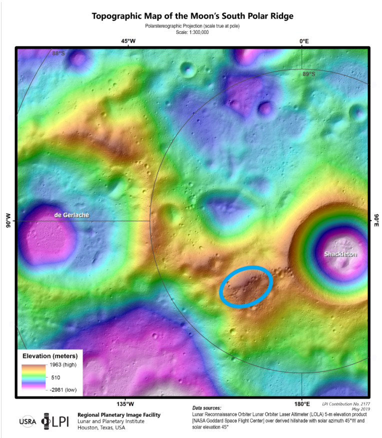


Figure 6.4: Region of interest imposed on topography map near Shackleton Crater

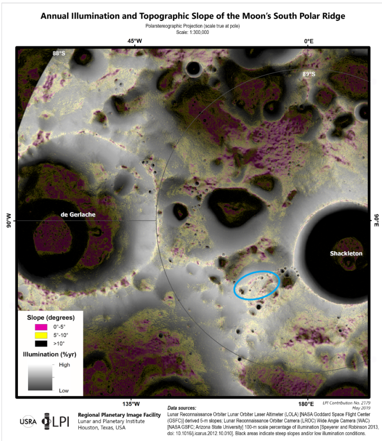


Figure 6.5: Region of interest imposed on illumination map near Shackleton Crater

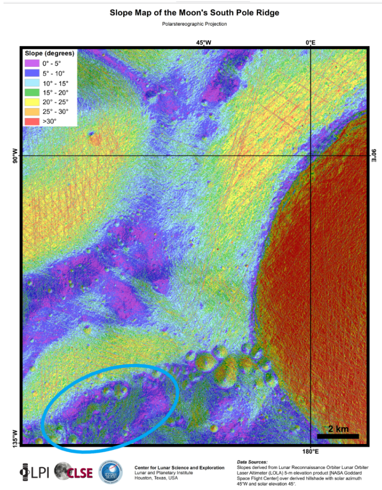


Figure 6.6: Region of interest imposed on slope map near Shackleton Crater

In Figure 6.7, the final configuration can be found of the different location sites. They are numbered in the same way as the above enumeration. Both habitat and safe house are drawn bigger due to the limited resolution. However, the solar farm, excavation site and landing site are roughly true to scale.

6.3. Habitat concept

A habitat capable of hosting four astronauts for an extended period of time should provide sufficient space for the crew to live. This prevents the rise of negative psychological effects linked to living in small space [92] [97]. The most similar, existing example related to a lunar habitat is the International Space Station (ISS). Estimations for the habitable volume based on this mission's duration and the ISS are between 65 and 75 m^3 per crew member [92].[9] Using this data, an upper bound of 300 m^3 habitable volume is used to find the dimensions of the habitat.

A trade-off was conducted between several structural configurations to choose the one that would be most optimal. It was concluded that an inflatable habitat would be the most lightweight option, and one that would be able to fit into a single launch vehicle. The precise dimensions, along with the final layout, can be seen in Figure C.1.

For the inflatable to be fold-able and stored correctly, its walls should still be relatively thin. As detailed in Chapter 4, protecting from the lunar environment requires mass to ensure astronaut safety. Hence, in-situ resources should be used. The habitat would be covered with bags of regolith to stop solar radiation, meteoroids, and cold temperatures from reaching the habitat. Regoliths unique combination of properties make it an excellent natural wall against these hazards. The regolith is bagged to enhance re-usability for future expansion opportunities, to reduce the total amount of regolith needed, and to make the process of covering the inflatable less complex.

[9]URL: https://www.nasa.gov/feature/facts-and-figures [cited 19-05-2021]

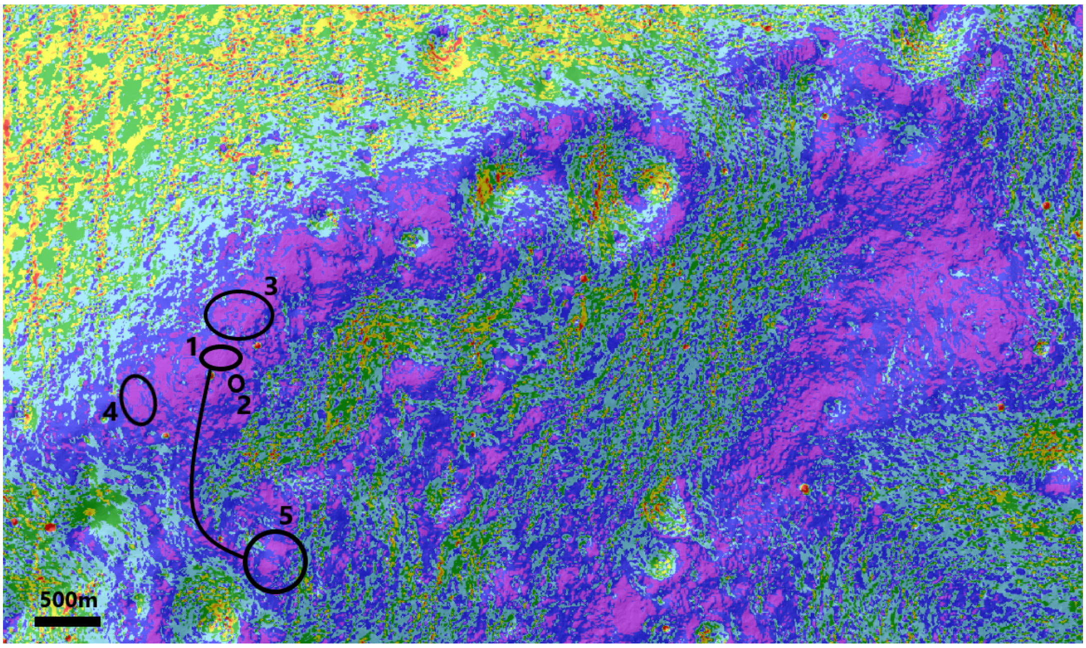


Figure 6.7: Figure 6.6 zoomed in to show location sites and the path from landing to habitat:
1.Habitat 2.Safe house 3.Solar farm 4.Excavation area 5.Landing area
Note: Purple means a slope between 0-5°

The habitat was also chosen to be robot-assembled. The robots would also set-up the power systems, insrt the necessary systems into the inflatable, and excavate, bag, and place the regolith where it would be needed. It would not be feasible to have astronauts take part in these steps.

The structure also should incorporate an airlock, which would preferably be rigid for the sake of structural stability. It also allows the inflatable to be stored in a rigid structure during launch and transportation segments.

The risks detailed in Chapter 5 entail the need for a secondary structure. It should be built to be more resilient than the first and protect against SPEs. Since it would only be used for limited amounts of time, this structure does not have the space requirements of the main habitat, and can therefore be built out of rigid materials.

Before the detailed design phase began, it was already decided that two main types of launches would be utilised: launches to bring the habitat, robots, systems, and all other necessary material for the construction of the settlement, and launches for the astronauts and their vehicle. The precise number of launches was left as a free variable until mass approximations were accurate enough to obtain a reliable number.

After the end of the first 12 months, or in case of severe failure, a return mission was planned to take the astronauts back to Earth.

Part II
Detailed Design

7. Structures

The structure of the habitat is what keeps the astronauts safe for the great majority of the mission, and their design must be detailed enough to showcase the feasibility of the chosen design concepts, while minimising launch mass and risks. For this, the loading analysis and a proper sizing of the inflatable layers was conducted. Additional structural elements were sized, alongside the regolith shielding and the airlock.

7.1. Structures Requirements

Requirements related to structures were derived from the functional flow as well as the risk analyses. Precise values for the requirements are taken either from literature, a short analysis, or from engineering intuition.

The list of requirements regarding the structures and materials is listed below.

- **ST-SYS01-07.1**: The habitat walls shall have radiation shielding of at least 1 kg/m^2
- **ST-SYS01-08.1**: The habitat shall have a total minimum thickness of 40 cm
- **ST-SYS01-10**: The habitat shall be regolith resistant
- **ST-SYS02-04**: The habitat shall be structurally stable for the duration of the mission
- **ST-SYS02-04.2**: The habitat structure shall not fail due to fatigue loading induced by 10 years worth of day-night cycles
- **ST-SYS02-04.3**: The habitat shall be able to resist meteoroid impacts
- **ST-SYS02-04.4**: The habitat shall be able to withstand a temperature range of -70°C to 40°C
- **ST-SYS02-04.5**: The habitat shall be able to withstand the largest moonquake intensity so far observed
- **ST-SYS04-01.1**: The entrance and exit shall have a minimum size of 2m^2
- **ST-SYS04-03**: The habitat shall be constructed in such a way that expansion will not cause catastrophic failure of a system
- **MLO-SYS08-01**: The habitat shall have a dedicated method for astronauts to leave and enter regularly
  - **ST-SYS08-01.1**: The habitat shall contain entrance and exit points
  - **ST-SYS08-01.2**: The entrance and exit points shall stop regolith from entering the habitat
  - **ST-SYS08-01.3**: The entrance and exit points shall contain an airlock

7.2. Loading Analysis

The loading on the habitat must be analysed in order to perform further structural calculations and check the feasibility of the design. The sizing of the structural elements must be done for the extreme load cases, with taking a safety factor into account.

Firstly, one important change that affects the loading of the habitat significantly is the addition of an anchoring system. This is required for stabilisation of the habitat, and can prevent the habitat from turning due to torsional loads. It is described further in Section 8.2.1. It should also be noted that the excavation of the trench under the habitat will have a flat surface. These changes will shape the habitat so that the habitat is pushed down and have a flat surface on its bottom side. The length of the flat surface on the bottom is designed to be 4.3 m, due to storage space designed for the bottom of the habitat.

Internal pressure

The main loading case the habitat will experience on the Moon is the internal pressure load from the inflation. This pressure will affect all surfaces of the habitat and inflate the habitat so that it has equal pressure on all sides. This will be the case until the habitat is constrained by the anchoring system and the trench, and have a flat surface on the bottom. This will create a net force away from the lunar surface, since there is more surface area of the inflatable above the center line and the net pressure heading away from the ground will be greater than the net pressure heading towards the ground.

Weight of regolith shielding

The regolith shielding is in the form of bags that are placed directly on top of the habitat. This will press down on the habitat from above, and has been modelled to be a uniformly distributed load since the thickness is 1 m around the circumference. This is an assumption, since on the side, there will be more regolith bags that press down on the habitat, however, that weight will be supported by other bags beneath them.

The total mass of the regolith shielding is 1350 t. However, since only the mass of the regolith bags above the habitat are considered, it should be calculated separately. Assuming a uniform distribution around the top perimeter of the habitat, the weight of the regolith shielding over the perimeter that spans 160° of the habitat is calculated.

Calculating this over 1 m length of the habitat, the mass of the regolith bags is 12 600 kg, uniformly distributed over 8.38 m of the top surface of the habitat. Since the acceleration due to Moon's gravity is weight 1.625 m s^{-2}, the total weight of the regolith shielding that acts on the habitat is 20 400 N.

Weight of the habitat and internal systems

The weight of the inflatable is calculated using the thickness and density of the inflatable skin and the acceleration due to lunar gravity. This is a variable that changes every iteration since the thickness of the inflatable depends on the stress calculations. The weight of the habitat was modelled as a point load that acts through the centre of the inflatable, heading towards the ground.

The weight of the internal systems were summed up and modelled as a uniform distributed load acting over the flat surface of the habitat on the bottom. This is since the weight of the internal systems act on the floor of the habitat and they are supported by load-bearing boxes underneath that deliver the load to the flat bottom side. The weight of internal systems is not equally distributed over the length of the habitat, and hence two distinct scenarios were considered. One is that the full weight of the internal systems is acting over 1 m of the habitat, while the other scenario is that there is no weight of internal habitat acting over that flooring section of the habitat. The weight of the astronauts were also included in this calculations.

Loading per metre length of the habitat

Figure 7.1 shows the habitat loading case as described above. p_i is the internal pressure, with arrows indicated the direction of pressure on the local surface. F_p is the resultant loading due to pressure that pushes the habitat away from the ground. W_r is the weight of the regolith shielding, modelled as a uniformly distributed load. F_t is the force acting on the inflatable due to the anchoring system, and is modelled as a point load acting on the top of the habitat. W_h is the weight of the habitat, modelled as a point load acting through the centre of the circular cross-section. W_i is the combined weight of the internal systems, and is modelled as a distributed load acting on the bottom side of the habitat. R is the reaction force from the lunar surface.

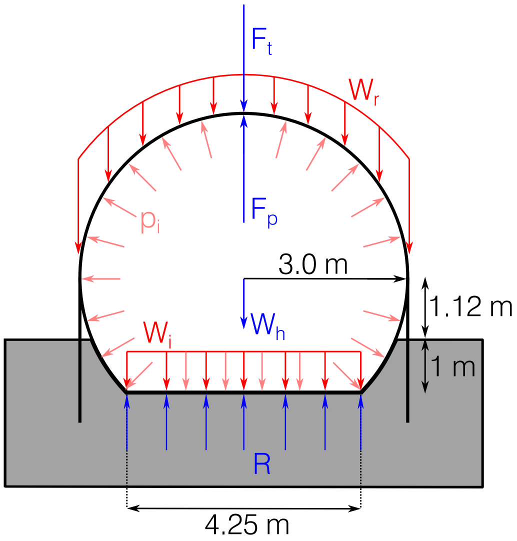


Figure 7.1: Loading diagram of the habitat

Torsion loads can be present on the habitat if the bags of regolith are stacked on one side before another, and apply their weight more on one side of the habitat. However, they were not modelled, as this can be mitigated by stacking the regolith bags equally on all sides.

Table 7.1 shows the values of each load mentioned in Figure 7.1. Note that these values are per metre length of the habitat. The mass values were multiplied by the acceleration due to lunar gravity to calculate the weight loads.

Table 7.1: Loads acting on the habitat

Load	Symbol	Value	Unit
Internal pressure	p_i	101325	Pa
Force from pressure due to irregular shape	F_p	47900	N
Regolith weight	W_r	20400	N
Habitat weight	W_h	25750	N
Internal systems weight	W_i	13650	N
Force due to anchoring	F_t	7500	N
Acceleration due to lunar gravity	g_M	1.625	m/s^2

7.3. Material Optimisation

To ensure that the habitat is light enough to be carried while meeting all the necessary requirements, the potential materials should be investigated and then chosen based on the set of characteristics most adapted to fill their role. The structure needs to be able to do the following:

- Protect the structural integrity of the habitat from any internal sources of damage
- Remain airtight to avoid pressure losses
- Protect the astronauts from GCR
- Resist the tension loads induced by the pressure differences between the habitat and the skin
- Provide thermal protection by minimising heat loss

The properties that are required to be able to fulfill each of these roles vary substantially, and no single material would be capable of fulfilling all of the needs. The proposed solution is to incorporate a multi-layer structure, with each layer being capable of dealing with one or more specific hazards.

Meteoroid impact and solar radiation protection are both dealt with best by having a large amount of mass and material volume in between the hazard and the volume. Since such a solution would be heavy and difficult to deploy, ISRU is the preferred option.

7.3.1. Habitat Layers

To safely tackle each of these aspects, the inflatable is designed with various layers of fabric, each one dedicated to stopping a specific hazard [109]. Crew related hazards are minimized using a lining, which must therefore be abrasion and fire resistant, depressurisation is avoided using multiple spaced layers of bladder, cosmic radiation is avoided using a custom layer of low-molecular-weight material, and structural integrity is maintained using a strong and lightweight layer. Finally, a thermal layer covers the inflatable to reduce damage due to UV radiation and heat during construction. Each layer must also be able to be folded and compressed to fit within the launcher.

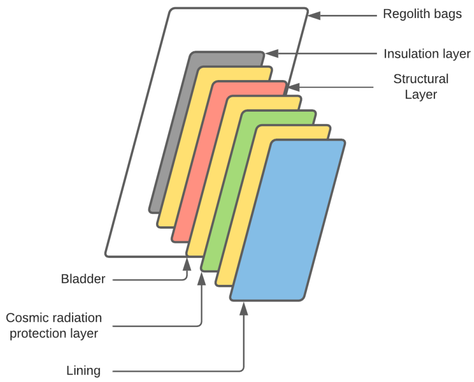


Figure 7.2: Schematic of the different layers of the habitat

7.3.2. From Fibre to Fabric Properties

Layers of the habitat will be made of fabrics, some of which are made from fibres. Translating fibre properties to that of fabrics adds additional layers of complexity. First, the fibres must be made into a yarn. This results in a loss of tenacity which depends heavily on the specific fibre and type of yarn [74]. The yarn is then weaved into a fabric, for which the pattern has a meaningful effect. A plain weave preserves most mechanical characteristics, while a ripstop weave protects against shear, and satins are often used for clothing for their specific texture. Literature focuses on geometric properties of fabrics and internal effects of the weave, rather than on mechanical properties such as strength and stiffness, which are generally covered qualitatively due to their dependence on the yarn. Existing models use non-deterministic models (genetic methods and neural networks) to make mechanical predictions, which is a layer of complexity beyond what this report aims to achieve [6]. This section details how the complexity of modeling fabric performance is tackled to achieve a sufficient level of correctness.

From fibre to yarn

The model employed to translate fibre properties to that of a yarn is based on a percentage loss in tenacity as predicted by a statistical analysis. Basing the analysis on a percentage loss allows for its use for various units, which is a key issue when discussing yarns and fabrics, due to the ample use of linear densities and other metrology conventions.

A 2012 study on yarns produced in mills [74] showed a linear relationship between compact yarn and fibre tenacity can be achieved with an R^2 value of 0.84. Setting an intercept for said relations yielded that tenacity can be approximated by taking 64% of fibre tenacity. The model did however underestimate tenacity loss for high tenacity fibres (>$45[cN/tex]$). The largest overestimation observed is of the order of 8%.

From yarn to fabric

Fabric modelling complexity begins with the interaction of yarns within a weave. Hence, the weave should be chosen first. Due to its relative simplicity and its generally superior mechanical properties [6], a plain weave is employed at this stage of the design. The next major factor is that of crimp, which is the ratio of fabric length to yarn length. Fabrics will always be shorter than the yarns, as the latter is compressed to fit the weaved shape as showcased in Figure 7.3. Additional crimp leads to additional density due to the additional fabric, but to a smaller Young's Modulus as it allows the yarns to slide between each other before being stretched.

Tensile strength of the yarn is also impacted by being placed in a weave. This is mainly due to the specific tensile behaviour of fabrics, have three major components: the initial region, where stiffness comes mostly from inter-yarn friction, the decrimping region. where stiffness lowers due to sliding of the yarns, and the yarn extension region, where the yarns stretch. These can be seen on the stress-strain curve on Figure 7.4. The final region covers approximately 50% of the entire stress/strain curve [33], but the final part thereof should be avoided, as it may cause jamming of the fabric, which would be detrimental to the functioning of the layers.

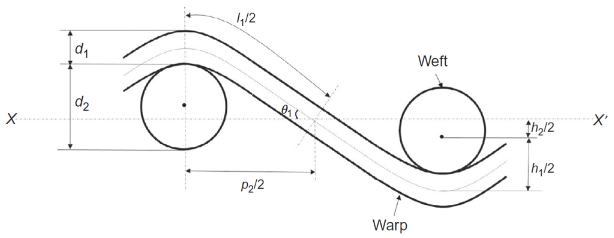


Figure 7.3: Schematic of a plain weave [6]

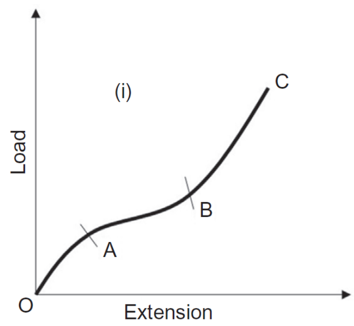


Figure 7.4: Common tensile behaviour of a fabric. Initial region [OA], decrimping region [AB], and yarn extension region [BC] are all exaggerated for visibility

This leads to the assumption that translating the strength of a yarn to a fabric leads to an effective loss of 50% tensile strength. This also assumes that the directional cover factor is close to 100%, which is achievable and desired as minimising thickness of the layers will make deployment more achievable [29]. Cover factor is divided into weft and warp cover factors, with one of the two directions requiring a smaller load using a pressurised cylinder approximation [39]. Having a cover factor of 100% in one direction and a smaller one in the other has the effect of reducing effective density of the fabric, but will also make it a less effective radiation shield [30].

The stiffness of the fabric depends on the applied load, but with the assumption that the fabric will avoid the yarn-extension region, a decrease in Young's Modulus can be foreseen. A geometrical idealisation of the fabric leads to the Young's Modulus being dependant on crimp and directional cover factor of the fabric. In this case, the ratio of warp and weft cover factors and the crimp both have an effect on the effective stiffness [30].

$$E_{fabric_1} = \frac{p_2}{p_1} \cdot \frac{E_{yarn}}{1 + c_1} \tag{7.1}$$

Where c_1 is the crimp in the direction of the stretch, and p is the cover factor.

This means that the modulus in the stronger direction will also be larger, which is desired to ensure that the structural layer carries most of the load.

The last mechanical characteristic being considered is that of density. The areal density of the fabric can be approximated using the cover factor [30]. The mass of fabric needed should be computed for each direction independently and then summed to estimate the total fabric mass required. The mass required does not translate directly into a thickness because of the weave, and a cover factor of 100% in one direction translates into twice the required thickness (but with no change in mass).

7.3.3. Layer by Layer Optimisation

The material for each layer is optimised to fulfill its required role. Using single materials to protect against multiple hazards is avoided at this stage of the design to ensure that each hazard is dealt with properly and material usage is done quantitatively and using mature technologies. It also avoids single points of failure, as failure of a single layer will allow the astronauts to survive and possibly perform maintenance activities.

Layers are made of fabrics, which have additional layers of complexity when compared to the fibres they are made of due to weave patterns and yarn unevenness (among other factors). However, the model employed to compute fabric properties assumes that fabric properties can be obtained using linear relationships, and therefore using fibre properties for comparison is justified.

Lining

The lining protects the habitat from damage from internal sources. It has to be abrasion resistant to avoid cuts and friction damaging other layers, and should also be fire resistant so that fires don't damage the bladder and cosmic radiation protection. As such, the main factors are abrasion resistance, the limiting oxygen index (LOI), and density.

High performance fabrics with sufficient LOI to be considered flame retardant (LOI>25) [30] are the following [1]:

- Para-aramids: LOI = 25-40
- Meta-aramids: LOI = 27-38
- Polybenzimidazoles (PBI): LOI = 41
- Polybenzoxazoles (PBO): LOI = 68
- Oxidized Polyacrylonitrile (PAN): LOI = 50-55

In terms of abraison resistance, aramids perform very well as showcased by their extensive use in protective clothing. PBO fibres such as Zylon also showcase favorable abrasion performance [2]. PBIs and PANs have only been extensively used in abrasion resistant fabrics when combined with other high performance fibres such as aramids [3], and are therefore not considered further for this application.

Hence, the considered materials for the lining are para-aramids, meta-aramids, and PBOs. Meta-aramids have the lowest density among these [108], and have previously been incorporated into inflatable structures as a lining [109]. Hence, Teijin-conex is used for the lining. Its mechanical properties can be found in Table 7.8.

Cosmic radiation protection layer

If not properly addressed, prolonged exposure to galactic cosmic rays can lead to multiple serious health complications for astronauts. Unlike solar radiation, GCRs can not be stopped by accumulating mass in between the astronauts and the environment. GCRs are composed mostly of small particles, and therefore require materials with small atomic numbers to be stopped. Research shows that optimal shielding can be measured by the ratio of electric stopping power to nuclear stopping power [73]. As such, the material index in Equation 7.2.

$$M_{rad} = \frac{Z}{\rho O^{2/3}} \tag{7.2}$$

A quick analysis of the formula reveal that liquid hydrogen would be the most well suited material. It should however be ruled out due to its non-compatibility with the chosen structure, which comes from its temperature range and tendency to evaporate. Hence, a solid material with large hydrogen content would be ideal. Low density polyethylenes (LDPE) have extensive previous usage as a radiation shield due to their high content of hydrogen atoms [80], and are therefore the main considered material of the layer. Table 7.8 gives an overview of its mechanical characteristics[4].

Bladder

The bladder layer has the main function of ensuring that the oxygen losses through the inflatable are minimised. Multiple layers will be implemented to provide redundancy. Each layer must therefore have very low permeability and be lightweight. Like all the other layers, it should also be a fabric so that the deployment mechanism works effectively. Materials used for food packaging tend to display most of these features.

A filtered search on Matweb [5] based on low oxygen transmission was performed. Ethylyne vinyl alcohol, nylon, polyester film, and polyethylene are the main competing materials, with Vectran also showing favourable permeability properties.

Since the habitat will need to be inflated while the materials are still ductile, only materials which retain their ductility at -60°C are considered.

[1]URL: https://www.teijinaramid.com/mooring-lines/96b27bfbff8f0ca54eaf038ff7efd299.pdf, accessed on 04/06/2021
[2]URL: https://www.toyobo-global.com/seihin/kc/pbo/zylon-p/bussei-p/technical.pdf, accessed on 04/06/2021
[3]UR: https://frontlinesafety.com.au/fire-fighting-trouser-pbi-matrix-gold, accessed on 04/06/2021
[4]URL:http://www.matweb.com/search/DataSheet.aspx?MatGUID=63842754b2f34872af5c4fe284069e20, accessed on 09/06/2021
[5]URL: http://www.matweb.com/search/PropertySearch.aspx, accessed on 08/06/2021

Table 7.2: Considered materials for the bladder along with main properties

Material	Permeability $[cc - mm/m^2 - 24hr]$	Density [g/cm^2]	Minimum temperature [°C]
Nylon film	0.5-9.14	1.13-1.15	-40
Polyester film	0.02-102	1.0-1.51	-70
Liquid crystal polymer (fibre)	0.04-0.064	1.38-1.77	-60
Polyvinyl Dicholride	0.01-0.12	1.26-1.78	-60

Polyvinyl dichloride (PVDC) is the most promising material family, as it consistently shows extremely favourable permeability properties. Among PVDCs, Saran is commonly used as an oxygen barrier already [6], and is therefore a well suited material for this layer of the habitat. Liquid crystal polymers are not considered as the sole bladder material as their permeability behaviour when woven into fabrics is likely to under-perform, and the necessary cover factor to avoid porosity would increase the weight required.

The precise properties of the layer are given in Table 7.8 [7] [8]:

Permeability can be further increased by metalising polymer exteriors [98] [9]. Metalisation of the surfaces is expected to only increase the layer weight by 5%[10], while potentially reducing oxygen transmission by a factor of 10 (assuming a thickness of 0.025 [mm]) [11].

Restraint layer

The restraint layer is meant to sustain the internal loads of the habitat, which will mainly be a distributed pressure load. It's main desired characteristics are therefore high strength and low weight. The material should also be able to withstand temperatures as low as -60°C, be abrasion resistant, and avoid loss of properties due to UV radiation. The latter can be mitigated using appropriate coating, or by adding an additional layer to the structure.

Fibres are compared assuming that they can are weaved into a plain weave, and that they will mostly experience tension. Effects of shear, out-of-plane forces, and an-isotropic effects are discussed after the selection.

The considered fibres for the fabric are chosen based on specific tensile strength and Young's modulus. Table 7.3 showcases the primary considered materials and material families for this application. Twaron and Technora are both para-aramids that were considered for the lining of the habitat. Dyneema SK99 (a novel form of Uutra-high-molecular-weight polyethylene) displays the best specific strength while also having a Young's modulus which is likely to be much be greater than that of the other structural layers. Although Dyneema's edge in terns of its physical properties is only slight, it also has a considerably smaller carbon footprint than its competitors[12], which justifies its use over fibres with more previous uses in the space industry.

Table 7.3: Mechanical properties of high performance fibres for the structural habitat layer. The chosen material (Dyneema SK99) is shown in italics.

Material	Young's Modulus (GPa)	Tensile Strength (GPa)	Density (g/cm^2)	Strength/Density
Twaron 1000	72	2.9	1.44	2.0
Twaron 1055	144	2.3	1.45	1.6
Twaron 2200	130	2.8	1.45	1.9
Technora	65-85	3.2-3.5	1.39	2.4
PBO	180-270	5.8	1.54-1.56	3.7
Dyneema SK99	*178*	*4.4*	*0.98*	*4.5*
Carbon	230-550	3.3-6	1.73-1.91	3.1 (max)
PBI	5.6	0.4	1.4	0.3

The main disadvantage of Dyneema is its temperature performance, with it having a lower working temperature range than para-aramids or Technora (-150° C to 140 ° C versus -200 ° C to 250 ° C for Twaron) [59]. This

[6] URL:http://www.matweb.com/search/DataSheet.aspx?MatGUID=edfa48fdb4c04f839a31cef344deb159, accessed 09/06/2021
[7] URL: http://www.matweb.com/search/DataSheet.aspx?MatGUID=edfa48fdb4c04f839a31cef344deb159, accessed on 15/06/2021
[8] URL: http://www.matweb.com/search/datasheet.aspx?matguid=57c29e222a7749d58267c18e9e18b637, accessed on 15/06/2021
[9] URL:https://www.polyprint.com/understanding-film-properties/flexographic-otr/, accessed on 20/06/2021
[10] URL:https://galvanizeit.org/design-and-fabrication/design-considerations/weight-increase
[11] URL: https://www.polyprint.com/understanding-film-properties/flexographic-otr/, accessed on 18/06/2021
[12] URL: https://www.dsm.com/dyneema/en_GB/applications/ropes-lines-slings-chains/mooring-and-tow-ropes/low-environmental-footprint.html, accessed on 15/06/2021

is most likely also the reason why it has had limited space applications. However, the chosen habitat location exhibits a temperature range well within Dyneema's optimal conditions. Only the SK99 variant is considered as it's the strongest and stiffest per unit weight, and therefore automatically outperforms other variants [13].

Fatigue performance is neglected, as high performance fibres tend to last over 30,000 cycles with amplitudes and frequencies much larger than what is expected for the habitat [99]. Properties of the fabric are presented in Table 7.4. The properties assume that a crimp factor of 8% is used, which falls in line with that of other woven fabrics[14]. Table 7.4 summarises the main fabric properties that should be included in the structural calculations. In the structural calculations it is assumed that the cover factor is equal in both directions, hence $p_r = 1$.

Table 7.4: Mechanical properties of Dyneema SK99 fabric. p_r is the ratio of material needed in the weft and warp directions.

Tensile Strength	**Young's Modulus (weft)**	**Young's Modulus (warp)**	**Density**
1.41 GPa	165 p_r GPa	165 $\frac{1}{p_r}$ GPa	0.98 kg/m^3

Insulation layer
The insulation layer protects the inner layers from radiation, cold temperatures, the vacuum environment, and other hostile activity on the Moon. Traditionally, spacecraft have used polyimide and polyester film layers to achieve said results [15]. To avoid failure of the layer, more material than is strictly necessary to cover the habitat will be used. This will help in avoiding failure both due to tension and impacts of regolith bags. Opacity to UV radiation is also valued, but can be compensated for with coatings.

The filtered material search on Matweb is therefore focused on low thermal conductivity, low density, and a minimum temperature below -60° C. Among the results, foams and rigid materials were neglected to prevent deployment issues along with materials that were not suitable to create a fabric-like structure. Polyster and Polyimide were found to be the best performers.

Polyimide film includes Kapton variants and Upilex, with Kapton being the most technologically mature for space applications. Polyester films are restricted to Mylar, as other types have limited data concerning thermal conductivity or surpass the maximum value. Among each of these new material families, individual examples tend to have very similar thermal conductivity and density (e.g. all forms of Kapton have a thermal conductivity of 0.12 [$W/m \cdot K$]).

Table 7.5: Considered material families for the thermal insulation layer. Figures taken from Matweb

Material	**Thermal Conductivity ($W/m \cdot K$)**	**Density (g/cm^3)**	**Specific heat capacity ($J/g \cdot C$)**
Kapton	0.12	1.42	1.09
Upilex	0.289	1.47	1.13
Mylar	0.155	1.39	1.17

Table 7.5 shows that Kapton and Mylar are both more adapted than Upilex for this specific use. This falls in line with the expected result, being a combination of polyimide (Kapton) and polyester (Mylar). Both of these materials have been used in conjunction with each other for space applications and have proven to also provide valuable impact resistance [16].

Hence, the insulation layer will be made with a combination of Kapton and Mylar. The resulting layer is assumed to have the average characteristics of each of these materials, as seen in Table 7.8.

This layer would also benefit greatly from being metalised to decrease overall permeability and heat flow while increasing radiation protection [17]. The process is expected to increase the weight of the layer by approximately 5%. The change in emissivity (up to 0.004) would also be very beneficial in the building phase of the mission[25].

[13]URL: https://www.infexion.eu/info/types-of-dyneema/, accessed on 04/06/2021
[14]URL: https://textilelearner.net/crimp-percentage-of-woven-fabric/, accessed on 07/06/2021
[15]URL: https://medium.com/teamindus/space-blankets-and-how-the-teamindus-spacecraft-will-survive-in-space-21f779ca7291, accessed on 08/06/2021
[16]URL: http://pluto.jhuapl.edu/Mission/Spacecraft/Systems-and-Components.php, accessed on 08/06/2021
[17]URL: https://www.dunmore.com/products/aluminized-polyimide-film.html, accessed on 15/06/2021

7.4. Additional Structural Elements

In this section, additional structural elements that were not discussed in detail in the Midterm Report are described. These are the flooring of the habitat and the structural reinforcement element.

7.4.1. Flooring

The details of the flooring of the habitat has to be determined, especially for the part of the habitat that will be inflated on the Moon. This part of the flooring, which is 21 m long, will be an inflatable structure as well. This makes the deployment easier, since it can be folded up with the habitat.

For its strength, a tensairity® slab structure will be used for the flooring[18]. A tensairity® structure is an inflatable with additional reinforcement elements that strengthens the inflatable against bending and buckling. Precisely, the inflatable flooring will have cables in tension that wrap around the flooring which will strengthen the flooring as mentioned.

The flooring will be connected to the internal wall of the habitat and rolled up too. However, to prevent the shear stress from becoming too large on the walls, there will be load-bearing boxes in the storage area, with dimensions $1.3 \times 1.2 \times 0.8$, to transfer the load from the floor to the bottom of the habitat. These load-bearing boxes will be empty, with stringers on their walls to increase their stiffness and resistance to bending and buckling. They are also slightly larger than the storage boxes to provide clearance for the storage boxes.

Since details on tensairity® slab structures are limited, and as the inflatable can be designed in preferred dimensions, the parameters for the flooring were obtained with comparisons to another tensairity® slab structure. It was determined that the flooring will have a thickness of 1.5 mm, and they will be 10 cm thick when fully inflated. Similar materials will be used for the flooring, but the flooring will only consist of a lining layer and a restraint layer. The flooring is placed 1.3 m above the flat bottom side of the habitat, which makes the length of the flooring to be 5.77 m. The estimated mass of the flooring over the 21 m is 2000 kg.

7.4.2. Structural Reinforcement Element

A structural reinforcement element is required in case of an emergency that leads to a collapse of the habitat, such as deflation of the inflatable, which is dangerous since the regolith bags will crush the habitat. Hence, there should be a structure that can hold the load of the regolith bags in case the habitat is deflated, at least for a certain duration that can buy time for the astronauts to escape the habitat.

It was determined that the walls of the habitat that the astronauts set up inside after their arrival will act as the structural reinforcement element, and they will transfer the weight of the regolith bags to the load-bearing boxes under the flooring. Also, the walls and the ceiling of the personal rooms in the back will act as supporting structures that serve the same purpose. These structures will prevent immediate collapse of the habitat, and will buy time for the astronauts to evacuate.

7.5. Inflatable Mass Estimation

Estimating the mass of the inflatable is a necessary step to ensure that the entire mission can be launched as planned. The calculations assume an idealised cylindrical shape with endcaps, as this summarises the majority of the inflatable mass, The airlock mass is calculated separately.

The total surface area of the inflatable is therefore estimated to be:

$$A_{\text{inflatable}} = 2\pi rl + 2\pi r^2 = 565.5[m^2] \tag{7.3}$$

By using a thin walled approximation, the total volume of each layer can simply be calculated by multiplying the thickness of the layer with the inflatable surface area. Hence, the thickness of each layer must now be calculated.

7.5.1. Lining thickness

The lining can be quite thin, as a thin layer would already be abrasion and fire resistant. Assuming that two layers are used, in case the inner-most one suffers damages, and that typical fire-retardant para-aramids have thicknesses surrounding 0.2[mm], a total thickness of 0.4 [mm] is assumed. This leads to a total lining mass of 0.031 [t].

[18] An expert suggestion from Joep Breuer during a discussion about the flooring of the habitat. More information about tensairity® structures can be found on https://www.tensairitysolutions.com/.

7.5.2. Radiation protection thickness

As detailed in Chapter 4, a layer of 2 [cm] of polyethylene, along with the layer if regolith, provides sufficient GCR protection to fulfill the corresponding requirements. Hence, a layer of 2 [cm] is chosen. Current SPENVIS simulations suggest that the figure could be lowered further in future iterations, but the current thickness is not a problematic aspect and therefore additional GCR protection is worth including.

7.5.3. Bladder thickness

Saran has a permeability as low as 0.032 [$(cm^3/m^2) * mm/24h$] at one atmosphere of pressure, meaning that a sheet that is 1 [mm] thick and with an area of 1m^2 would transmit 0.032cm^3 of oxygen over a period of one day.

Metalising the layer would also lower oxygen transmission, as it only affects the surface of the layer while considerably permeability. Permeability losses due to metalisation are measured using 0.025[mm] layers of material, which is thin enough to assume that it would not affect the entirety of the layer, but would instead serve as an additional small layer [19]. Assuming metalisation decreases permeability by a factor of 10 in a section that represents $\frac{1}{40}$ of the entire layer, the entire permeability can be estimated. Adding a 0.25 [mm], metalised layer would decrease the oxygen transmission to 0.013 [$(cm^3/m^2) * mm/24h$] at one atmosphere.

Accounting for the inflatables dimensions, this leads to a total loss of oxygen of 2.8 [l] per year. Nitrogen loss is 20 to 40% of oxygen loss in the habitats conditions, meaning that a conservative estimate nitrogen loss would be of 1.1 [l] per year.

Hence, a thickness of 1 [mm] is sufficient, even if the specific conditions of the habitat were to increase the number by an order of magnitude.

Since layers beyond the first are used for redundancy rather than for decreased permeability, every layer is set to be 1 [mm] thick, leading to a total layer mass of 0.32 [t] (accounting for the increase in weight due to metalisation).

7.5.4. Restraint layer thickness

The thickness of the restraint layer could be determined by estimating the maximum amount of stress expected in the inflatable skin, and sizing the layer thickness to withstand the estimated stress value.

It was assumed that the maximum amount of stress in the habitat will be when the habitat is initially inflated fully. This is because once the regolith bags are stacked above the habitat, they will counteract the pressure load and decrease the stress in the skin. The anchoring system also counteracts the pressure loading and further decreases the expected stress. Hence, the skin of the inflatable was sized for the scenario that the habitat is fully inflated without any other loading than the internal pressure. A safety factor of 4 was initially used to size the thickness of the restraint layer.

A set of assumptions were used in the stress calculations. They are as follows:

- The pressure is distributed equally around the circular cross-section of the habitat.
- There are no manufacturing defects or shape deformations that may cause a stress concentration at any local point.
- The cross-section of the habitat is constant.
- The excavation of the habitat location is done perfectly, with no rocks or unusual shapes that can affect the shape of the habitat.

To calculate the stress through the skin, all of the layers and their material strengths were considered. Since the load path would be through the stiffest layer, this was determined as well, and was confirmed that the restraint layer is the stiffest and will be the load path through the skin of the inflatable.

Thicknesses of each layer which were known already were taken into account, and for thicknesses that weren't known yet, an arbitrary value was assumed and the results were iterated if the values changed. Then the thickness ratio of each layer was calculated, and the tensile strength of the entire inflatable skin was calculated using Equation 7.4. This is important since the tensile strength of the inflatable should be used for the stress comparison and not the tensile strength of individual layers, and the tensile strength of inflatable skin matters on its thickness ratios and the constituting layer's tensile strengths.

[19] URL: https://www.polyprint.com/understanding-film-properties/flexographic-otr/, accessed on 28/06/2021

$$\sigma_{inf} = \sum \sigma_{layer} \times \frac{t_{layer}}{t_{inf}} \tag{7.4}$$

For a pressurised cylindrical structure, the highest stress is the hoop stress, which can be calculated with Equation 7.5. Longitudinal stress is one half of the radial stress value, and shear stress is a quarter of the radial stress. The stress values were compared to the tensile strength of the combined layers. The result is shown in Table 7.6.

$$\sigma_r = \frac{p_i r}{t} \tag{7.5}$$

Table 7.6: Calculated stress values and strength of the inflatable

Parameter	Value	Unit
Hoop stress	10.6	MPa
Longitudinal stress	5.3	MPa
Shear stress	2.7	MPa
Tensile strength of the inflatable skin	78.6	MPa

Even considering the safety factor of 4, the inflatable tensile strength is much greater than the maximum stress expected in the skin. This surplus tensile strength can account for any localised stress concentrations that were not included in the calculations, as described in the assumptions above. It can be concluded that the habitat is structurally stable and will not fail under normal operation conditions.

7.5.5. Insulation layer thickness

To calculate the thickness of the insulation layer of the inflatable, the first step was to calculate the heat flow in and out of the habitat. A preliminary thickness value for the insulation layer was selected to compute the heat flow, and an iterative calculation process was conducted with optimising the thickness of the insulation layer at each iteration.

The heat flow of the habitat could be separated into three main groups: heat flow in and out of the habitat to the environment, heat generated by internal systems, and heat generated by the astronauts.

Firstly, the chosen habitat location has a temperature range between -60°C and 40°C. These were set as the lowest and highest temperature of the environment. Since all the machinery can operate at this temperature, insulation was calculated with the operational phase in mind, and not the manufacturing and assembly, since the operational phase was deemed more critical.

The heat gain and losses to the environment were calculated, with four distinct scenarios. These scenarios are: lunar day with incident solar radiation; lunar day, in the shadows; lunar night; and finally, heat lost due to conduction with the lunar surface, below the habitat. The last scenario will be further referred to as sub-surface heat loss.

The surface area that equates to each of those scenarios were then calculated. For the sub-surface heat loss, this includes the area of the cylindrical inflatable that directly touches the lunar surface, and the segments of the circular cross-section in front and at the back that are underground. For the lunar night scenario, this includes all the exposed surface above the lunar surface. For the lunar day with incident solar radiation scenario, this was calculated as half of the cylindrical habitat's exposed surface area, while the lunar day in the shadows scenario was calculated as the other half of the cylindrical habitat with the exposed surface area of the circular cross-sections on both ends.

The internal temperature of the habitat has been set to 22°C for optimal living conditions. With this value, the temperature difference for each scenario was calculated. Also, the conductivity of the exposed section and underground sections were calculated using Equation 7.6.

$$\frac{1}{\text{Conductivity}} = \sum \frac{1}{\text{Surface heat transmission coefficient}} + \sum \frac{\text{Layer thickness}}{\text{Layer thermal conductivity}} \tag{7.6}$$

Then, the heat flow in and out of the habitat could be calculated using Equation 7.7. The conductivity and the surface area of the entire section for the given scenario were used for the calculation.

$$q = C \cdot A \cdot \Delta T \tag{7.7}$$

The internal heat gains from the power use of electrical devices inside the habitat were calculated too. It was assumed that the electrical systems will eventually turn most of its energy into heat, unless if it was clear that they would convert the energy to light energy or potential energy, then their efficiency was taken into account. Since the internal systems work differently at day and night, the internal heat gains were calculated separately for both lunar day and lunar night scenarios. The power use of the internal systems are described in Section 9.2.1.

The heat generated by the astronauts is a difficult parameter to estimate, since it will vary diversely per person, and their activity. However, it was possible to find the internal heat gains report for a simulated LUNARES habitat [55], where the precise heat generation of the inhabitants were calculated. Using information from this report, an average value of 250 W was used as the occupational heat load generated per person.

Then, the total heat flow in and out of the habitat could be calculated, for lunar day and lunar night scenarios. The thickness sizing of the insulation layer then could be done, in parallel to the sizing of the thermal control unit. An iterative process to calculate the optimal thickness value with mass and insulation capability was conducted, along with sizing of the thermal control unit.

Table 7.7 shows the heat values that are related to the habitat operations during its normal operation. A positive heat value indicates that the habitat has gained or has received heat, while a negative value indicates that the habitat has lost heat. In conclusion, a maximum of 8300 W should be removed from the habitat per day with a thermal control unit.

Table 7.7: Values related to sizing of the insulation layer

Parameter	Value	Unit
Temperature inside the habitat	22	°C
Temperature of area during lunar day with incident sunlight	40	°C
Temperature of area during lunar day in the shadows	17	°C
Temperature of area during lunar night	-60	°C
Temperature of the environment at 1m deep underground	-23	°C
Thickness of insulation layer	5	mm
Heat lost to the underground surface	-60	W
Total heat flow from environment during lunar day	210	W
Total heat flow from environment during lunar night	-1287	W
Heat produced by one astronaut per day	250	W
Total internal heat gained during lunar day	7000	W
Total internal heat gained during lunar night	7700	W
Total internal heat flow during lunar day	8210	W
Total internal heat flow during lunar night	7413	W

7.5.6. Summary of inflatable layers

Table 7.8 below shows the summary of the layers used in the inflatable skin, in the order they are set in the inflatable. The bladder layer exists between every layer.

Table 7.8: Summary of the materials used for the inflatable layer

Layer	**Purpose**	**Chosen Material**	**Density** [g/cm^2]	**Tensile Strength** [MPa]	**Young's Modulus** [GPa]	**Other properties**	**Thickness** [mm]	**Number of layers**
Lining	Protect habitat from internal hazards	Teijin-conex	1.38	0.51-0.86	7.9	LOI = 27-38	0.2	2
Bladder	Minimise oxygen loss through inflatable skin	Saran	1.78	89.6	0.55	Permeability = 0.013 $cc - mm/m^2 - 24hr - atm$	1	3
Radiation protection	Protect astro-nauts from galactic cosmic rays	LDPE	0.922	10	0.32	High hydrogen content	20	1
Restraint	Sustain the main loads	Dyneema SK99	0.98	1410	165		0.2	1
Insulation layer	Minimise heat flow through inflatable skin	Aluminised Mylar and Kapton	1.48	124	3.5	Thermal conductivity = 0.137W/mK emissivity = 0.004	5	1

With these thickness values known, the mass of the inflatable skin can be calculated.

Table 7.9: Summary of key habitat parameters

Parameter	**Value**	**Unit**
Inflatable length	27	m
Inflatable radius	3	m
Inflatable height	5.12	m
Length of flat surface on the bottom	4.25	m
Inflatable mass	15800	kg
Inflatable skin thickness	28.6	mm
Floor mass	2000	kg
Total structural mass	17800	kg

7.6. Regolith Bags

Regolith bags are used in the habitat to offer constant passive solar radiation protection, as well as thermal and meteoroid protection, while minimising the launch mass and launch volume of the mission.

7.6.1. Required Regolith for Radiation and Thermal Protection

Regolith properties are summarised in Table 7.10. A density of 1.5 g/cm^3 is used in all further calculations, as the regolith will be dug from the top layer, which tends to be less dense [82].

Table 7.10: Main properties of lunar regolith [75][82]

Property	**Value**
Density	1500-1600 kg m^{-3}
Porosity	~45%
Cohesion	2.35 kPa
Angle of Internal Friction	18.5°
Bearing capacity	31 kPa
Thermal Conductivity	0.9-1.6 W/mK
Specific Heat	750-1000 J kg^{-1} K^{-1}
Specific Area	0.5 m^2 g^{-1}
Dose Reduction per unit density (g cm^{-3})	0.7~1%

Chapter 4 showed that having one meter of regolith covering the habitat would be more than sufficient to protect the astronauts from meteoroids, solar radiation, and the cold temperatures. Hence, the design will incorporate it in the form of regolith bags stacked around and over the habitat. Bags are chosen over simply distributed regolith to reduce the amount of digging required, to improve the structural stability, and to reduce the amount of loose regolith in contact with the habitat.

The total volume of regolith required can be computed. To do so, the cross section of the habitat is drawn along with the required regolith. It is assumed that stacking the bags at 45° is sufficient to maintain structural integrity, which is a conservative assumption considering that similarly shaped bags have proven to be stable in much more vertical structures [99]. Two parts of the habitat are covered in regolith: the main living space and the inner-most chamber of the airlock. These can be visualised in Figure C.1 and Figure 7.10 respectively. Both cross-sections. along with the regolith covering, can be seen below.

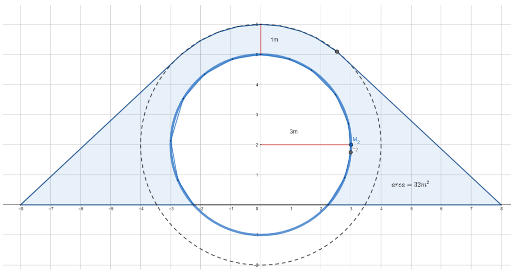


Figure 7.5: Cross section drawing of the regolith over the livable section of the habitat. Dotted lines show the minimum amount of regolith required, and the blue area shows where the regolith bags would be

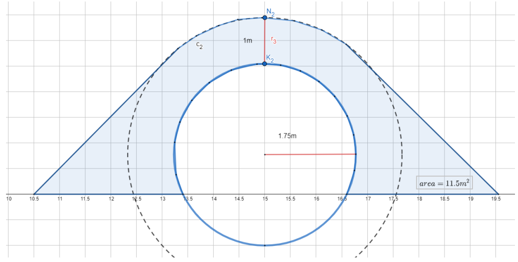


Figure 7.6: Cross section drawing of the regolith over the innermost airlock chamber. Dotted lines show the minimum amount of regolith required, and the blue area shows where the regolith bags would be

Figure 7.5 and Figure 7.6 show that the cross sectional regolith area required are 32m^2 and 11.5m^2 for the living space and the airlock chamber respectively. The living area is 27 [m] long and the airlock chamber is 3 [m] long, meaning that a total of 900m^3 of regolith are required to fully cover the habitat. Assuming a density of 1.5 [g/cm^3], as would be expected when digging one meter into the ground[82], a total mass of 1,350 [tons] of regolith needs to be dug up and placed in bags.

7.6.2. Regolith Bag Shape and Size

The shape of the bag should be chosen so as to minimise the amount of fabric needed while also avoiding unnecessary stitches, as these are natural failure points. The most volume per unit of surface area would be found in a sphere. However, these would not be easily stackable. A cubic shape also has a favorable volume to surface area ratio and is much easier to stack, but requires multiple stitching lines and is unlikely to conserve its shape. A 'pillow' shape was therefore chosen, as is most common for sandbags on Earth. These provide good stack-ability, few stitching lines, and can be made to a large size.

The size of the bags should be maximised, as more bags entails a smaller total fabric surface are. However, the chosen to ensure that at least two layers are placed on top of the habitat, as the accumulation of layers increases the thermal efficiency. The weight of the bag filled with regolith should also be able to be carried by a crane while not slipping or causing the bag to fail due to the weight it carries. Hence, the bag height should be of the order of magnitude of 50 [cm], and the length should be sufficient to comfortably remain over the habitat.

The following dimensions are chosen:

- Length: 200cm
- Width: 50cm
- Height (when filled): 50cm

The dimensions are similar to those found to those employed in a self-standing structure experimentally build by NASA as part of a lunar garage prototype [99], which shows that the bags can be made structurally stable even without a supporting inflatable.

7.6.3. Regolith Bag Mass Estimate

To estimate the mass, first the bag material must be chosen. The material must provide good specific strength, be UV radiation resistant, and suffer low wear due to abrasion with regolith, machinery, and other bags.

The main considered materials are therefore high-performance fabrics, as they can provide great strength characteristics at a very low weight, and tend to have favorable abrasion and fatigue characteristics. Hence,

Table 7.3 is used to compare candidates. Additionaly, UV resistance characteristics are compared using literature [99] [20].

When accounting for UV resistance on top of the considerations for the restraint layer, Dynerma SK99 is again the most appealing option. It also shows favorable abrasion, chemical, and temperature properties[21] and shouldn't suffer from its main disadvantage of having a lower maximum operating temperature than its competitors. Hence, properties of the bag fabric can be found in Table 7.4.

The next component of the mass comes from the amount of bags that are required to ensure that the entirety of the habitat is covered in regolith. A total regolith volume of 717m^3 was calculated. Hence, a total of 1435 bags will be required, leading to a total fabric surface area of 9327m^2.

Finally, the last step is determining the thickness required in each bag to feasibly hold the regolith in place. First, the thickness required to avoid failure assuming that the bag is grabbed from one tip is calculated.

$$A_{cs} = \frac{F}{\sigma_{f_{SK99}}}$$

$$t_{bag} = \frac{m_{bag} \cdot g_{Moon}}{2\sigma_{f_{SK99}}} << 0.1[mm] \tag{7.8}$$

Hence, weight of the regolith is not a constraining factor when deciding the bag thickness. The thickess is therefore taken to be 0.2 [mm], which is similar to other bag thickness estimates in similar environments and accounts for other loads the bags would endure [4][10]. Multiplying this figure by the total surface area required and by Dyneema SK99's density leads to a total empty bag weight of 1.83 tonnes. These section results are summarised in Table 7.11.

Table 7.11: Regolith bag summary table

Regolith required	**Bag size**	**Number of bags**	**Fabric area**
1,347t	50 x 50 x 200 cm when filled	1800	11680m^2
Fabric Thickness	**Fabric Density**	**Total Fabric Weight**	**Fabric to ISRU weight ratio**
0.2mm	0.98g/c^3	2.38t	0.17%

The fabric is recommended to be woven into a ripstop pattern, as the loss of strength when compared to a plain weave is not the most constraining factor, and stopping rips from developing further into the bag is desirable for structural cohesion and to ensure better thermal insulation. The difference between these weaves can be seen in Figure 7.7.

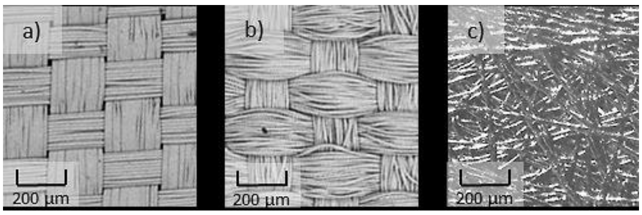


Figure 7.7: Comparison of plain woven (a), ripstop woven (b), and non-woven (c) fabrics[21]

7.7. Airlock

The airlock ensures that pressure losses are minimised through the operational life of the habitat, by providing a room separate from the inflatable that is capable of being pressurised and depressurised. Although current trends in airlock technology aim at making airlocks themselves inflatable[70], the added security of a rigid structure containing rails, rigid staircases, and easier access to spacesuit systems make the latter a more adapted choice. As such, the airlock is designed considering that it should be rigid.

7.7.1. Airlock Design

For the airlock, the double chamber airlock design has been chosen. This concept is chosen over the single chamber airlock because of the following reasons:

[20] URL: https://www.marlowropes.com/material-properties, accessed on 11/06/2021
[21] URL:file:///C:/Users/dsanc/Downloads/Dyneema_Folder_201611_Gruschwitz_63994.pdf, accessed on 11/06/2021

- Safety: In case of a seal failure in the double chamber airlock, it can still operate as a single chamber airlock.
- Dust mitigation: The outer chamber will be used for cleaning of regolith dust.The inner chamber (the one connected to the habitat) will be used for donning an doffing.One of the requirements set for the mission is **ST-SYS01-11**: No regolith shall enter the habitat when astronauts return from an extra-vehicular activity (EVA). With this system regolith contamination of the habitat will be eliminated.
- Decompressing and compressing takes time. A single chamber airlock can not be used for doffing, donning or suit maintenance when other astronauts are using the airlock to leave or enter the habitat. With the double chamber design, astronauts can be leaving the habitat in the outer chamber while other astronauts are doing suit maintenance in the inner airlock simultaneously.

The first (inner) part of the airlock will be larger then the outer. The inner chamber will be sized for crew maneuverability. Donning an doffing need to be done here as well as suit maintenance and suit storage. A total of four Mark III suits will be stored here. More on these suits in the life support chapter Section 10.5. The outer part solely need to accommodate the astronauts in their space suit. It therefore can be designed much smaller. The outer airlock chamber will is designed for two astronauts to enter or exit the airlock at a time.

Airlock Cycles

Entering and leaving the habitat trough the airlock must be done in a save way and in such a way to minimise air volume losses. Therefore the following procedure is performed when going trough the airlock. The inner chamber connected to the habitat is called chamber 2. The outer chamber which gives access to the lunar surface is called chamber 2.

Entering the habitat: Bot chambers will be non-pressurized (np) when starting the procedure.

- Step 1: Astronauts enter chamber 1. There they can use high pressure air to clean the regolith off their suits. The air in this chamber will be constantly circulating trough regolith-proof filters in order to be cleaned
- Step 2: Astronauts enter chamber 2. The hatch between chamber 1 and 2 will be closed afterwards
- Step 3: Chamber 2 will be pressurized. After this, the astronauts can start putting-off their space suits
- Step 4: The hatch between chamber 2 and the habitat will be opened, the astronauts can enter the habitat

Leaving the habitat: Both airlock chambers will be in the state they have been left for after Step 4 in entering the habitat.

- Step 1: Astronauts enter chamber 1, simultaneously chamber 2 will be compressed. While this is happening the astronauts put on their suit. The pressure in chamber 1 is constantly kept a little bit higher then the pressure in chamber 2. This is to create a constant airflow into chamber 2 once the hatch between them is opened. Chamber 2 will still have some minimal regolith contamination. The airflow will prevent regolith from going into chamber 1.
- Step 2: The hatch between both chambers is opened and the astronauts enter chamber 2. The hatch is closed behind them.
- Step 3: Both chambers are decompressed. Note that as the bigger chamber is sealed-off from the lunar environment lead to air volume loss.
- Step 4: The hatch to the lunar surface is opened and the astronauts move outside the airlock.

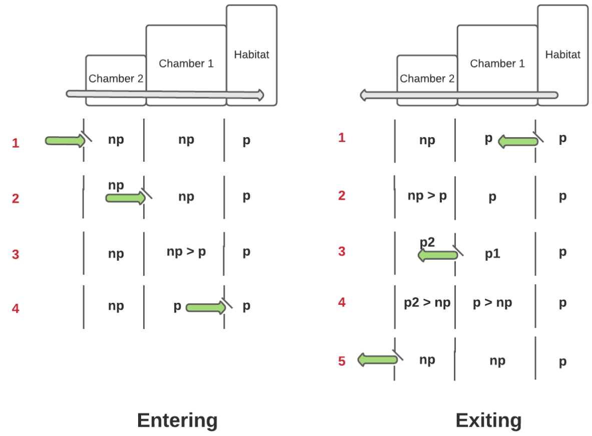


Figure 7.8: Airlock cycle steps

7.7.2. Airlock Dimension

Both airlock chambers are designed based on the Mark III suit dimensions. These can be find in the life support chapter Section 10.5. In the figures below one can see the airlock dimensions. A minimal ceiling height of 2 meters is required for the astronauts to maneuver while in their space suit.

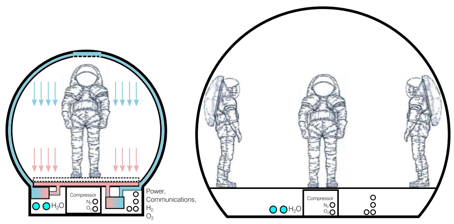


Figure 7.9: Front view of the airlock, showing the necessary equipment and pipes to connect with outside systems.

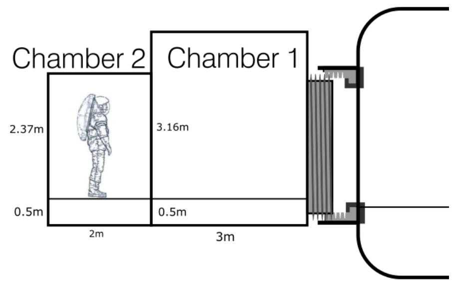


Figure 7.10: Side view of the airlock including the dimensions

Airlock equipment and outside connections

Below both chamber floors, 50 cm height of space will be available for a compressor. This space will be also used for running wires and tubing from outside the habitat to inside the habitat. In chamber 2 additionally, this space will also be used for air circulation and regolith filters for decontamination. Air will be blown from the top over the astronauts towards the floor. The floor structure is made from a mesh material letting air trough after which the air is guided towards a regolith filter. After this filter comes the compressor which circulates the air around. In figure 1.5 on the left the circulation is illustrated. Trough the airlock floors and to the lunar environment go hydrogen and oxygen tubing to reach the cryogenic storage tanks, a power cable from the solar field, communication cables to the antenna and two water tubes (waste water and fresh water) for resupply (after the one year mission). These all run trough the habitat, trough the floor of airlock 1, trough the floor of airlock 2 and to the left and right exiting airlock 2. The water exits on the right while all other tubing exits on the left. This is done so that in the future a lunar vehicle can dock against the water tubing port on the right and after which it can pump water in or out the habitat.

7.7.3. Airlock Radiation Protection

In chamber one astronauts will walk around without their suits on. Therefore it needs to be protected from radiation. Chosen is regolith shielding just like the main habitat. In chamber 2 astronauts will enter with their suits on. Therefore no regolith cover is needed on chamber 2.

7.7.4. Airlock Mass Estimate

The mass of the airlock itself is computed through the use of existing analogous airlocks. The main point if reference is the Dual-Chamber Hybrid Inflatable Suitlock (DCIS)[43], as it was designed for planetary bases and is similar in size to that which is required for the mission. It also contains a valuable folding mechanism which allows for more flexibility for the mission logistics. Moreover, it is extremely similar in scale and therefore presents few interpolation challenges.

The DCIS weight can be estimated by comparing it to the Quest Joint Airlock used on the ISS, which weighs 9900 [kg]. Previous research estimated that the DCIS would be both lighter and less complex than the Quest, and therefore its mass is estimated to weigh 90% of a Quest airlock scaled down to the size of the DCIS. Hence, a 13.3m^3 DCIS is estimated to weigh:

$$M_{DCIS} = 0.9 \cdot \frac{13.3}{34} \cdot 9900 = 3485[kg] \tag{7.9}$$

Since the outermost chamber a volume of 13.02m^3, the same weight as the DCIS is employed to account for a more rigid design that doesn't incorporate a suiting. The suiting chamber mass calculated analogously.

$$M_{chamber1} = \frac{V_{chamber1}}{V_{chamber2}} \cdot m_{chamber2} = 10.9 \cdot 10^3[kg] \tag{7.10}$$

Hence, the total airlock mass is 14.4 [t].

7.7.5. Integration of Airlock and Inflatable Structure

Integrating the airlock, which is a rigid structure, to the inflatable structure is an important section within the assembly part of the project. The two structures have to be integrated in such a way that it is airtight and can withstand the shear forces being exerted when the inflatable structure is inflated. There are several possible solutions. One solution would be applying adhesive bonding between the inflatable structure and the rigid airlock, then bolting certain sections of it to prevent peeling of the inflatable structure [22]. The advantage of this method is that it adds relatively little weight to the overall habitat structure, which would reduce the launch cost. However, the downside of this method would be that it requires drilling of both the airlock and the inflatable structure to install the bolts, which would not be ideal in keeping the habitat airtight. Further reinforcement would be necessary near the area of drilling. The second method is the method which uses a male and a female coupling, just like the method used in the fire hose. It requires a male coupling on one end and a female coupling on the other end to link the inflatable structure and the airlock together. There will also be a silicon rubber O-ring on the male coupling section, which seals the habitat, making it airtight. Although this method is heavier to bring to the Moon, it is more durable and reliable in keeping the habitat safe, which is more important. Figure 7.11 shows how the airlock and the inflated structure will be coupled together with a male and a female coupler. In between the couplers, there should be a silicon O-ring and the pre-assembled section will be in vacuum before it goes into the launcher.

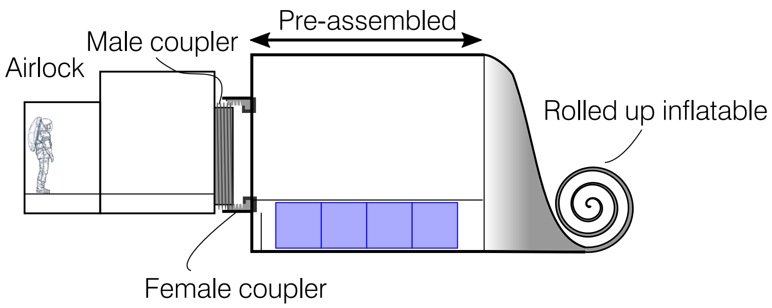


Figure 7.11: Coupling System of the Airlock and the Inflatable Structure

Since the design is modelled after that of a hose connector, it makes sense that the material out of which its made is analogous. The main employed materials are stainless steel, aluminium, and brass [23]. Aluminium is chosen due to its superior strength to weight ratio.

The connection also requires an elastomer to be used as part of the male connection, which would allow it to be airtight. The elastomer should be capable of maintaining its elastic properties even at temperatures as low as $-60°C$. Silicone rubber products are the most adapted to this specific use[24] [25], and are therefore the material of choice.

[22] Citation: Private communication with Dr Otto Bergsma, Faculty of Aeropsace Engineering, TU Delft, The Netherlands
[23] URL:http://www.frankberg.nl/storz-couplings.html, accessed on 16/06/2021
[24] URL:http://atevitratech.com, accessed on 20/06/2021
[25] URL:http://www.matweb.com/search/DataSheet.aspx?MatGUID=f1ecff5f176749faa6d6840abf942683&ckck=1, accessed on 20/06/2021

7.8. Manufacturing and Assembly on Earth

As some manufacturing and the assembly procedures are too complex to perform on the moon, they'll take place on Earth. Such procedures would include the manufacturing process of the inflatable structure and assembling it with the airlock. After the inflatable structure has been integrated with the airlock, the entire structure would have to be packed in an efficient method to fit into the launcher. The following texts will be explaining how these procedures will be implemented.

7.8.1. Manufacturing of the Inflatable Structure

The manufacturing process of the inflatable structure takes place on Earth, which makes the process rather easier to perform as abundant resources are available on Earth compared to the Moon. The first method uses the process of calendaring, seaming tape and stitching. The advantage of this method is that it is relatively simpler to perform as it just requires a calendaring machine, heat seal seaming tape and an industrial scale stitching machine. The disadvantage would be that since the calendaring and stitching process is conducted before the sealing of the seams process, it is harder to securely seal the seams. The second method would still use the process of stitching and seaming tape, but would replace the calendaring process by utilizing a large autoclave-like lab, which allows the outer environment to be in vacuum while the inflatable structure is inflated with pressure. The advantage of this method is that since the stitching process comes before the autoclaving process, the seams can be sealed more securely by using sealant pastes on top of the heat seal seaming tapes. However, the disadvantage of this method is that it requires a special lab, which could create a vacuum environment around the habitat, while the habitat is being pressurized. If such a lab environment is not available, one could over pressurize the internal structure of the inflatable while vacuumizing the gap between the layers [26]. The method we have decided to adapt to was the one which uses the process of calendaring as it doesn't require such a special facility and seaming tape itself could be used to seal the seams of the inflatable structure.

The first step involves applying the seaming tape on the future seams of the bladder layers, on both sides of each bladder layer. The word future is used as the bladder layers have not yet been stitched. As unintuitive as it may sound, the seaming tape has to be applied before stitching the layers as there is no other way to reach the bladder layers in between other layers after the entire inflatable structure has been calendared.

To make sure every layer works together as a single unit when inflating, a manufacturing process called calendaring is performed. It is the process of taking multiple layers of fabric and forming it into one single thicker layer. Calendaring is necessary to be performed as making the entire inflatable structure act as a single unit can be very difficult to do by pure stitching. Calendaring also makes sure no air is trapped in between layers which is beneficial as having air in between layers would later cause the layers to separate when the air expands in space [27]. To get a basic idea of what the process of calendaring may look like, one can refer to Figure 7.12. The different colours indicate the different layers of fabric and the thick green layer represents the calendared fabric. Different layers are fed through the heated rolls which the prints out a continuous sheet with a desired thickness [90].

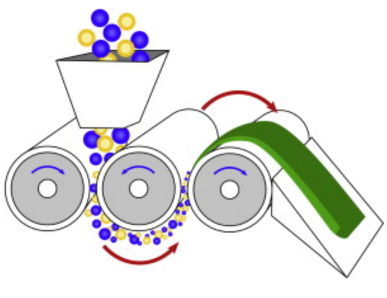


Figure 7.12: Calendaring Process [91]

Stitching is the common fabrication process used on fabrics. Stitching is necessary to be performed as it helps shape the inflatable structure in the cylindrical shape we'd want it to. The layers will be stitched with a high tenacity thread [24]. Although it sounds unintuitive to stitch the bladder layers which keeps the habitat airtight, it is a necessary process as there is no other way to shape the inflatable structure in the shape we would want it to be. Thus, further processing is necessary to seal the seam for the bladder layers.

As the seaming tape has been applied on top of the seams before it has been stitched, to seal the seams for

[26] Citation: Private communication with Dr Otto Bergsma, Faculty of Aeropsace Engineering, TU Delft, The Netherlands
[27] Citation: Private communication with Dr Otto Bergsma, Faculty of Aeropsace Engineering, TU Delft, The Netherlands

the bladder layer, heat seal seam tapes are applied, using a hot air taping machine[28]. The heat is applied locally, on top of the seam where the tape was applied. This would then seal the bladder layers again[29]. After this process, each layer will have the desired cylindrical shape and the bladder layers would be airtight again.

7.8.2. Rolling of the Inflatable Structure

The entire habitat is composed of a rigid airlock, a pre-assembled section of the inflatable structure, and a rolled up inflatable section as seen in Figure 7.11. The pre-assembled section would be in vacuum, to minimize the volume that would need to be taken into the launcher. The rolled up inflatable structure would need to be rolled up in a compact form, to minimize the volume intake in the launcher but also in such a way that when the inflatable were to be unrolled, it would efficiently be carried out. Two folding methods were considered as possible solutions for the folding of the unpressurized inflatable structure. The first method considered was the ”Z-fold” method, similar mechanism as an accordion, while the second method considered was the rolling method.The problem with the first method is that it would extend in a horizontal direction, skidding the inflatable structure across the lunar surface, possibly damaging the structure. Also, certain sections of the ”Z-fold” would need to be rigid, which would not be an ideal structural characteristic for this project. Final disadvantage of the Z-fold method would be that it would create sharp corners when it’s been folded, which could be detrimental to the inflatable structure as it may exceed the failure strain. The second method would involve rolling the inflatable structure in a somewhat cylindrical form as seen on Figure 7.13. The orientation of the rolled up inflatable structure in Figure 7.13 seems like it has been rolled vertically, but the actual inflatable structure is rolled horizontally. The second method would be the choice of preference, as the first method has too many disadvantages.

To be able to roll the inflatable structure in a smooth cylindrical shape, a cylindrical rod would be placed horizontally on top of the end of the inflatable structure. With the cylindrical rod on top of the inflatable structure, it will help guide the rolling process. To have some grip between the inflatable structure and the cylindrical rod, a strip of Velcro would be placed horizontally along the cylindrical rod. Then, the inflatable structure will be rolled up, just until the pre-assembled section of the inflatable structure. In order to hold the cylindrical shape of the rolled up inflatable and allow for a predictable unfolding mechanism, a passive control system will be implemented. The cylindrical rod, which is in the middle of the rolled up inflatable, will have cables coming out from both ends, extending until a device on the airlock. On top of the rigid airlock, this device can remotely extend or retract the cables. Once the cable extends to its maximum length and a certain horizontal force is exerted on the device, the cable comes loose from the device. Further explanation of the extension mechanism will be explained in Section 8.2.1. Figure 8.3 shows the inflatable structure when it is unrolled and the cable extension mechanism.

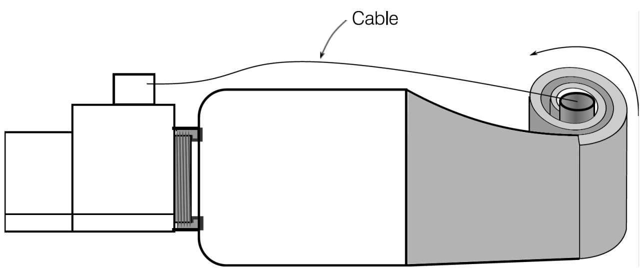


Figure 7.13: Rolled Up Inflatable Structure

Rolling of the inflatable structure has to be performed in such a way that the strain imposed on the inflatable structure doesn’t go over the failure strain of one of the layers within the inflatable structure. The layer with the lowest failure strain is the restraint layer, composed of Dyneema SK99. It has a failure strain of around 4.4% [30]. To ensure that the strain introduced from rolling the inflatable structure doesn’t go over the limit, the limit strain that has been set was 4%. To calculate the inner diameter of the inflatable structure, Equation 7.11 can be used. Since the limit strain was set to 4%, the value can be set to 1.04. The numerator in the equation is the strain imposed circumference while the denominator in the equation is the original inner circumference of the rolled up inflatable structure. The radius value of the numerator has an additional 0.056 meters added as the thickness of the inflatable structure is 0.028 meters and when rolling the inflatable structure, the thickness to consider is doubled as can be seen from Figure 7.14. After conducting the equation, the diameter of the innermost circle can be calculated to be 2.8 meters as seen from Figure 7.14. With the help of an online spiral

[28] URL: https://www.can-dotape.com/product-category/products/heat-seal-tapes/seam-tapes/
[29] Citation: Private communication with Dr Sybrand van der Zwaag, Faculty of Aeropsace Engineering, TU Delft, The Netherlands
[30] URL: https://www.teijinaramid.com/wp-content/uploads/2016/07/Product-Brochure-Twaron.pdf

calculator, the number of turnings is around 2.28, while the outer diameter is around 3 meters [31].

$$\frac{2 \cdot \pi \cdot (r + 0.056)}{2 \cdot \pi \cdot r} = 1.04r = 1.4 \tag{7.11}$$

Once the habitat has been rolled up, the entire habitat's dimension can be measured by adding the dimension of the airlock mentioned in Figure 7.10 as well as the dimension of the pre-assembled inflatable section which is mentioned in Appendix C. This can be seen in Figure 7.15.

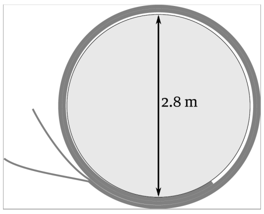


Figure 7.14: Dimension of Rolled Up Inflatable Structure

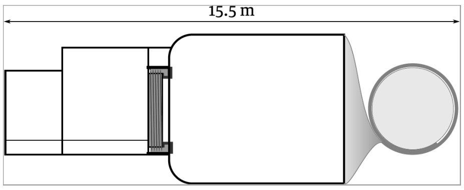


Figure 7.15: Entire Habitat Dimension

7.9. Compliance Matrix

The compliance of requirements are listed in Table 7.12, which were listed in Section 7.1.

Table 7.12: Compliance matrix for the structures subsystem

Requirement Code	Requirement Text	Compliance	Method	Justification	Source
ST-SYS01-07.1	The habitat walls shall have radiation shielding of at least 1 kg m^{-2}	Yes	Analysis	Regolith bags are great at stopping radiation	Chapter 4 [82]
ST-SYS01-08.1	The habitat shall have a total minimum thickness of 40 cm	Yes	Analysis	One meter of regolith on top of impact resistant insulation and restraint layers	Chapter 4
ST-SYS01-10	The habitat shall be regolith resistant	Yes	Inspection	Regolith is bagged, multi-layer structure is compact and non-permeable, dual-chambered airlock	Sections 7.4-7.7
ST-SYS02-04.2	The habitat structure shall not fail due to fatigue loading induced by 10 years worth of day-night cycles	Yes	Inspection	Load-bearing layer is extremely fatigue resistant	[99]
ST-SYS02-04.3	The habitat shall be able to resist meteoroid impacts	Yes	Inspection	Impact resistant materials and one meter regolith layer	Section 4.2
ST-SYS02-04.4	The habitat shall be able to withstand a temperature range of -70°C to 40°C	Yes	Inspection	All subsystems and materials employed operate within the temperature range	Section 7.3
ST-SYS02-04.5	The habitat shall be able to withstand the largest moonquake intensity so far observed	Yes	Inspection	Inflatable structures are inherently great at damping vibrations	[46]
ST-SYS04-01.1	The entrance and exit shall have a minimum size of 2m^2	Yes	Inspection	The airlock dimensions show that it's at least 2 m^2	Section 7.7.2
ST-SYS04-03	The habitat shall be constructed in such a way that expansion will not cause catastrophic failure of a system	Yes	Analysis	Airlock allows for connections needed for expansions	Section 7.7.5
ST-SYS08-01.1, ST-SYS08-01.2, ST-SYS08-01.3	The habitat shall contain entrance and exit points Those points shall stop regolith from entering the habitat Those points shall contain an airlock	Yes	Inspection	Airlock designed with safety and regolith cleaning as top priority	Section 7.7
ENV-SYS17-01.3	Materials used from the earth shall be kept to a minimum	Yes	Inspection	The regolith layer is by far the heaviest part of the structure, and only requires 0.17% in Earth materials to be stable	
MLO-SYS18-02	Available materials at the sites shall be documented	Yes	Inspection	Regolith properties are detailed and played a major role in the design concept selection phase	Section 7.6.2
ENV-SYS19-02	The sublimation of materials of the habitat shall be measured and documented on a two week basis	Yes	Inspection	Magnesium in the safehouse is the most likely to suffer from sublimation, and the maintenance plan accounts for its inspection	[82] Section 16.3

[31] URL: https://planetcalc.com/9063/

8. Manufacturing and Logistics

This chapter will introduce the manufacturing, assembly, and logistics plan for the habitat on the Moon. It will go over the different steps that will need to be taken on the Moon. Additionally, a schedule will be provided in the form of a Gantt chart as well as the production plan.

8.1. Lunar Cargo Logistics

A major challenge that needs to be addressed is regarding the lunar logistics. All the cargo will need to be removed from the lander and transported to the correct location. As described in Chapter 13, the Starship will be used to travel to the lunar surface. Following a press conference by SpaceX the Starships lander will have an elevator inside the payload bay to easily lower the payload to the surface [1] a render of what this will to look like can be found in Figure 8.1.

To transport cargo to different locations a robot called (Tri-)ATHLETE (All-Terrain, Hex-Limbed, Extra-Terrestrial Explorer) developed by NASA will be used. This transporter consists of 2 vehicles of 3 wheels each that can linked in order to lift and transport a cargo plate. The vehicle can transport payloads up to 14500 [kg] and is able to transverse over obstacles up to heights of 6.5 [m] due to its ability to us the wheel bases as legs by locking the wheels and moving the limbs independently [2] an image of the transporter can be found in Figure 8.2. An additional advantage of the transporter is that it can also have attachments on its 'legs' such as a small grapple to lift cargo. The movement abilities of these legs are however not precise enough for certain delicate actions such as the connection of cables. For these purposes it is recommended to install a robot arm on the transporter. A recommended arm for this purpose would be the COLDArm [3] in development for NASA's Artemis mission, by MOTIV Space Systems who also designed the robot arm on the Perseverance rover [4]. The arm is designed to be operable even in temperatures of -180°C.

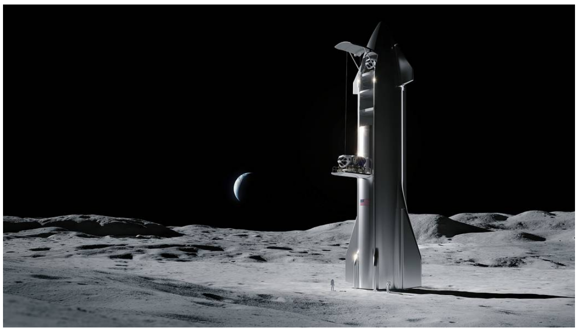


Figure 8.1: Concept of the Starship cargo bay [5]

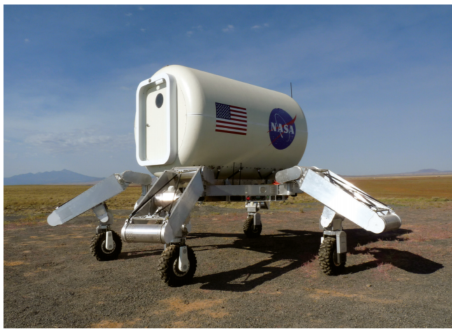


Figure 8.2: Image showing the ATHLETE cargo transporter [6]

8.2. Manufacturing and Assembly on the Moon

In section 7.8, the manufacturing and the assembly process which occurs on Earth has been discussed. Although the manufacturing process of the inflatable structure and the integration of it to the airlock is of paramount importance, there is more complexity in the manufacturing and assembly process which occurs on the Moon. Once the habitat has been placed at the habitat location, it has to be assembled in such a way that it doesn't damage the habitat and it's time efficient. The habitat has to roll out in a straight parallel line just like the trench where the habitat will be placed on top of. The following texts will discuss in depth of the assembly process of the habitat, how it will be unfolded, inflated, and anchored to the ground.

8.2.1. Assembly of the Inflatable Structure

The assembly of the habitat can be divided into three phases: I. Installation of the habitat; II. Initial deployment; III. Anchoring of the inflatable structure; IV. Inflation and pressurization.

The first phase involves placing the habitat in the exact planned location, so no deviation occurs during the

[1] https://www.youtube.com/watch?v=9SZ3mVGBiil&t=1989s
[2] https://www.nasa.gov/pdf/390539main_Athlete%20Fact%20Sheet.pdf
[3] https://motivss.com/tech-dev/cold-operable-lunar-deployable-arm
[4] https://motivss.com/coldarm-v-the-robotic-arm-of-the-mars-perseverance-rover

unfolding phase. This will be done by the transporter which will bring the habitat to the habitat location from the launcher as mentioned in subsection 8.1. Before bringing the habitat to the habitat location, the trench would have to be dug up by the excavator.

Phase 2 is the unrolling phase of the inflatable structure with passive control. The advantage of unrolling the inflatable structure with a remotely controlled extension device placed at the airlock is that it adds stability and control to the operation. However, the disadvantage of this method would be that it adds additional complexity, mass, and cost to the operation. Once the habitat has been placed in parallel with the trench, the remotely controlled device, which controls the cable, will slowly unwind the cable, so that the inflatable structure can slowly unroll itself. The inflatable structure will be slowly laid on top of the trench, so no damage is induced during the unrolling phase. Once the inflatable structure is fully unrolled, the cylindrical rod which is placed horizontally at the end of the inflatable structure can then be removed with the help of the transporter pulling it away horizontally from the habitat. Figure 8.3 helps visualize how the inflatable structure will look like after it has been fully unrolled. Placing the rigid airlock at the precise location is important as deviating by just a few degrees could lead to the inflatable unrolling in an undesired direction.

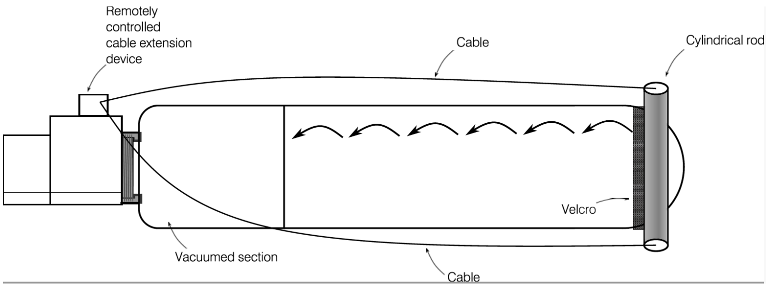


Figure 8.3: Unrolled Inflatable Structure

Once the inflatable structure is perfectly unrolled in the desired location, the inflatable structure has to be anchored to the ground so that later when the regolith bags are being stacked, no tip over is induced from an unbalanced torsional load. There are other reasons to why the habitat needs to be anchored. One other reason would be so that the trench does not have to be dug in a smooth semi-circular shape, which is complex to perform on the moon, but rather in a trapezoidal shape. Another reason would be that it removes the need for a second floor board for the storage area. If the bottom of the inflatable becomes flat, due to the anchoring cables bringing the inflatable structure down, a second floor wouldn't be necessary to cover for the circular shape at the bottom of the cross sectional area, in the case that the inflatable would not be anchored. This can be visualized on figure 8.4. The cables can be placed over the inflatable structure before being inflated by the cranes used to stack the regolith bags. However, to properly anchor the cables, a drilling robot or machinery would be necessary. Future technological development would be necessary to deploy robots or machinery on the lunar surface to drill and anchor the cables. Multiple cables may be required, depending on the total amount of tension the cables have to withstand. There are multiple drilling methods for this, which include, helical anchoring, suction drilling, circular wedge anchoring, and claw anchoring [56].

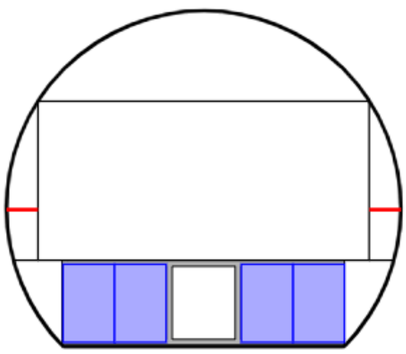


Figure 8.4: Cross Sectional Area for Inflatable Structure

Phase 4 is the pressurization step, which involves inflating the habitat and pressurizing it. To inflate the habitat properly, the nitrogen tank, the oxygen tank, and the vaporizer brought from Earth will be used. The nitrogen tank will be stored in the pre-assembled section of the habitat while the oxygen tank will be placed outside the habitat. The combination of the two elements will be turned into gas to pressurize the inflatable structure with the help of a vaporizer. The entire process to fully pressurize the habitat is expected to take no longer than one hour.

8.2.2. Internal Assembly of the Habitat

Figure 8.2 displays the athlete carrying a mock-up habitat during a test. The cargo that will need to be placed inside the habitat will be stored in a cargo bay that closely resembles this mock-up habitat. The cargo bay will be 3.5 $[m]$ in diameter and 8 $[m]$ long [42] and weights 1300 $[kg]$ [44]. The door of this transporter is able to directly connect to the airlock, thereby greatly reducing the risk of too much lunar dust entering the habitat, as presented earlier in Chapter 5. To move the cargo in and out of the cargo bay a sliding table mechanism will be used, this mechanism is also present in the Japanese airlock on the ISS to easily slide cargo through the airlock [7] an image of the airlock can be found in Figure 8.5.

From this sliding table the cargo will be further transported to the correct location by an AGV (automatic guided vehicle) that uses a pump truck like cargo bay to place the cargo boxes further inside the habitat such as the cargo storage underneath the habitat floor. As a reference robot the NIPPER was used, see Figure 8.6, a compact automatic pump truck. For the application in space this vehicle would require a few adaptions such as a higher slope tolerance and the size could be reduced even further.

Figure 8.5: KIBO airlock in the ISS [8]

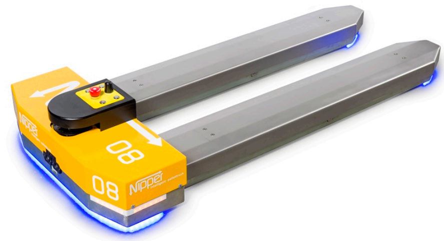


Figure 8.6: Image showing the NIPPER cargo transporter [9]

The cargo boxes will be stored underneath the floor of the habitat. Additionally several load bearing elements will be present in order to support the flooring from underneath. To get underneath the floor a small elevator is present in the rigid first section of the habitat. These elements will be added once the floor has been partially inflated in order to have enough clearance when moving them. A sketch showing how the cargo will move through the habitat from the transporter can be found in Figure 8.7. After all the cargo boxes and load bearing elements have bee placed the habitat floor can be fully inflated. The final assembly of furniture inside the habitat will be completed by the astronauts upon arrival inside the habitat.

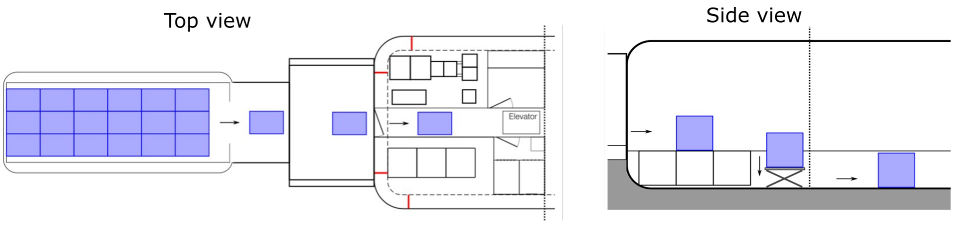


Figure 8.7: Sketch showing the transportation of cargo through the habitat

[7]https://iss.jaxa.jp/iss/kibo/al_e.html
[9]https://www.nasa.gov/mission_pages/station/research/news/SSSH_20aug18
[9]https://www.dinostretchhood.com/wp-content/uploads/2016/11/20171002_NipperSpecsheet_lr.pdf

8.2.3. Manufacturing of the Regolith Bags of the Habitat

To provide the habitat with sufficient thermal insulation and radiation protection about 1 [m] of regolith will be placed on top of the inflatable structure. There are several steps involved that will be required in order to complete this task.

Firstly the regolith must be gathered, then this should be bagged and brought to the habitat and placed in the correct location. Following from requirement SH12 this should be completed without the presence of astronauts. Therefore several robots are required to complete this task. To gather the regolith a robot developed by NASA, the RASSOR (Regolith Advanced Surface Systems Operations Robot), is recommended to be used. This robot was specially developed to be able to operate in low gravity environments and uses two counter rotating drums to gather regolith. It can then transport this regolith and dump it at the desired location by rotating the buckets in the other direction [77]. Each robot has a mass of only 66 [kg] and the ability to gather up to 2700 [kg] of regolith a day [78]. To complete the task of gathering the required amount of regolith in the allotted building time, 3 of these robots will be required.

The next challenge will be to fill the bags with the excavated regolith. Additional information on these bags can be found in Section 7.6.2. Most previously designed systems that were found used astronauts to complete (part of) the bagging procedure. Such as this proposal to use a pneumatic vacuum like robot to gather the regolith and letting the astonauts fill it with a hose [93]. The most suitable solution for the design of H.O.M.E. was determined to be a design proposed by NASA [12]. The regolith is first dropped into a hopper, to prevent excessive dust formation a cover is placed on top of the hopper to catch most of the dust created. The regolith is then transported using a conveyor belt system and placed into a metering funnel to ensure the bags are not overfilled. The automatic opening and closing is performed using a magnetic system. This design is able to fill about 60 bags of about 30 L an hour. The weight of the entire system would be about 112 [kg] and use 3.76 [kW] of power [12]. It is however important to note that this system was proposed in 1990 and it is highly recommended that additional research is performed to design a more compact and efficient bagging system. The design of what this would look like exactly was deemed to be beyond the scope of this DSE and this bagging system by NASA will be used as reference for the further planning of the H.O.M.E mission.

The final step will be to place the regolith filled bags on the habitat, this will be achieved using a crane. The crane chosen is the LSMS (Lightweight Surface Manipulation System), another NASA invention. The crane is able to lift up to 6907 [kg] while only weighing 169 [kg] [49]. It was decided to use 2 cranes to ensure that the bags can be placed symmetrically onto the structure, thereby preventing loading only one side of the structure and reducing the amount of time spent moving the crane around. The bags of regolith are brought to the cranes using the ATHLETE transporter described in Section 8.1. An overview of all the robots discussed in this chapter and their specifications can be found in Table 8.1.

Table 8.1: Overview of the required robots and their specifications

Robot Type	Name	Number Required	Mass [kg]	Power [kW]	Ability
Transporter (external)	ATHLETE	2	2340 [10]	2 [42]	Transport up to 14500 [kg][11]
Crane	LSMS	2	169 [49]	1 [12]	Lift 6907 [kg] [49]
Transporter (Internal)	NIPPER	1	200	0.8	Transport 1000 [kg] [13]
Excavator	RASSOR	3	66 [78]	10.8 [78]	Excavate 2700 [kg]/day [78]
Bagging system	SKITTER	1	710 [12]	5.7 [12]	Fill 54 bags a day [12]

8.3. Gantt Chart of the Manufacturing Process

To provide an overview of the schedule that will be adhered to during the manufacturing of the habitat on the moon a Gantt chart was created, which can be found in Figure 8.8. It provides an overview of the most important steps to be taken from the moment the first launch reaches the lunar surface. As can be seen on the chart the most time intensive task will be the excavation process. The allotted time presented in the chart follows from the time it would take 2 excavators to complete the task and a third excavator is brought for redundancy purposes. This is done to mitigate the risk [HA04] presented in Chapter 5. Both the transportation

[10] https://www.nasa.gov/pdf/390539main_Athlete%20Fact%20Sheet.pdf
[11] https://www.nasa.gov/pdf/390539main_Athlete%20Fact%20Sheet.pdf
[12] Estimated using potential energy and typical crane efficiencies found in [64]
[13] https://www.tbwb.nl/nipper_agv.html (for all in this row)

and placement of the regolith bags could also still be completed in the allotted time if a either a transporter or crane were to fail.

A minimum of 2 days was estimated for each operation presented to ensure the habitat will be completed in time for the astronauts. The time between the first landing and the arrival of the astronauts is about 13 months and the time between landing 2 and 3 is 6 months, thereby meeting both requirements SH14 and SH15 (see also Table 8.2), regarding the scheduling of the mission. The expected task duration was determined based on data found regarding the robots performance as well as outputs from other design processes such as the scaling of the solar farm, the regolith protection layer thickness, and the dimensions of the habitat. For several processes a precise value was not managed to be found, most importantly regarding the precise testing procedures required for the life support systems and potential leaks. However, it is expected that this will not pose a problem as sufficient time can be taken to complete these actions. Additionally, the unloading duration could not be pinned down exactly as there are still many unknowns regarding the abilities of the Starship at this point in time. This could potentially cause some delays in the construction program.

8.4. Robot communications and maintenance

In order to properly utilise all these previously introduced robots several considerations must be taken into account. Firstly, and most importantly they require power to function. Unlike previous robots sent to planets these do not contain a power generator in their design, but instead require external charging. Especially for the robots outside the habitat this will pose a challenge due to the interference with the lunar dust. Therefore using an ordinary plug and socket system will be extremely hard. A more likely solution is to use wireless charging technology to solve the problem. NASA and WiBotic, a wireless charging company, are working to bring this technology to the lunar environment during the Artemis mission [14]. Since the technology is still very novel further research will be required to determine the exact components this system will require, following contact with the company the team was provided with a very rough estimate of 200 [kg] for a charging station which will be used as a reference during the mission design.

A second challenge can be identified regarding the communication of the robots, both which each other and to the ground stations on Earth. There is an 2.6 second delay between communication going Earth > Moon > Earth, which is manageable but could still pose problems. It requires slow and careful movements of the machines to ensure accurate results. This constant communication also means that quite a significant amount of data needs to be communicated to and from Earth. The communication system required will be elaborated upon in Chapter 14.

The third challenge identified is regarding the maintenance required for the robots. During the manufacturing phase no astronauts are present to perform repairs. Therefore as previously discussed redundancy is introduced for most of the critical robots to ensure the habitat can still be constructed in case of failure. Additionally, spare parts will be brought to the habitat so that later maintenance can be performed by the astronauts.

8.5. Sustainability in Manufacturing and logistics

Sustainability in manufacturing and logistics is the ability to carry out the necessary current tasks without compromising the future manufacturing and logistics tasks. As the equipment and the machinery used to manufacture and assemble on the Moon run on solar energy, the entire manufacturing and assembly process can be seen as sustainable. Solar energy is unlimited to use and as long as the solar farm is functional. The regolith inside the regolith bags and the bags can be reused if necessary. Additionally, by completing a significant portion of the manufacturing process on the moon, a significant amount of launch mass is saved thereby greatly reducing the propellant needed during launch.

The biggest sustainability issue regarding the manufacturing on the moon follows from the large amount of lunar regolith that will be excavated thereby altering the lunar environment. Although this regolith is not processed further thereby it is possible to easily reuse the regolith filled bags or even place it back.

[14] https://www.techbriefs.com/component/content/article/tb/supplements/bt/features/articles/39148

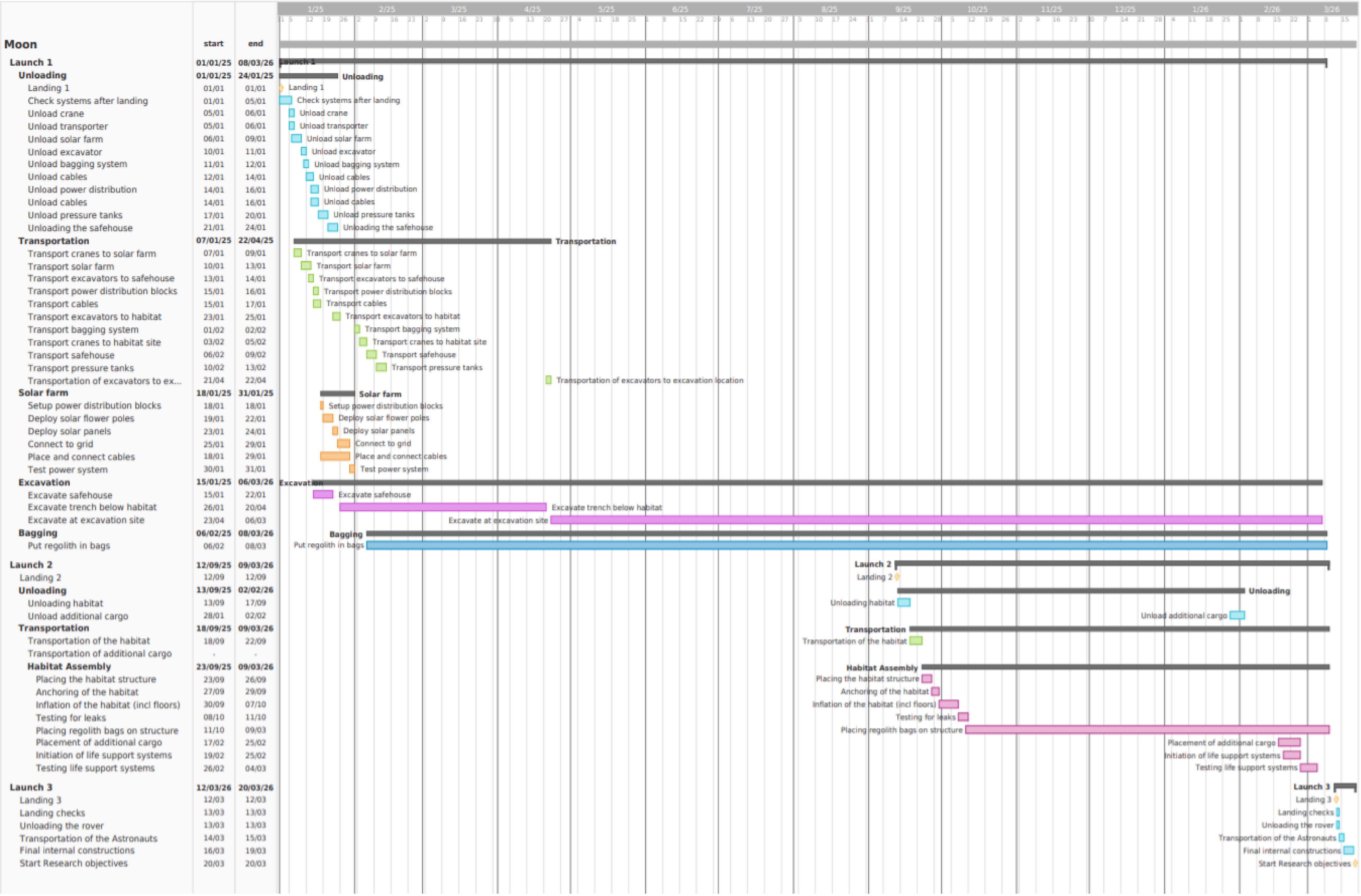


Figure 8.8: Gantt chart on the lunar manufacturing process

8.6. Compliance Matrix

Table 8.2: Compliance matrix for manufacturing

Requirement Code	Requirement Text	Compliance	Method	Justification	Source
MN-SYS12-01	"All initial material processing shall be performed without the need of human presence"	Yes	Demonstration	Self-Evident	section 8.2.3
MN-SYS12-02.1	" The maximum time required for a structure to be built shall be 24 months minus the time between the first arrival and the arrival of the necessary components"	Yes	Demonstration	Self-evident	Section 8.3
MN-SYS12-02.2	"An overview of the current state of production shall be provided to the ground segment "	Yes	Demonstration	Self-evident	Chapter 14 and Section 8.4
MN-SYS12-03.1	"The automated equipment shall be able to be operated from the ground"	Yes	Demonstration	Self-evident	Chapter 14 and Section 8.4
MN-SYS12-03.2	"The equipment shall be designed using a fail-safe approach, where the failure of a single piece of equipment shall be able to be compensated by the use of another piece"	Yes	Analysis	Additional robots brought for redundancy purposes as well as spare parts	Chapter 8 and Section 8.2.3

8.7. Production Plan

The production plan is an outline of the manufacturing, assembly and the logistics activities in a chronological order. Figure 8.9 shows the production plan of this project. As can be seen from figure 8.9, some white boxes are stacked on top of each other, in parallel, which means that those activities will be performed in parallel. The white box indicates an activity, while the red rhombus shape represents any testing, checking, verifying or validating phase. The rhombus shaped boxes are in a feedback loop, where if the validation result turns out to be negative (N), it loops back to the previous activity. If the result turns out positive, it proceeds to the next activity.

The production plan is initiated only when the design of the habitat is finalized. The first main task would be manufacturing the inflatable structure, which will compose the majority of the habitat. Once the habitat has been manufactured, the product has to be validated through testing and inspection. Once the validation process has been passed, the inflatable structure has to then be integrated with the airlock. This phase also has to be validated as it is important to have the entire habitat airtight. This validation process can be conducted through demonstration and inspection. Show that the entire habitat can be inflated at its optimal pressure and show no signs of pressure loss over a period of time. Once the habitat has been tested to be airtight and durable, the habitat then has to be deflated, so that part of the inflatable structure can be rolled up. The rolling and unrolling of the habitat will have a control device as described in subsection 7.8.2. This extension and retraction mechanism of the cable has to be demonstrated and inspected through the process of validation. Once this process has been validated, the entire habitat will be compactly positioned in the launcher. Once the launcher has reached the lunar surface, the payload will be unloaded and the transportation of the payload will begin. The transportation of the power system and cranes to the solar farm location would take place first, as setting up a power system is at its utmost importance. Once the solar farm has been set up, and the deployment has been checked, it will then be connected to a grid system.

The excavation of the trench for both the main habitat and the safe house then begins, with the help of an excavator. Once the trench has been dug, the habitat and the safe house will then be placed at the exact desired location by a transporter. Once the habitat has been placed at the desired location and the deployment has been checked, the excavation of the lunar dust to fill the regolith bags will begin. The filling of the bag process will continue for quite some time as there are a lot of bags to fill. Before the inflatable structure becomes pressurized, the inflatable structure has to be anchored down to the ground with a certain length of cable going over the top of the inflatable. Once the anchoring system has been deployed, it then has to be checked to see if it has been properly deployed.

Then comes the part of inflating the inflatable structure. The inflation will occur with pressurized gas canisters brought from Earth. Once the inflatable structure has been fully pressurized, the deployment will be checked, then the stacking of the regolith bags will begin. The crane will be used to stack the regolith bags and it would be desired to stack the bags as symmetrically as possible to prevent from exerting overload on the structure. After the construction of the habitat has been completed and the habitat and its related systems have been checked, the astronauts can then be sent to the moon. Once the astronauts land on the moon, they will be transported to the habitat location and they can then begin their internal construction in the habitat.

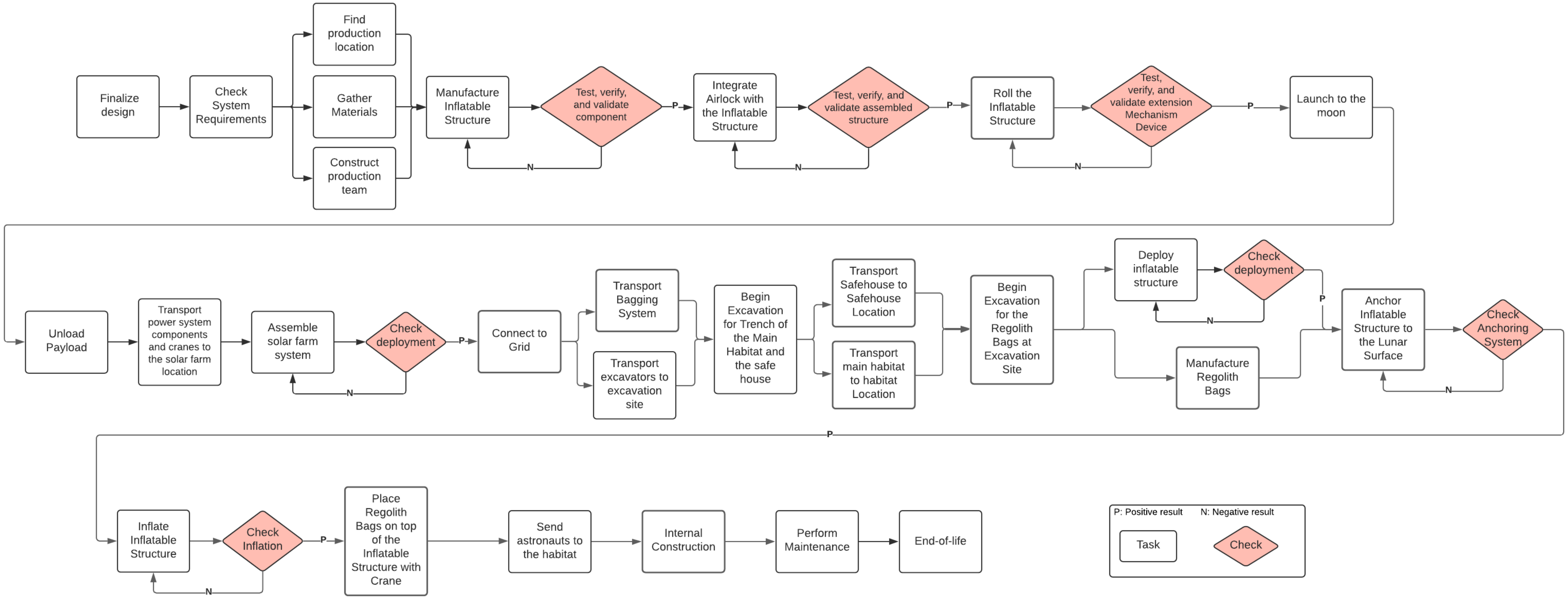


Figure 8.9: Production Plan

9. Power Generation, storage and distribution

In this chapter, the power generation, power storage and power distribution will be covered for the habitat. The phase from the first launch until the end of habitat construction is referred to as the construction phase. While the phase from when the astronauts move into the habitat on wards is called the operational phase. Power generation on other celestial bodies like the Moon proves to be quite a challenge and requires careful planning and design. The power system has been optimised with regards to launch mass and towards sustainability. Being able to power the lunar base during long periods of darkness was also a driving factor. The power system of this particular design was composed of two major components: the solar generation facility that powers the habitat and its construction during periods of sunlight; and a regenerative fuel cell (RFC) system running on hydrogen and oxygen, powering the habitat during periods without sunlight as well as the first stages of the habitat manufacturing period (which will be discussed further in Section 9.2). First however, the general overview of the power system will be given in Section 9.1, after which the logistics of the power system and the manufacturing will be touched upon in Section 9.2. Finally the solar power generation system and the hydrogen fuel cells systems will be further discussed in more detail in Section 9.3 and Section 9.4 to Section 9.6 respectively. The chapter will finish with the sustainability analysis of the power system as well as the compliance matrix of the power system design of the mission.

9.1. General Overview

The main power generation system of the habitat will be solar power generation system using photovoltaic cells. This system will power the habitat during its nominal mission life as well as the manufacturing of the habitat. The main advantage of solar power is that its source (the Sun) is an almost inexhaustible resource, but a major drawback is that is does not provide power in the absence of sunlight. For this reason, a second power generation and energy storage system was added to the design, which comprises of a regenerative fuel cell system using hydrogen and oxygen, that will provide the habitat with power in periods of darkness, as well as the first stages of the habitat manufacturing. Namely, in the first launch, the solar farm will be brought to the Lunar surface (see also Section 13.4). Before the farm can provide power to the manufacturing process and the habitat, it has to be set up first which requires power on itself. In this stage the RFC will also provide power to the mission until enough of the solar farm has been set up to make the solar farm erection process self sustainable.

A final note on the RFC system is that it will be a closed loop system. During times of darkness and the first stages of the mission, the fuel cells system will convert hydrogen and oxygen, into water and energy to provide power to the mission. At times of sunlight, the solar farm will provide power to the electrolysis system which converts said water back into hydrogen and oxygen to produce again an energy buffer for the habitat for periods of darkness. The overall closed loop system can be seen in Figure 9.1.

9.2. Power demand and logistics

9.2.1. Power demand

The mission power demand changes trough out the mission. In the table below one can see the power profile for the mission.

Table 9.1: Mission phases and corresponding power profile

	Phase	Powered by	Power Required	Time (for RFC only)
1	*Solar field construction*	RFC	6kW	6 days
2	*Habitat construction*	Solar field	55.3 kW	∞
3	*Habitat power (lunar day)*	Solar field	35.2kW	∞
4	*Habitat power (lunar night)*	RFC	6.3kW	18 days

- In the first phase the solar field needs to be constructed. The power for this will be delivered by the RFC system. Transporters and cranes will be powered by electricity generated from this system. A total capacity of 6kW for 6 days is delivered by the system.
- After that the habitat is going to be constructed. Mainly the regolith excavation and bagging uses a lot of energy. This will all be provided by the solar panels. A total of 75kW will be made available as long

as the construction lasts.

- After the astronauts move into the habitat, the lunar base has a power draw of 35.3kW for normal operations. This will all be provided by the solar panels.
- During lunar night solar power will not be available. During these night the RFC system will be used. The base uses less power during the night due to reduced activity and due to absence of electrolysis power draw. The habitat only uses 6.3kW during this time.

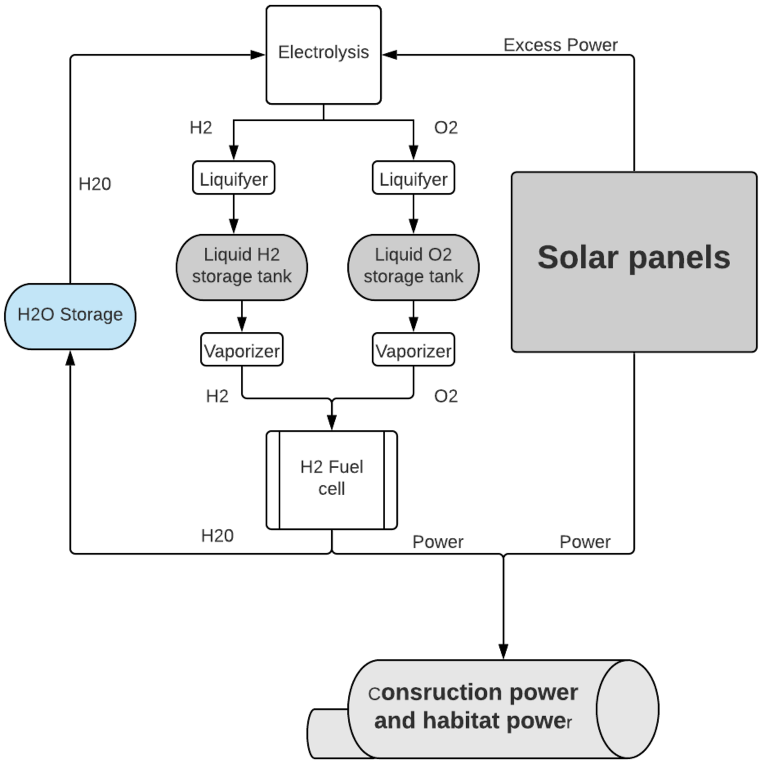


Figure 9.1: Overview of regenerative fuel cell system

A complete overview of the power draw of the habitat can be seen below. Four power scenarios are presented. Peak power during day and night. With all systems turned on and on their maximum load. This in reality will never be necessary. Therefore, the average load is also presented. The solar cells are designed for to deliver enough power for the construction phase. This requires more power then the peak habitat load during lunar day time. Therefore the day power demands are fully covered by the solar panels. During night time the RFC system is designed to deliver the average power needed in the habitat. Strategically not using using systems that are not essential for habitat operations (washing machines, dishwasher etc) considerably saves power and therefore system mass. A safety factor of 1.1 is used in case power loads turn out higher then expected.

The RFC system will be designed such that it provides enough energy to let the habitat survive on average load for 18 days with a continuous power draw of 6.3kW (without solar power). If the crew switches of more systems, this period can be stretched. The 18 days follow from the day night cycle on the lunar south pole. Extremely long days of more then 200 days are not uncommon. The longest night is 4.5 days. However multiple ones of these are followed up by each other with relatively short day periods in between.[35] The electrolysis device recharges the RFC with a rate of 1 day of RFC power (6.3kW) per 1 day of solar power. From this it has been calculated that 13.5 days worth of energy needs to be stored in the RFC system. 5 extra days are accounted for in case of system failure with the electrolysis device. The Crew then has an additional lunar night of 4.5 days to solve the issue without losing power.

The lunar night is the most energy intense load on the RFC. Therefore it has been sized to this amount of hydrogen and oxygen. Based on the load of 6.3kW for 18 days, the total energy has been computed. This includes a 10% safety factor to account for higher power loads then expected. The fuel cell efficiency of 55% [1] been taken into account. Also a safety factor of 1.25 has been applied to account for lower efficiency fuel

[1] https://www.californiahydrogen.org/wp-content/uploads/files/doe_fuelcell_factsheet.pdfhas

cell reaction. Conditions on the lunar surface can fluctuate which affects the fuel cell performance. From all this a total energy of 22267 MJ is required from the RFC system. divided this number by the specific density of liquid hydrogen which is 120 MJ/kg, gives a hydrogen mass of 185.56 kg and oxygen mass of 1484.51. Additionally, hydrogen is needed for the carbon dioxide removal assembly (CDRA).

Table 9.2: Subsystem power usage and totals for different scenarios

Subsystems	Peak power day [W]	Peak power night [W]	Mean power day [W]	Mean power night [W]
WPA	800	800	300	300
Water quality	80	80	30	30
UPA	600	600	91	91
Toilet	455	455	72	72
Atmosphere Control	219	219	200	200
Airlock Pump	1000	0	83	0
MMAC (Cooling)	1000	500	480	240
OGS	3573	3573	1470	1470
CDRA/Sabatier	500	500	500	500
Dishwasher	1150	0	96	0
Washer/dryer	750	0	20	0
Kitchen appliances	540	540	270	270
Control system	300	300	300	300
sensors	10	10	10	10
Control room	336	336	336	336
Window screens	168	168	168	168
Communication	2000	1000	500	500
Projector	300	300	150	150
laptops	160	160	160	160
Wireless router	10	10	10	10
Illumination	300	300	300	300
Electrolysis	18055	0	18055	0
Fuel cell	0	0	0	0
Liquifyer	5200	0	5200	0
Total	37509	8783	29150	5179
Scientific experiments	41260	9661	32065	5697
Total +10% safety margin	**45400**	**10600**	**35300**	**6300**

Table 9.3: Summation of the hydrogen and oxygen needed to bring.

	Liquid H2 [kg]	*Liquid O2 [kg]*	*Notes*
CDRA/Sabatier	267.47	0	Sabatier system uses hydrogen for carbon dioxide removal
Life support oxygen	0.00	495.00	Oxygen that needs to be pumped into the habitat after vaporizing
Lunar night propellant	185.56	1472.87	Based on a power need of 6.3kW for 18 days. Including fuel cell efficiency of 55% and a safety factor of 1.25.
Total	**453.04**	**1967.87**	

9.2.2. Liquefaction and boil-off losses

As mentioned before the mission will make use of a RFC system. This requires a hydrogen fuel storage facility. Due to volume constrains in the launcher, cryogenic storage will be used. Cryogenic storage comes with two major drawbacks however. Energy needs to be put into making gas hydrogen into liquid hydrogen. This can cost as much as 12kWh per kilogram hydrogen. This amounts to 36% extra power input. This can be provided with solar power which is available in abundance. Another option would be the use of a pressurised hydrogen system instead of liquid hydrogen. This mitigates the need for liquefaction and therefore the energy inefficiency. However a study carried out by NASA[61] pointed out that such a RFC gas system would result in a higher overall power system mass compared to a liquid RFC system[2]. The second drawback is boil-off. No liquid hydrogen tank is capable of completely sealing of its content to energy influx from the outside world.

[2]URL: https://www.sintef.no/globalassets/project/hyper/presentations-day-2/day2_1105_decker_liquid-hydrogen-distribution-technology_linde.pdf

High standard multi layer insulation cryogenic storage tank have a boil of rate of in between 0.05-0.01% [3] per day. This gas needs to be vented out of the tank otherwise the pressure will build up and rupture the tank. During the construction phase boil of hydrogen and oxygen gas will be directly fed to the fuel cell to react into water. During the operational phase the boil of hydrogen will be fed to the Sabatier carbon dioxide reduction assembly (CDRA). Boil of oxygen gas will be fed to the habitats oxygen tank for life support. The target is to not vent any hydrogen or oxygen in space. With this strategy hydrogen and oxygen boil of losses will be minimized for the mission and subsequently launch mass will be saved.

9.2.3. Power logistics

During the construction phase the habitat is not yet available for placing the electrolysis device and the fuel cell. They can not just be left on the lunar surface. This is namely due to the temperature but also due to the regolith dust. A 3 cubic meter "power box" will house two fuel cells, an electrolysis device, a water storage tank, a liquifyer, a vaporiser and a power distribution system. The fuel cell produces heat which is used inside the assembly to keep the subsystem components on operating temperatures and to prevent the water from freezing. This power assembly can easily be moved around due to its compact size by the transporters. It will be placed close to the habitat immediately at the start of the mission together with the liquid hydrogen and liquid oxygen tank. A power distribution and charging block for the lunar robots will also be placed in the center of the soon to be constructed solar field. In between these two blocks a long power cable will be laid.

During the operational phase, all energy generation and regeneration systems are located in the habitat. They have already been build into the habitat on earth. In the habitat the ambient conditions are suitable for all power sub system's. The habitat will have two fuel cells, two electrolysis devices, two liquifers a water storage tank and two vaporisers. The only thing left outside on the lunar surface are the storage tanks. The vacuum of space is actually beneficial for insulation. The storage tanks will be surrounded by four sun walls to prevent direct solar radiation hitting the tanks. Boil-off will be reduced with these walls. They will also act as protection against meteorite impact.

9.3. Solar Power

The solar power system will power the habitat for most of its mission and will be sized based on the largest power draw as seen in Section 9.2. As a general picture, the solar generation system will consist of a grid of so called solar towers, each containing a set amount of solar panels and photovoltaic cells. These towers will all provide power to a single power accumulation point in this grid, after which the power will be bundled together and sent to the habitat. The base for this system will be the photovoltaic power cell, for which an Gallium Arsenide photovoltaic cell with high TLR has been chosen. The particular photovoltaic cell is manufactured by *AzurSpace* [4], a company that specializes in (space) solar power generation technology. The specifics of the photovoltaic cell can be found in Table 9.4 and will be used in the further design of the solar farm.

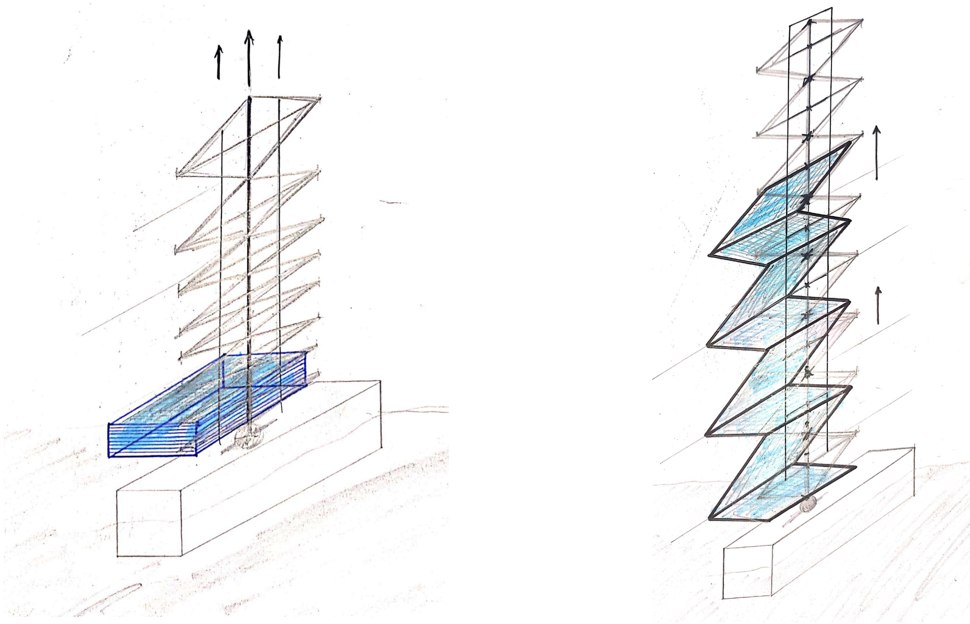
Table 9.4: Relevant parameters of the photovoltaic cell used in the design of the solar generation sytem.

Azurspace Triple Junction GaAs Solar Cell	
Base Material	GaInP/GaAs/Ge on Ge substrate
Dimensions	40.15 mm x 80.15 mm $\pm$ 0.1 mm
Cell Area	30.18 cm^2
Average Weight	118 mg / cm^2
Thickness	280 $\pm$ 25 μm
Average Open Circuit Voltage	2690 mV
Average Short Circuit Current	519.6 mA
Voltage at max. Power	2409 mV
Current at max. Power	502.9 mA
Average Efficiency	29.3 %
Operation Temperatures	-150 to 250 °C
Absorptivity α	0.91
Solar Constant on Moon I_{Luna}	1361 W/m^2

Using these photovoltaic cells and their properties as seen in Table 9.4, a conceptual design for a solar tower was developed. The solar tower will exist of an support structure made out of CFPR tubes assembled in a

[3]URL: https://www.utwente.nl/en/tnw/ems/research/ats/chmt/m13-hendrie-derking-cryoworld-chmt-2019.pdf
[4]URL: http://www.azurspace.com/index.php/en/

jack-like structure, that will be self deploying using the crane described in Chapter 8 and self locking joints that lock once the support structure has been fully extended. The support structure design has been inspired by the SunFlower design from *TransAstra Corporation*[5], a company that specializes in the areas of space resources and transportation. The tower furthermore consist of a solar panel assembly, consisting of 8 solar panels that are comprised mainly of the photovoltaic cells, an aluminium load bearing structure, a tempered glass front cover, a CFPR back cover and other additional layers that are encorporated in the safety factor for the solar tower mass as seen in Table 9.5. Furthermore the solar tower will have to be connected to the power grid using electric cables. A concept drawing of the solar tower assembly can be seen in Figure 9.3b.



(a) Deployment Mechanism of support structure of solar tower, inspired by the SunFlower design from *TransAstra Corporation*

(b) Conceptual unfolding mechanism of solar panels

Figure 9.2: Solar tower assembly unfolding mechanisms.

As already briefly mentioned, the entire solar tower and thus solar farm will be foldable for minimum storage in the launch vehicle. The solar tower will be transported in a *Solar Tower Assembly Box (STAB)* that contains the folded support structure, solar panels, and the required cable length to connect a particular solar panel to the power grid as seen in Figure 9.3a. During the first stages of the mission, these STAB's will be transported from the landing site to the solar farm site (see also Section 6.2), were they will be put in the correct grid according to Figure 9.3a. Then with the use of the crane robot, first the solar support structure will unfold after which the solar panels will be unfolded according to Figure 9.2a and Figure 9.2b respectively. The transporter robot will then transport the dedicated tower cable to the accumlation point of the tower grid which is positioned in the middle of set grid (Figure 9.3a). For detailed information on the robots used in the mission, the reader can refer to Chapter 8. For more information on the required power and energy for this first stage of the mission, the reader can refer back to Section 9.2.

9.3.1. (Grid) Sizing

With the general idea of the solar farm and the conceptual design of the solar tower and its deployment established, more precise numbers can be calculated based on the power requirement. From Table 9.1 is becomes evident that the manufacturing phase of the mission will put the most strain on the power generation system, and thus sizing will be done based on this mission phase. The solar towers will have a fixed dimension of 3.5 x 10 m for the support structure and 2 x 8 m for the solar panels. This means each tower will have 16 m^2 of solar panel area available. From this, the power generated by one single tower can be calculated using

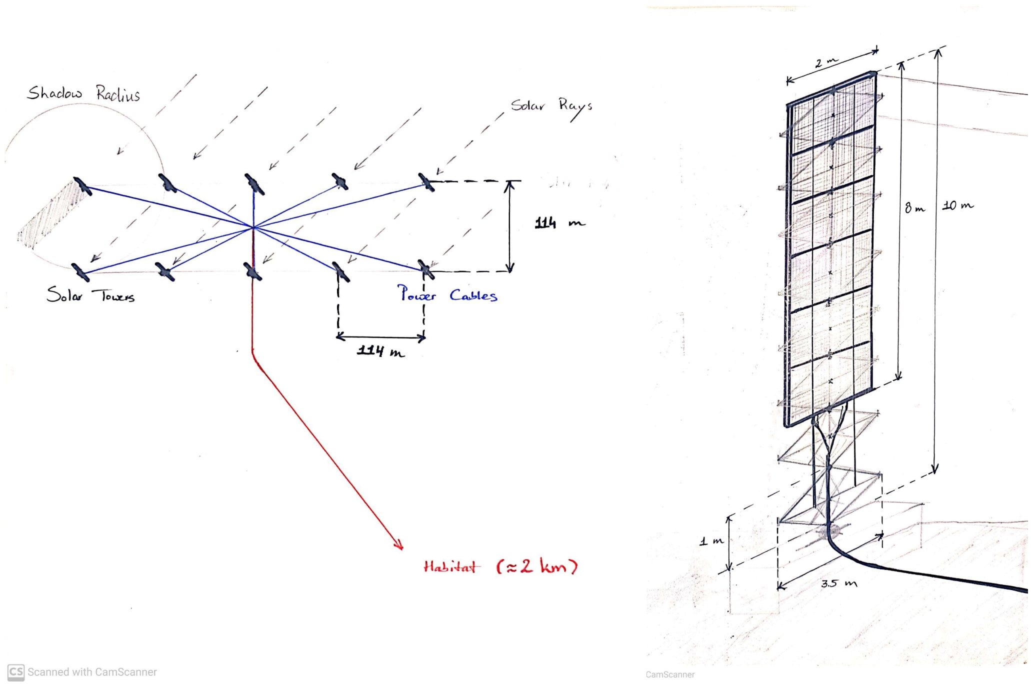
[5]URL: https://www.transastracorp.com/lunar-power-generation [cited on June 22, 2021]

Equation 9.1.

$$P_{tower} = I_{Luna} \cdot A_{panel} \cdot \eta_{solarcell} \cdot \alpha \cdot cos(\iota) \tag{9.1}$$

Where ι equals 5° due to the low inclination of sunlight at the Shackleton Crater as previously discussed in (Section 6.1). Filling in all values yields a power generation capability of 5784 W per tower. Dividing the total required power draw by this number, taking into account a 2% degradation of the solar panels per year for a total duration of 2 years (as this is the maximum duration manufacturing should take (**MN-SYS12-02.1)**, and after which in theory new solar panels can be brought), and rounding up the answer, results in a total of 10 solar towers needed to power the habitat during its most power intensive phases. An important note to make it that the solar panels will not be inclined with respect to the solar support structure to increase the power generation (as then the inclination of the sunlight can be perpendicular), as this produced negligible power increases with respect to the increase in launch mass it would produce due to additional structure. The solar towers themselves, however, will co-rotate with the Sun during the Lunar day to make sure the solar rays are always perpendicular on the solar panels as seen from above (see Figure 9.3a).

Although the low inclination of the solar rays is beneficial for the solar power generation, is does mean the solar panels need to be set up vertically, which brings the disadvantage that the solar towers produce long shadown due to the low solar inclination. To ensure the towers will not cast their shadow on eachother, and thus to increase the solar power generation efficiency, the solar towers are spaced in such a way that the shadow of one tower, will never touch the solar panels of another tower, no matter the orientation of the Sun. This *shadow cone* with a radius of 114 m is visualized in Figure 9.3a.



(a) Grid of the solar farm. The spacing of the towers was determined using the shadow lengths of the towers at the Shackleton Crater.

(b) Conceptual design of solar tower, the structure of which is completely foldable

Figure 9.3: Solar power generation system infrastructure

The last design aspects to mention regarding the solar towers regards the electrical design of the solar towers. Each tower will supply a total of 9.64 A and 600 V to the power grid, by connecting its solar cells in series and parallel accordingly. Each tower will be connected to the centre of the grid by a dedicated cable ranging in lengths from 235 m to 57 m depending on the position of the tower. In the power accumulation centre of the grid, the towers will be connected in series to increase the voltage to 6000 V to minimize transmission losses, after which the power is transported to the habitat approximately 500 m away (Section 6.2). The cables will have a core consisting of silver (lowest resistivity metal), which actually transports the power to the habitat, and will have additional protection layers similar to those of submarine power transmission cables to protect against the harsh Lunar environment [6]. All solar tower and solar farm design parameters are summarized in

[6]URL: https://www.emworks.com/blog/high-voltage/3-phase-high-voltage-submarine-power-cables [cited on June 22, 2021]

Table 9.5.

Table 9.5: Solar power generation system specifications

Support Structure			
Width	3.5 m	Rhombus Width	3.5 m
Height	10 m	Rhombus Height	1.0 m
Total Support Structure Mass			*81.4 kg*
Total Support Structure Volume			*0.039* m^3
Panel Assembly			
Width	2 m	Total Mumber of PV Cells	4776
Height	8 m	PV Cell Mass	17.0 kg
Support Structure Mass	38.15 kg	Support Structure Material	Aluminium
Front Cover Mass	249.8 kg	Front Cover Material	(Tempered) Glass
Back Cover Mass	153.7 kg	Back Cover Material	CFPR
Other Systems	45.9 kg		
Total Panel Assembly Mass			*504.5 kg*
Total Panel Assembly Volume			*0.21* m^3
Solar Tower Assembly			
Panel area	16 m2	Power	5784 W
Voltage Supply	600 V	Current Supply	9.64 A
Total Solar Tower Mass (incl. SF=1.5)			*878.8 kg*
Total Solar Tower Volume (incl. SF=1.5)			*0.37* m^3
Transmission Cables			
Total Length	3468 m	Core Material	Silver
Outer Diameter	25 mm	Core Radius	5 mm
Cable Density	1.4 kg/m	Total Cable Resistance	0.70 Ω
Total Cable Mass			*4855 kg*
Total Cable Volume			*1.70* m^3
Entire Solar Farm			
Spacing	114 m	Total Number of Towers	10
Total Solar Panel Area	160 m2	Box Dimensions	3.5x2x0.5 m
Transmission Loss	<100 W	Total Power Produced	57840 W
Total Voltage Supply	6000 V	Total Current Supply	9.64 A
Total Launch Volume			*35.93* m^3
Total Launch Mass			*13643 kg*

9.4. Fuel cell

A fuel cell is a device that converts hydrogen and oxygen into water and simultaneously convert a lot of the hydrogen's chemical energy into electrical power.Fuel cells have an efficiency of 55% if operated in their ideal operating conditions. An Automotive fuel cell design is chosen for the habitat. They have high power to weight ratio, good reliability, wide operating temperatures and are designed in a way with maintenance in mind. Ideal for a lunar base. The fuel cell used for the contruction phase and for the operarional phase is the Ballard FCgen HPS Fuel cell[7]. This fuel cell is able to output 140kW on full power if necessary. Two will be used during the constriction phase and two more will be used in the habitat during the operational phase. Two more will be stored as spare in the habitat in case of failures.

[7]URL: https://www.ballard.com/about-ballard/publication_library/product-specification-sheets/fcgen-hps-spec-sheet

Table 9.6: FCgen HPS Fuel cell by Ballard characteristics

Power max	140kW
Current	645 A
Voltage	202 V
Mass	55 kg
Volume	0.052 m3
Operating temp	-28C till +95C
Minimum storage temp	2C

Figure 9.4: FCgen HPS Fuel cell by Ballard

9.5. Electrolysis

To regenerate the hydrogen fuel to survive the lunar night, an electrolysis device is needed. This device splits water back into hydrogen and oxygen. Electrolysis devices have an efficiency of 80%[8]. This means that if you power the machine with 10kW it produces 8kW of hydrogen. The habitat has a dedicated electrolysis device used for high power loads. Two 10kW systems are available which can convert 16kW worth of hydrogen (due to the efficiency).The system is separated from the life support electrolysis with reliability in mind. The high load electrolysis device for the RFC system have not been used in space before. It is regarded to be more sensitive to failures. Spare electrolysis devices are taken in the cargo as a back up. Liquifiers are used to liquefy the gas into a liquid for storage.

9.6. Power storage

As mentioned before the storage method used for the RFC system is cryogenic. A separate liquid hydrogen and liquid oxygen tank is used. Structurally, the tanks do not need to be very resistant to stresses caused by the propellant because these are stored at atmospheric pressure. Some boil up may occur so the tanks must be resistant against some internal pressure. The tanks are made out of 2 cylindrical shells. An inner and outer layer is used to add an extra insulation. The inner layer will be a spherical aluminium pressure vessel and the outer shell will be a concentric aluminium shell with 90 layers of multi layer insulation. This multilayer insulation will be designed such that it can isolate the tanks content in the different environments acting upon it during launch, during orbit and boost, in space, during a landing, and most importantly during storage on the lunar surface. Being in the vacuum of space is a huge advantage as convection trough air is no longer a problem. On earth these tanks try to keep vacuum between the two layers. In space this is naturally easily achieved [61]. Below one can see the sizing of the propellant tanks:

a 5% percent tank volume was also added to accommodate for the maximum filling level possible. The two tanks will be completely shielded off from direct sunlight by deploy able walls which will be set up around the storage tank. These walls together with the outer layer of the fuel tank will also protect the fuel tank against micro meteoroid impacts

If some hydrogen is needed for power generation or the life support system it needs to be heated up and turned from liquid into gas. Vaporisers are used for this. they use the heat generated by the fuel cells to warm up the liquid hydrogen flowing towards the fuel cell itself.

[8]URL: https://www.carboncommentary.com/blog/2017/7/5/hydrogen-made-by-the-electrolysis-of-water-is-now-cost-competitive-and-gives-us-anothe

	unit	Liquid H2	Liquid O2
Propellant mass	kg	453.04	1967.87
Tank mass	**kg**	**750.54**	**234.08**
Propellant volume	m3	6.70	1.81
Inner tank radius	m	1.00	1.00
Outer tank radius	m	1.20	1.20
Tank height	m	2.13	0.58
Tank volume	**m3**	**9.65**	**2.61**

9.6.1. Accumulator

In the habitat a 923kg lithium ion battery will be present to prevent any power outage from happening in case of a power system failure. The battery would take up an approximate volume of 0.91m3 and is scaled with the VL51ES Li-ion space battery by Saft [9] as reference. This battery has already done multiple missions in micro gravity and is therefore a reliable option. Which is very important as it will act as the last power back up for the habitat. The accumulator will be able to deliver 5kW for 24 hours.

9.7. Power distribution

The RFC system will be installed next to the habitat. It will be connected to a charging block on which the lunar vehicles can be charged with direct current. A few hundred meters further in the solar field, a charging block is also present. Lunar vehicles and machines can directly charge in both areas. Both areas will be connected to each other so that during the construction phase power from either solar or the RFC system is available on both sites at all time. When the habitat construction is completed, both the PV arrays and the RFC system will be connected to the power distribution system of the habitat. Below the schematic power distribution of the mission can be seen. PV current goes to the charge controller. This control unit consists out of transformers as well as a regulation unit that decides which amounts of power goes where and what power source is used.

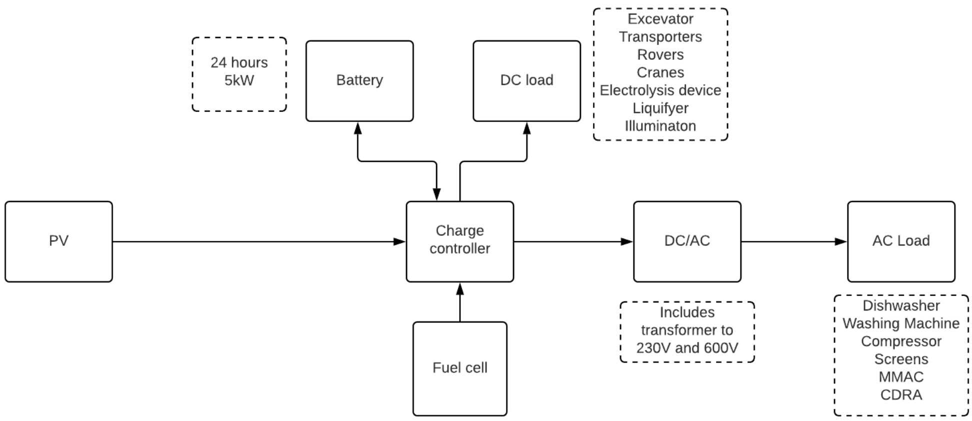


Figure 9.5: Power distribution overview

Below one can see the software diagram for the power system:

[9] https://www.saftbatteries.com/products-solutions/products/vl51es-battery

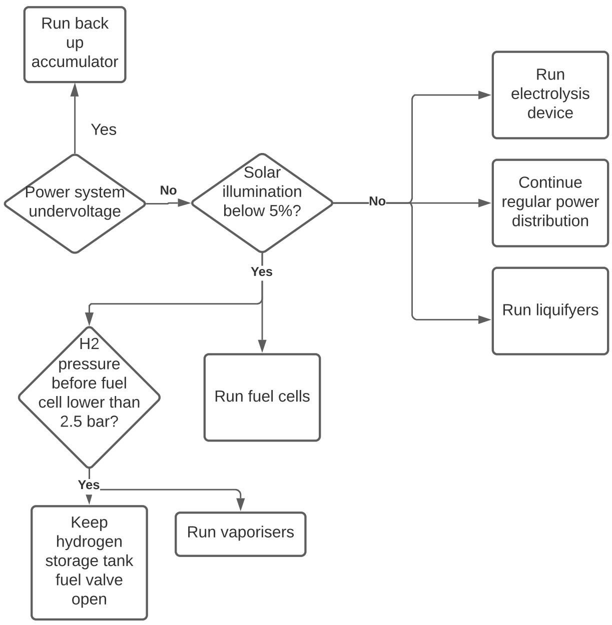


Figure 9.6: Software diagram of the power distribution system

9.8. Power system launch mass summary

Below one can see a summary table of all power subsystems and their corresponding launch mass. Also shown is the total power system launch mass which is 18440.60kg. All masses are taken from the previous sections. The liquifyer and vaporiser masses are taken from a NASA regenerative fuel cell case study[61].

Table 9.7: Launch mass sum of power subsystem components and total mass of power system

Subsystem	mass [kg]
Hydrogen mass	453
Oxygen mass	1968
Hydrogen tank	751
Oxygen tank	234
Water tank	47
Fuel cell	100
Electrolysis device	110
Vaporisers	26
Liquifiers	186
Accumulator	923
Solar panels	13643
Total launch mass	**18440**

9.9. Sustainability in Power Configuration

The largest part of the mission life, solar power will be used as the main power generation system, which is a sustainable resource. The solar power farm will power the habitat and manufacturing almost right from the beginning of the mission, meaning additional power systems can be scaled down meaning less mass is launched from Earth, which has positive environmental effects on Earth. Furthermore, the hydrogen and oxygen of the regenerative fuel cells will be generated on earth from sustainable sources. By making use of a RFC system instead of electrical chemical battery's, multiple tons of launch mass is saved. This will have a positive effect on the environment as the payload of the launcher can be used more efficiently. Moreover, by avoiding electrochemical batteries, hazardous resources such as lithium do not need to be mined. The RFC system infrastructure can also be used for future missions. The most heavy components such as the fuel tanks do not degrade. Also, the hydrogen and oxygen are used in a closed loop, which means no gasses are emitted into the Lunar environment.

9.10. Compliance Matrix

Table 9.8: Compliance matrix for the power subsystem

Requirement Code	Requirement Text	Compliance	Method	Justification	Source
PW-SYS21-01	The power System shall deliver a maximum of 300kW of power during Moon days	Yes	Analysis		
PW-SYS21-02	The power System shall deliver 6.3 kW of power during Moon nights	Yes	Analysis		
PW-SYS21-03	The power system shall be able to store 223 GJ of energy	Yes	Analysis	The hydrogen fuel storage tank houses 185.56kg	
PW-SYS21-04	The power supply shall be located outside of the habitat	No	Inspection	The fuel cells are located in the habitat	
PW-SYS21-05	The power supply shall cover a maximum solar panel area of 1000 m^2	Yes	Analysis	Solar panel area for the base is 160 m^2	
PW-SYS21-06	The power system shall have an emergency power system	Yes	Inspection	The habitat has a back up accumulator	
PW-SYS-21-07	Nuclear power in any form shall not be used	Yes	Inspection		
PW-SYS-21-08	The habitat shall provide 10% of its power to the scientific research sub part	Yes	Analysis		

10. Life Support

In this section, the required systems for life support are discussed. Radiation protection has already been covered in Chapter 4.

10.1. Requirements

As discussed in the Midterm report[39], the life support systems will be present in the pre-assembled part of the inflatable. The primary reason for this is the sheer complexity of connecting all the various subsystems and the required storage tanks. To minimise the amount of volume needed in the pre-assembled part of the inflatable, each subsystem size has to be determined. Sizing and selecting the subsystems will have to be done using the requirements established in the Baseline report [38], which are listed below.

SH01: Four astronauts shall be able to survive on the Moon

- **LS-SYS01-01**: The habitat shall provide breathable air
  - **LS-SYS01-01.1**: The habitat shall maintain an air mixture of at least 21% Oxygen during the entire mission
  - **LS-SYS01-01.2**: The habitat shall have an air storage tank
  - **LS-SYS01-01.3**: The habitat shall maintain air pressure of at least 30 kPa.
- **LS-SYS01-02**: The habitat shall provide water
  - **LS-SYS01-02.1**: The water system shall provide 16 litres of drinkable water per day
  - **LS-SYS01-02.2**: The water system shall provide 7 litres of water for hygiene
  - **LS-SYS01-02.3**: The water system shall provide 2 litres of water per day for scientific experiments
  - **LS-SYS01-02.4**:The habitat shall have water filtering systems for ”recycling” used water
  - **LS-SYS01-02.5**: The habitat shall have a spare water supply of 500 litres.
- **LS-SYS01-03**: The habitat shall provide edible food
  - **LS-SYS01-03.1**: The habitat shall provide 12000 calories of food per day (Reference taken from ISS 3000 calories per day)
  - **LS-SYS01-03.2**: The habitat shall provide a healthy balanced diet containing all macro- and micro-nutrients needed for optimal health

10.2. Sub-system Description

To determine all types of systems necessary to fulfill the requirements, similar life support applications were examined. Currently, the most comparable and usable life support system is that of the International Space Station. Not only is it similar in the amount of crew members it has to support, but also the extensive mission duration (10 years for Exodus). A basic overview of the major sub-systems within the Water Recovery and Management System of the ISS can be seen in Figure 10.1.

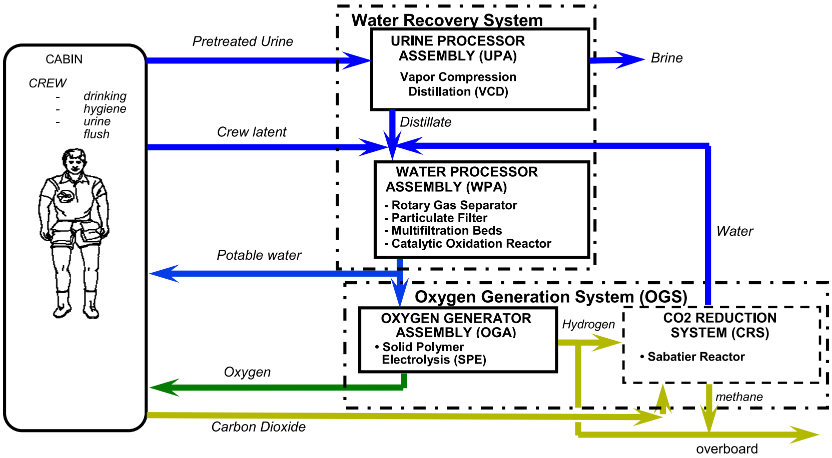


Figure 10.1: Diagram showing the interactions between the subsystems of the Water Recovery and Management System of the US ISS segment. [14]

Water Recovery System

The Water Recovery System (WRS) consists of the Urine Processor Assembly (UPA) which recovers the water in the crew members' urine, the Water Processor Assembly (WPA), which produces potable water for crew use, and distilled water for the Oxygen Generator Assembly. After providing a breathable atmosphere, potable water is the most essential life support subsystem. In terms of total mass, *"water regeneration streams are the most massive of the life support subsystems"* [2, p. 62]. The WRS in the ISS occupies two International Standard Payload Racks (ISPR), which can be seen in Figure 10.2.

Figure 10.2: The ISS Water Recovery System consisting of WRS 1 on the left and WRS 2 on the right. The Water Processor Assembly is contained in WRS 1 and the right half of WRS 2. The Urine Processor Assembly is in the remaining left half of WRS 2.[1]

The Water Processor Assembly can process 6.8 kg/h of combined wastewater and is therefore able to meet requirement LS-SYS01-02.1. The Urine Processor Assembly is able to process 2.0 kg/h of urine plus flush water and is therefore capable of supporting 4 crew members' waste water production [86, p. 5]. All final values for water will be presented in a later section to support this statement. The most significant figures required as input for the sizing of other systems is presented in the table below.

Table 10.1: Mass and power draw for the Water Recovery System

Subsystem	Mass (kg)	Peak Power (W)
WPA	476	800
UPA	128	600

Oxygen Generation

Generating oxygen is the most critical part of life support. A sudden failure would not mean the immediate loss of life or loss of mission, but could cause future missions to encounter more resistance. Not only due

[1]https://www.nasa.gov/mission_pages/station/research/benefits/water_purification.html

to the added cost of replacement and redesign, but the risk associated with it as well. The basic equation of electrolysis (oxygen generation) is:

$$2\,H_2O \longrightarrow 2\,H_2 + O_2 \qquad (10.1)$$

For an operated use of 10 years, closed loop systems are inevitable. Oxygen generation using water electrolysis is the most widely used and suitable closed loop method of generating oxygen, as demonstrated in the ISS' OGS. In addition, the abundance of lunar ice, allows for the creation of oxygen using electrolysis, which further supports the choice of using electrolysis. The OGS in the ISS takes up one International Standard Payload Rack, and weighs 113 kg [86]. It is designed to generate oxygen at a rate of 5.4 kg/d. If more or less oxygen is required the OGS can operate at a rate between 2.3 and 9.2 kg/d [2]

CO2 Reduction System

As experienced during the notorious Apollo 13 mission in 1970, high CO_2 levels within the atmosphere pose an extreme threat to crew life and therefore mission success. For Apollo 13, Lithium Hydroxide canisters were used to remove CO_2 from the atmosphere. Although they were proven to be quite effective at saving the mission, for longer missions trade-off studies and NASA recommendations show that closed loop systems weigh less [2]. Available for this is the Bosch system and the Sabatier system. Trade off study's have shown that a Sabatier system is more reliable and more safe to operate. Additionally, it is also more mass-efficient[51]. Indeed, in the ISS a closed loop Sabatier Reactor system to remove CO_2 from the atmosphere has been used or multiple years successfully. The principle workings of the Sabatier reactor can be seen in the equation below[54].

$$CO_2 + 4\,H_2 \longrightarrow CH_4 + 2\,H_2O \qquad (10.2)$$

Evidently, the use of hydrogen fuel cells turns out to have additional advantages. 55% of the hydrogen is lost to methane which will be vented into space. The water can be used to produce hydrogen using the OGA as seen in Figure 10.1.

The Sabatier system for the mission is sized to cycle all CO2 out of the atmosphere produced by four astronauts. Assumed is 1kg of CO2 produced per astronaut per day[2]. The results of the system sizing can be seen in the table below. The result is a system the size of a small refrigerator weighing 263.67kg[3].

Table 10.2: Sabatier system characteristics

Sabatier CDRA system	
Mass	263.67
Volume	0.12 m2
H2 input	267.50 kg/year
CO2 removal	1460 kg/year
Power usage	0.5 kW

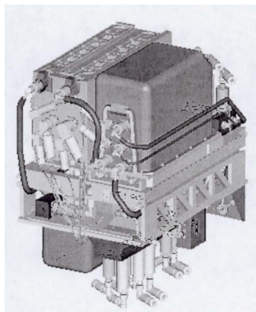


Figure 10.3: Sabatier system

The Sabatier system uses a CO2 compressor to collect CO2 out of the atmosphere, a CO2 accumulator, a phase separator and of course the Sabatier reactor itself. The products of the reactor gets condensed and the water is separated from the waste gases to be supplied to the water storage tanks. Sabatier system is on a mature TRL [57]. It furthermore has short start up times allowing for flexible use. After the initial 350°C operating temperature is reached, the reaction becomes self-sustaining, meaning that the heat of the reaction (energy comes from the hydrogen) keeps the conditions inside the reactor at the 350°C. This results in a low power draw compared to other carbon dioxide removal systems. [27]

There is some experimental research going on for recycling methane to get hydrogen back [37]. However the TRL is not advanced enough to be used for the lunar mission. Future research can be done in this area to save some hydrogen mass which will lower the resupply mass.

In the safehouse, Sabatier system isn't an option since the buffer time is much smaller than in the main habitat due to the volume difference. Therefore the open loop LiOH carbon scrub method is the prefered and chosen method. In addition to that, the system might have out-times or require maintenance and no tools will be available in the safe house in case of such events.

[2]URL: https://ntrs.nasa.gov/api/citations/20050207456/downloads/20050207456.pdf
[3]https://www.nasa.gov/mission_pages/station/research/news/sabatier.html

Food Sizing

To find the exact food sizing numbers three menu's for an Early Mars Diet were considered: *"Modified Shuttle Training Menu", "Low Moisture Content Menu"* and *"Menu Containing Some Frozen Food"* [2, p. 135]. The biggest factor in deciding the optimal menu is the moisture content. A higher moisture content in the packaged food, means less food re-hydration water and less additional drinking water. A low moisture content menu has a lower packaged food mass, precisely because it requires more re-hydration and drinking water. And as seen in the Power section, Chapter 9, water can serve another purpose: to use in the oxygen generator. Therefore, using water is the more attractive option because higher dry food mass serves no other function and increases solid waste. So to minimise the packaged food mass, the *"Low Moisture Content Menu"* is the best option. The menu has an energy content of 12.707 MJ/CM-d, which is equivalent to 12148 kcal/day, and contains all the required nutrients, thereby fulfilling both of the requirements.

Table 10.3: Properties and total sizing values for the *"Low Moisture Content Menu"*

Parameter	Unit	Value
Metabolic Load Nominal	MJ/CM-d	11.82 [(a)]
Metabolic Load High	MJ/CM-d	13 [(a)]
Safety factor	-	1.14
Food "as shipped"	kg/CM-d	0.99 [(b)]
Re-hydration water	kg/CM-d	1.38 [(b)]
Additional drinking water	kg/CM-d	1.42 [(b)]
Packaging mass	kg/CM-d	0.29 [(b)]
Locker mass	kg/CM-d	0.35 [(b)]
Locker volume	m^3/CM-d	0.00486 [(b)]
Trash mass	kg/CM-d	0.34 [(b)]
Total trash mass	kg	496.4
Total locker/food volume	m^3	8.1
Total food and locker mass	kg	2717.6

References

[a] *Life Support Baseline Values and Assumptions Document* [2, Table 4-66]

[b] *Advanced Life Support Requirements Document* [20, Table 3-1]

To size the final amount of food needed to bring to the lunar surface, the safety factor must be determined. The safety factor has been based on the ratio between nominal and high activity metabolic load of 11.82 and 13.489 MJ/CM-d respectively, giving a safety factor of 1.14. The values of the food mass and volume (including the storage), is presented in the table below.

According to requirements; *"The habitat shall provide 12000 calories of food per day"* and *"the habitat shall provide a healthy balanced diet containing all macro- and micro-nutrients needed for optimal health"*. Both of these requirements are not problematic to comply with. However, food is not recyclable like oxygen and water. Growing crops inside the habitat is a possibility and has additional benefits like removing CO_2 from the atmosphere and producing O_2. However, this requires extra habitat area and water and further increases the risk of microbial growth. Therefore as the first mission to the lunar surface, this is not considered a viable option, even though it is a promising avenue to explore in future missions.

Heating the food will be done similar to the ISS. Food is either heated with hot re-hydration water or by the use of conductive heating. Although microwaves seem the most obvious, some food packaging contains metallic foil to increase shelf life. Furthermore, the variable moisture content of food and cold spots make microwaves a less consistent heating method [2, p. 116]. For the safehouse the same approach is taken, only now for 21 days, resulting in a total mass and volume of 156 kg and 0.47 m^3 respectively.

Clothing

Clothing might not seem part of life support but after extensive periods of time, bodily fluids will get into clothes and produce an unpleasant odour. The two main choices to be made for clothes in outer space is if they are single use, or recycled using washing equipment. Single use clothing has the advantage that one does not require laundry equipment and the additional water it requires. However, for long missions, the overall mass of single use clothing can exceed that of the water, laundry and recyclable clothing all together. Studies have been conducted determining the amount of days to be break even. For a lunar outpost using a closed loop

Table 10.4: Properties and total values for clothing on the main habitat and the safehouse.

Main habitat

Parameter	Unit	Value
Clothes mass	kg/CM-d	0.0373[(a)]
Clothes volume	m^3/CM-d	0.00022[(a)]
Total clothes mass	kg	54.458
Total clothes volume	m^3	0.3212

Safehouse

Parameter	Unit	Value
Single use mass	kg/CM-d	0.343[(a) (b)]
Single use volume	m^3/CM-d	0.00135[(a) (b)]
Total clothing mass	kg	28.812
Total clothing volume)	m^3	0.1134

References

[a] *Life Support Baseline Values and Assumptions Document* [2, p. 113 Table 4-48]

[b] From all the single use values presented, this one is the most accurate as it is empirical data from the ISS. The single use volume was determined by averaging the three most similar studies.

water system accommodating 4 crew members, the break even point was found to be between 24 and 81 days [2, p. 114 Table 4-50]. This mission will have a duration longer than 81 days (1 year) so clothes will be recycled inside the habitat. The washer/dryer that will be used for this uses 51.3 kg of water per load and has a capacity of 4.5 kg/load [2, p. 114]. Based on these figures a washer dryer water usage of 2.62 kg/CM-d is found. So roughly every 5 days the laundry will be done.

For the safehouse, clothing only has to be provided for 21 days, which makes single use clothing the best option. The values for the clothing specifications is presented in the table below, both for the main habitat and the safehouse.

Temperature, Atmosphere and humidity control

As there is no air on the Moon, all of it will have to be transported from Earth. Knowing the final pressured volume of the habitat, the amount of gas needed can be calculated. According to requirements **LS-SYS01-01.1**: *"The habitat shall maintain an air mixture of at least 21% Oxygen during the entire mission"* and **LS-SYS01-01.3**: *"The habitat shall maintain air pressure of at least* 30 kPa*"*. Both of these requirements enforce a pressure similar to that of Earth. To elaborate; a 21% O_2 level with an atmospheric pressure of 30 kPa, would mean that per breath a crew member would inhale roughly 3 times less oxygen mass-wise with respect to standard Earth conditions. This is the equivalent to breathing on the summit of Mount Everest. This would, without a doubt, cause a loss of crew. Therefore, the habitat's internal pressure is 101.3 kPa. The amount of gas needed can be calculated using the ideal gas law, shown below.

$$pV = nRT \longrightarrow m = \frac{pVM}{RT} \tag{10.3}$$

where, p is the pressure, V the volume, R the gas constant, T the gas temperature and M the molar mass.

Besides the gases needed to fill the habitat, losses should also be accounted for. The most significant loss of gas that will be taking place during operation is that of compression/decompression of the airlock. Because a perfect vacuum is difficult to create within a limited time-frame, losses occur. For each cycle roughly 10 % will be lost to space and cannot be recovered. With more time and power for the airlock pump, loss rates as low as 5 % may be achieved [2, p. 142][11]. For the current estimation, a 10 % loss rate is used. The amount of airlock cycles has been set to once a week, so 52 during the mission. Furthermore, the inflatable itself also isn't completely air-tight. As explained in Chapter 7, the permeability of the inflatable allows 2.8 L of oxygen and 1.1 L of nitrogen to escape everyday. This is equivalent to 4.0 and 1.28 g/d of oxygen and nitrogen respectively.

A much more severe source of leakage is present within the life support systems. The leakage rate of life support systems is estimated to be at 1.81 kg/d [11, p. 452]. For the main habitat, a closed loop oxygen system is the most mass-efficient option. This can be seen in Table 10.5, where an open loop system would require 1194.28 kg of oxygen. For a closed loop system the total mass equals that of the required oxygen and the OGS resulting in a total mass of $497 + 113 = 610$kg, which is indeed more efficient. For the safehouse it is much more efficient to use an open loop oxygen system. To obtain the final oxygen and nitrogen needed to bring from Earth a 25% safety factor was taken.

Table 10.5: Properties and total values for gas storage

Parameter	Unit	Value (Main habitat)	Value (Safehouse)
Oxygen usage	kg/CM-d	0.818 [(a)]	
Open loop oxygen usage	kg	1194.28	68.71
Pressurised volume	m^3	805.82	50.70
Oxygen volume percentage	%	21	
Oxygen to fill volume	kg	223.59	14.07
Nitrogen to fill volume	kg	736.34	46.33
Number of airlock cycles	-	52	2
Airlock loss rate	%	10 [(b)]	
Lost oxygen due to airlock	kg	18.79	0.097
Lost nitrogen due to airlock	kg	61.87	0.320
Leakage rate	kg/d	1.83	1.81
Leaked oxygen	kg	155.23	8.87
Leaked nitrogen	kg	511.21	29.23
Total oxygen (25% safety)	kg	497.00	114.69
Total nitrogen (25% safety)	kg	1636.78	94.84

References

[a] *Life Support Baseline Values and Assumptions Document* [2, p. 66 Table 4-48]

[b] *Life Support Baseline Values and Assumptions Document* [2, p. 142] *The cost of life support in manned lunar bases* [11]

As explained in the Power section, to efficiently store gases volume wise, they have to be liquefied. The oxygen required for life support that has been calculated in Table 10.5 will be stored in the same tank as the electrolysis oxygen as seen in Table 9.3. The nitrogen will also be stored as a liquid of which the specifications are summarised in Table 10.6.

Table 10.6: Specifications of liquid nitrogen storage.

Parameter	Unit	Value
Liquid nitrogen temperature	K	121 [(a)]
Liquid nitrogen density	kg/m^3	509.5 [(a)]
Liquid nitrogen volume	m^3	3.22
Tank mass	kg	857 [(b)]

References

[a] https://www.engineeringtoolbox.com/nitrogen-N2-density-specific-weight-temperature-pressure-d_2039.html

[b] Based on 0.524 kg of tankage per kg of liquid nitrogen. *Life Support Baseline Values and Assumptions Document* [2, p. 68]

Having inflated the habitat, the atmosphere needs to be kept breathable. The most important atmosphere control unit has been discussed already which is the CDRA. Besides this, the atmosphere needs to have a pleasant temperature and humidity. Requirement **LS-SYS01-06** says the following: *"The temperature inside the habitat shall remain at an average of 22°C"*. Furthermore, a nominal relative humidity of 40 % will have to be maintained in the habitat [2, p. 66]. In the ISS, this was achieved by the Common Cabin Air Assembly (CCAA). However, the condensing heat exchanging unit's lack in reliability (based on the many repairs) [34] and it's performance has room for improvement [84]. A study done by (Noyes 2018, [84]) has developed a proof of concept and a detailed design whose aim is to be reliable and robust for long duration exploration missions. The unit itself is called Membrane Microgravity Air Conditioner (MMAC). It is capable of removing 4 kW of sensible heat and 1 kW of latent heat at 28 m^3/min of airflow. It's mass is 68 kg, and therefore is better on each front with respect to the ISS CCAA of 2.5 kW and 97 kg. The unit is self is cubical with dimensions of 0.67 m all around.

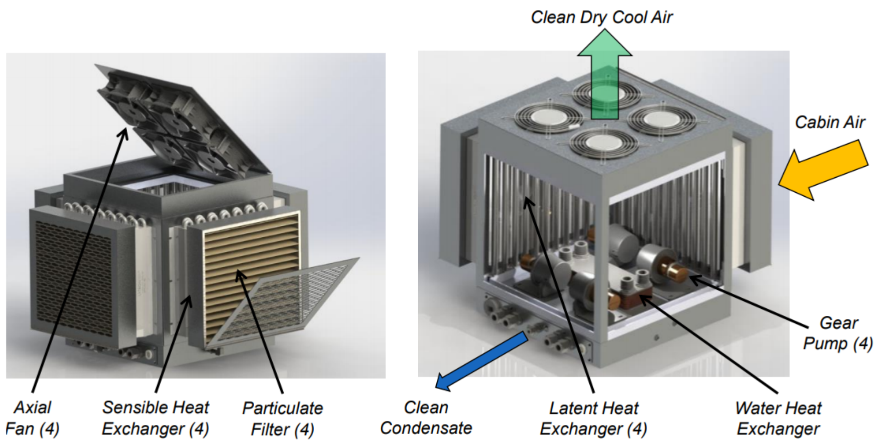


Figure 10.4: MMAC Conceptual Design render and it's sub components. [84]

From Section 7.5.5, it was calculated that the habitat produces more heat due to the internal electrical systems than the heat lost to the environment. Hence, to maintain the desired habitat temperature of 22 °C, the MMAC units are required to remove the heat from the habitat. 2 MMAC units will be used, since they can remove a total of 8 kW of sensible heat and 2 kW of latent heat. This is greater than the value of maximum heat generated that was calculated in subsubsection 7.5.5, and it can account for any unexpected additional heat generated due to certain systems working overtime or if the astronauts perform more physical activities and generate more heat.

Regarding the cooling of the heated coolant of MMAC, the lunar environment will be used as natural cooling, by connecting a pipe from the MMAC cooler connection to the airlock. The MMAC unit contains a water pump at the cooler connection, so the heated water can flow from the MMAC, flow through the pipe outside the airlock to the lunar surface, cool down via radiation, and the cold coolant will be pumped back into the MMAC. This pipe will be coloured in black, and will extend outside the habitat and lead back in through a separate opening.

Condensation of water vapour could happen on the walls of the habitat if the temperature difference between the wall and the atmosphere is large and the dew point is high. However, this can be mitigated with the MMAC as well, since the MMAC decreases the humidity of the habitat. Although this was never a concern, since the temperature difference between the wall and the internal atmosphere was smaller than the temperature difference that would cause condensation, the MMAC can ensure that condensation will not happen even at extreme local cases that were overlooked.

Air circulation

The air needs to circulate around the habitat so that oxygen generated can reach the sleeping area at the further end of the habitat. It will also circulate the carbon dioxide to where it can be collected, and also the heat can be spread out equally around the habitat with air circulation. For this, the internal walls of the habitat will contain holes on the top left and top right corners, with simple fans placed within them. One fan will push the air to the sleeping area, while the fan on the opposite side will operate in the opposite direction, effectively creating an air circulation pattern.

Water usage and waste production

To find the total amount of water needed to bring, all waste water production must be assessed first. These are listed in Table 10.7. The most important parameters are the UPA and WPA recovery rates. These drastically influence the amount of water loss, so ideally they are as high as possible. Currently the recovery rate for the UPA is 86 % but with improvements likely to become 90 % in the future . This is similar for the WPA going from the current 95 % to 96 %. [14].

Table 10.7: Values of the waste water generation, and total water mass.

Parameter	Unit	Value
Urinal flush	kg/CM-d	0.5 [(a)]
Urine	kg/CM-d	1.5 [(a)]
Crew latent (MMAC condensate)	kg/CM-d	2.27 [(a)]
Rehydration water	kg/CM-d	1.38 [(a)]
Additional drinking water	kg/CM-d	1.42 [(a)]
Hygiene water	kg/CM-d	7.17 [(a)]
Washer/dryer usage	kg/CM-d	2.62 [(b)]
Dishwasher usage	kg/CM-d	1.4 [(c)]
Payload water	kg/CM-d	0.5
UPA recovery rate	%	90
WPA recovery rate	%	96
Total waste water	kg/CM-d	15.96
Water loss	kg/CM-d	0.83048
Total water loss	kg	1212.5
Water reserve	kg	550
Total water mass (safety 25 %)	kg	2065.6

References
[a] *Life Support Baseline Values and Assumptions Document* [2]
[b] Elaborated on in the clothing section
[c] Assuming it is used once per day with a water usage of 5.6 kg/load. *A Dishwasher for the Space Station* [112, p. 6]

10.3. Fire Safety

The outbreak of a fire is one of the most feared events in a space habitat. This is because it can quickly lead to a mission failure or even loss of crew in the worst scenario. Therefore, fire control is of the up-most important part of the habitat design. Naturally, the habitat is designed in such a way to prevent fires from starting in the first place. In the unlikely event a fire does break out, there are three things that are very important for the survival of the mission/crew

- Fire detection: If a fire starts, it should be detected as quickly as possible so the crew can be alarmed. On earth a smoke detector makes use of buoyancy-driven flows caused by large amounts of hear released in fire zones which drive smoke up. With moon gravity, the buoyancy driven flows are less present. Therefore smoke detectors need to be placed in air ducts. In the habitat smoke detector will be placed strategically on each MMAC unit and at the entrance of air ducts. Smoke detectors are not suited for large open spaces such as the habitats living quarters. This is because the smoke development in these areas will take a long time to reach the smoke detectors located at the air duct. In this area flame detectors will be used. These monitor the wavelength of incoming electromagnetic particles (light). They are not depended on smoke particles moving trough the detector. Fire detectors are an ideal solution in large open spaces.
- Fire containment: If a fire is present it should not be allowed to spread quickly through the habitat. Doors in the habitat will be self-closing. The habitat walls and The inside of the inflatable will contain Teijin-conex material which is fire retardant. More about the lay-up of the habitat skin can be read in Section 7.3.3. The internal walls separating different areas in the habitat will also have layers of Teijin-conex material integrated in them. The floor will also have a layer of this fire retardant material. All this makes sure that a fire will be contained to the room where it has started.
- Fire suppression: If a fire breaks out in the habitat it needs to be suppressed as soon as possible. If the fire alarm is triggered the air circulation would be immediately stopped by the habitat control computer. Every room is equipped with a smoke or fire detection device. Next to this every room has an automatic foam fire extinguishers installed which fills the room with high expansion foam which suppresses the fire. In case the fire is still burning or the previous mentioned system fails, there are multiple carbon dioxide extinguishers located in the habitat on strategic locations. One in both airlock chambers, one in the entrance room with all the power distribution systems such as the fuel cells and life support systems, one in the scientific experiment room, one in the pantry, one in the living room and one in the corridor between the bedrooms. Carbon dioxide is chosen as the manual extinguishers because it does not destroy electric equipment. The automatic extinguishers do not make use of this because to much

carbon dioxide in the air would make the habitat atmosphere un-breathable by the astronauts.

In case a fire can not be stopped, the astronauts leave the habitat in the airlock, put on their suits and leave for the save house. The wall between the airlock and the habitat is made from a fire retardant material, keeping the astronauts save while putting on their suits [27].

10.4. Lunar gym

Humans experience muscle and bone loss when living for prolonged time in micro gravity.To much bone and muscle degradation can become permanent if losses are to high. Physical exercise is the only method to minimise these losses. Therefore in the habitat a dedicated room is reserved for working out. With reduced gravity on the moon exercise material using traditional weights would be to heavy. The gym will therefore be optimised for the use of restrain bands and harnesses. These make use of elastic force which is the result of elongation of restrain bands. This is independent from the gravity present and therefore is an ideal light weight solution for the habitat. The gym will be equipped with a wide variety of bands with different stiffness. Body harnesses will be available besides floor hooks on which restrain bands can be coupled. With this, basic exercises can be performed. A pull up bar and alight weight treadmill[4] will also be available which will be combined with the previously mentioned restrain bands.

10.5. Space suits

The space suits chosen for the mission is the Mark III. The suit is currently being used in simulated planetary field tests. Mark 5 offers superior mobility. The suit is made of hard structural composite material for the upper body, upper arms and legs. While soft fabric is used for the legs and lower arms. The suit also has hip abduction/adduction joints. Together with the soft joints and the bearings, all the expected lunar mobility tasks can be performed with acceptable effort. Being able to do tasks such as kneeling and picking up an object, manoeuvring uneven terrain, driving lunar rovers or doing maintenance tasks outside are prioritised. Therefore, the Mark III suit is selected. Fun fact: Handstands and somersaults could also be performed in the suit. Incorporating lightweight composite materials he Mark III suit itself has a mass of 36 kg and its Portable Life Support System (PLSS) a mass of 15 kg. (Semi-)hard shell suits are also preferred because regolith can be blown off relatively easy.[5] Below one can see the dimensions of the space suit:

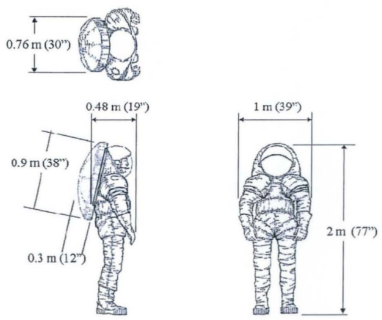


Figure 10.5: The Mark III space suit

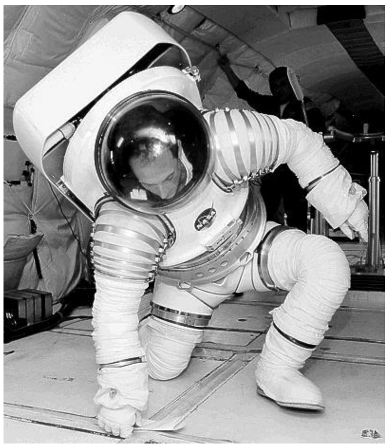


Figure 10.6: The Mark III space suit

The spacesuits will be stored into the inner airlock chamber (chamber 1)

10.6. Subsystem Interactions

As explained in the logistics section, starting the construction requires immediate power. Because the solar arrays and their connections have to be constructed first, an alternative power system was chosen which does not rely on the presence of solar radiation; namely hydrogen fuel cells. Because hydrogen fuel cells will also be used in the habitat itself, it makes sense to use the reaction product (water) to produce potable water. Seen below is the schematic overview of the resource flow between the subsystems in and outside the habitat. There is interface between the regenerative fuel cell system and the life support system. For instance

[4]https://www.urevosports.com/collections/urevo-treadmills/products/urevo-folding-treadmill-electric-running-machine
[5]URL: http://www.astronautix.com/n/nasamarkiii.html

the Sabatier CDRA system uses hydrogen which comes from the hydrogen tank which is also used for the fuel cell. This fuel cell produces water which on its turn can be used by the life support system.

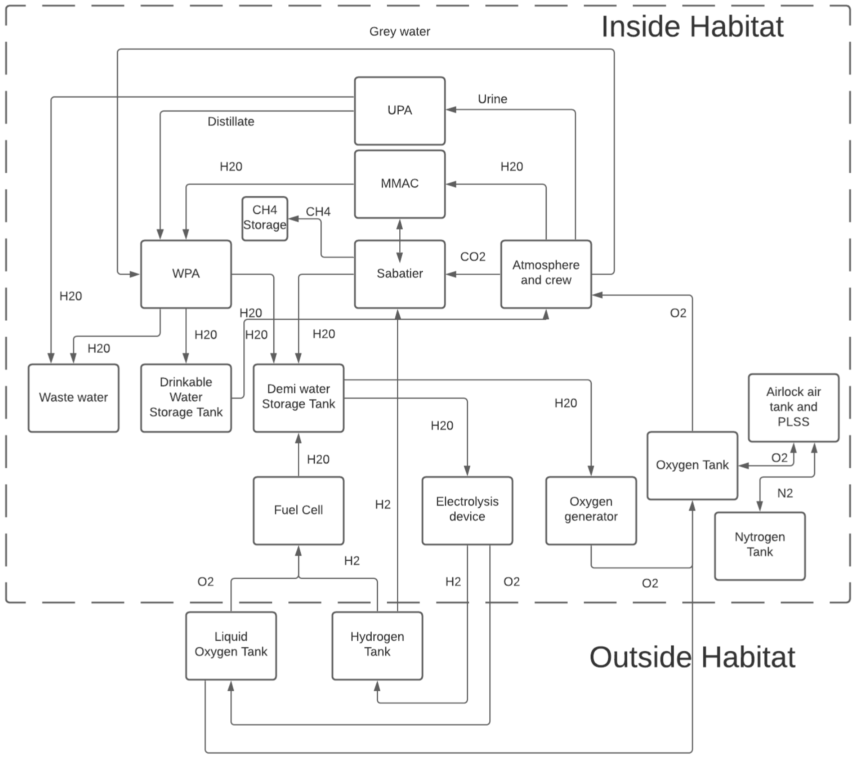


Figure 10.7: Hardware diagram of the life support and power subsystems

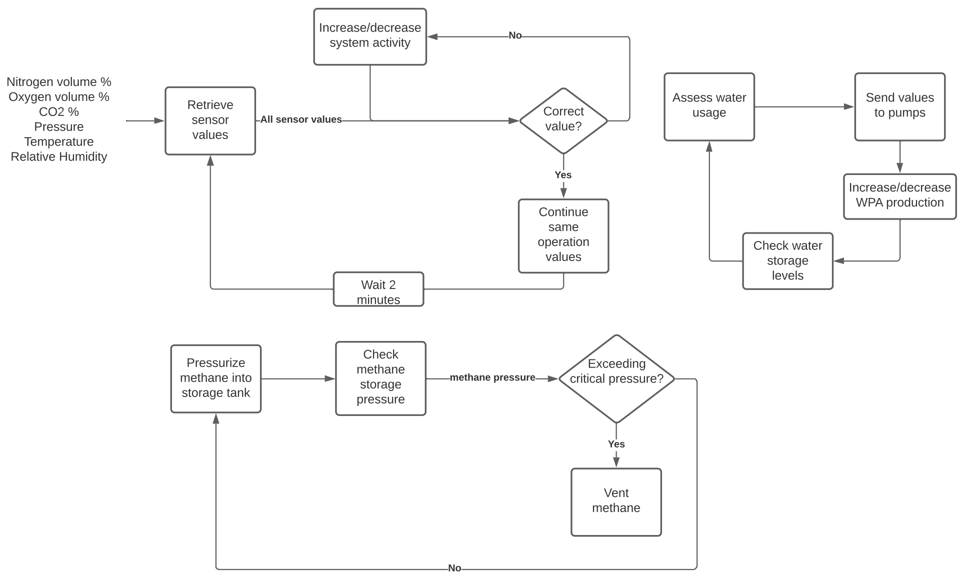


Figure 10.8: Software diagram of various subsystems within life support.

10.7. Compliance Matrix

The table below shows the most relevant requirements for life support and if they comply.

Table 10.8: Compliance matrix for the life support subsystems

Requirement Code	Requirement Text	Compliance	Method	Justification	Source
LS-SYS01-01.1	The habitat shall maintain an air mixture of at least 21% Oxygen during the entire mission	Yes	Inspection		
LS-SYS01-01.2	The habitat shall have an air storage tank	Yes	Inspection		
LS-SYS01-01.3	The habitat shall maintain air pressure of at least 30 kPa.	Yes	Inspection	Pressure is 101.3 kPa	-
LS-SYS01-02.1	The water system shall provide 16 litres of drinkable water per day	Yes	Analysis	6.8 kg/h	[86, p. 5]
LS-SYS01-02.2	The water system shall provide 7 litres of water for hygiene	Yes	Inspection	7.17 kg	[2]
LS-SYS01-02.3	The water system shall provide 2 litres of water per day for scientific experiments	Yes	Inspection	See final water table	-
LS-SYS01-02.4	The habitat shall have water filtering systems for ”recycling” used water	Yes	Inspection	Use of WPA and UPA	
LS-SYS01-02.5	The habitat shall have a spare water supply of 500 litres	Yes	Inspection	515 L based on ISS ratio	Table 10.7
LS-SYS01-03.1	The habitat shall provide 12000 calories of food per day	Yes	Inspection	12148 kcal/day	[2, p. 135]
LS-SYS01-03.2	The habitat shall provide a healthy balanced diet containing all macro- and micro-nutrients needed for optimal health	Yes	Inspection		[2, p. 135]

11. Internal Layout

11.1. Requirements

LS-SYS01-04: The habitat shall provide facilities for resting and privacy activities

- **LS-SYS01-04.1**: The habitat shall provide 4 sleeping blocks each at 2 m^3

LS-SYS01-05: The habitat shall provide facilities for personal hygiene related activities

- **LS-SYS01-05.1**: The habitat shall provide a bathroom
- **LS-SYS01-05.2**: The habitat shall provide a toilet
- **LS-SYS02-02**: The habitat shall provide leisure related material
- **LS-SYS02-03**: The habitat shall provide exercising equipment
  - **LS-SYS02-03.1**: The habitat shall have a dedicated work out area
  - **LS-SYS02-03.2**: The workout area shall have exercise equipment which are sufficient enough to maintain muscle and bone mass in low gravity conditions

SH09: The habitat shall allow the astronaut to perform scientific experiments while in the habitat

- **SR-SYS09-01**:The habitat shall have an sub part of 6m2 dedicated to scientific research
  - **SR-SYS09-01.1** The scientific research area shall have general tools
  - **SR-SYS09-01.2** The scientific research area shall have an emergency fire extinguisher system
  - **SR-SYS09-01.3** The scientific research area shall have an data acquisition system
  - **SR-SYS09-01.4** The scientific research area shall have a video recording system
  - **SR-SYS09-01.5** The scientific research area shall have a working bench
- **SR-SYS09-02**:The habitat shall accommodate 1000kg of scientific research modules (100 modules with a dimension of 30x30x15cm)
  - **SR-SYS09-02.1**: The scientific research testing modules should be easy accessible and storable in racks

11.2. Final layout

As mentioned before in Section 6.3 the habitat will consist of a rigid pre assembled part and a inflatable part. Most complex systems that are not possible to build up by hand and/or require a lot of connections to other systems will be placed in the pre-assembled part. These systems need to be working at the very start of the deployment of the habitat. In the appendix in Figure C.1 can see the top view of the habitat lay out. In the explanation about the lay out below, this drawing is referenced to.

11.2.1. Dimensions of the habitat.

To arrive at the dimensions of the habitat, the encapsulated volume has to be considered first. This primarily depends on the crew size and the mission duration. Based on studies ([92] [97]) and the ISS, the required habitable volume for a mission duration of 1 year should be between 60 and 75 m^3 per crew member. [1] Using this an inflatable diameter of 6 m was found in the Midterm report [39]. However, to make both storage and excavation more simple it was decided that the floor sitting on the surface should be flat. Additionally, this makes the inflatable more stable and reduces the risk of rolling during assembly. The final cross section including the storage was shown previously in Figure 8.4. The entire inflatable length is still 27 m, only now with a total pressurised volume of 805.82 m^3. The total habitable volume was found to be 247 m^3, or roughly 62 m^3 per crew member. One additional consideration: Standard values for corridor width, door sizing, minimal bedroom sizing, ceiling height, desk size and many more dimensions were taken from *"De Menselijke Maat"*, an architect handbook on human habitat interaction and design[41].

[1] URL: https://www.nasa.gov/feature/facts-and-figures [cited 19-05-2021]

11.2.2. Pre-assembled part

Starting from the left most part of the topview of the pre-assembled part. After entering through the airlock on both the right and the left one can find the most important subsystems for life support. Oxygen generation, Carbon dioxide removal assembly and the water processing units are all placed here. They are placed with accessibility in mind. By offering plenty of space around them the crew can do maintenance work on them without great difficulty. Also in this part the water storage is located, and the power subsystems such as the fuel cell, electrolysis device and the power distribution. Furthermore the gas storage tanks necessary for inflating the habitat are located below in the storage floor. Next comes the control room on the top. Here are all the important computers placed that control the habit. Two crew members at the time can work in this room behind a desk with pre installed monitors. They have access to all habitat control systems, data and communication. Across the control room is the hygiene area. This includes a toilet and a bathroom. Due to the complexity which includes the water connections required, it is placed in the pre-assembled part.

11.2.3. Inflatable part

The inflatable part comes as an empty room after it has been inflated. Upon the astronauts' arrival cargo boxes have already been brought into the habitat as described in Section 8.2.2. This part of the habitat will be constructed by the astronauts them self. They will first start with setting up the internal walls which separate the rooms. After that they build in the equipment necessary for each room. Starting on the top left, there is the scientific experiments room. Here 6m2 is available for doing scientific research. A workbench as seen in Figure 11.2 is available as well as closet storage for tools and consumables alike. Also present in the closet are all the scientific research modules. These are on-Earth pre-fabricated modules which each contain all material necessary for conducting an experiment. The room also has general experiment equipment such as heat guns, vacuum pumps, observation equipment etc.

Across this room on the bottom of the topview is the pantry area. It is installed next to the water processing assembly because in this room all water consuming systems are placed. This reduces tube/pipe length and therefore saves mass. Placed here is the pantry for food and drink preparation, a dishwasher, a clothes washing machine, food storage and a small fridge for beverages. Next to the pantry, comes the lunar gym, here the astronauts can work out. more on this can be read in section Section 10.4. Across the gym will be a general storage area where consumables can be stored which need frequent access.

Then comes the general living area. The crew uses this space to chill out and relax. For this a projector-TV is available as well as an inflatable couch, lounge chairs and a reading corner will give the astronauts a cozy home experience. They also use the space to have breakfast, lunch or diner. A table with 4 chairs is available for this. This table can also be used for playing board games taken within the leisure cargo box. There is also a working desk in this room behind which the astronauts can sit with their laptop and or paperwork. In this room also two virtual window screens will be placed on the walls to simulate an earth like environment.[2] This all is to make the habitat more live able on a psychological level for longer periods of time.

Next to the living quarters are the personal rooms for the astronauts. Each of the rooms has a bed, a small desk and a wardrobe. Here they have space for their clothes and private belongings. The rooms are in the back of the habitat far away from the sound producing subsystems such as the life support devices, the scientific experiment room and the gym. This is to ensue proper resting conditions. Each room gets their own color so a crew member really feels at home in their room. They can select their own wall decoration and get a small bulletin board on which pictures of family can be placed. A visual render to give the reader an impression and feel of the habitat can be seen in Figure 11.1.

[2] https://pid.samsungdisplay.com/en/digital-signage/video-wall-display/lti460hn09

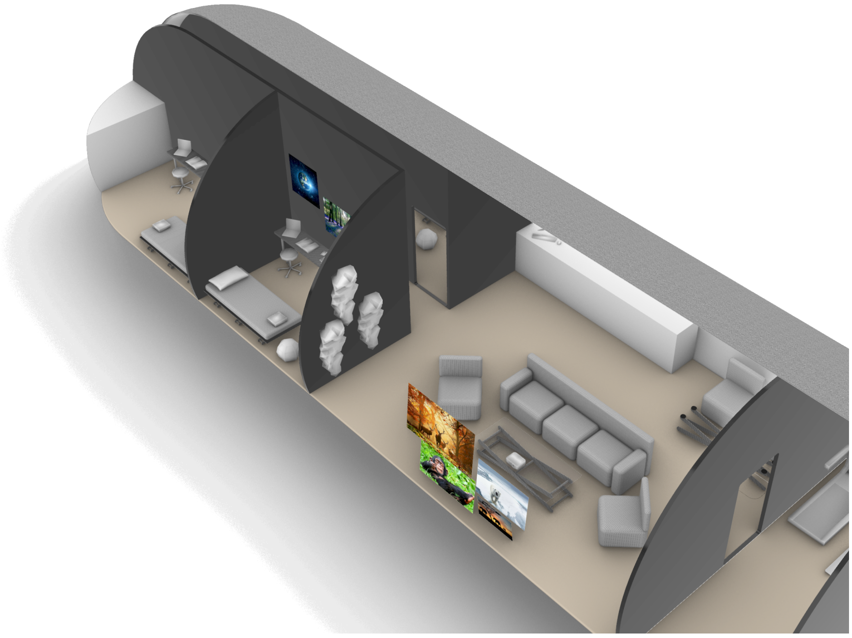


Figure 11.1: The living room and the bedrooms

Table 11.1: Compliance matrix for layout related requirements

Requirement Code	Requirement Text	Compliance	Method	Justification	Source
LS-SYS01-04.1	The habitat shall provide 4 sleeping blocks each at 2 m^3	Yes	Inspection	Area is 6.6 m^2 and volume 15.84 m^3	Figure C.1
LS-SYS01-05.1	The habitat shall provide a bathroom	Yes	Inspection	Area is 1.08 m^2	Figure C.1
LS-SYS01-05.2	The habitat shall provide a toilet	Yes	Inspection	Area is 1.08 m^2	Figure C.1
LS-SYS02-02	The habitat shall provide leisure related material	Yes	Inspection	Leisure is accounted for in the living area, and the amount of cargo blocks for personal belongings	
LS-SYS02-03.1	The habitat shall have a dedicated work out area	Yes	Inspection	Area is 6 m^2	Figure C.1
LS-SYS02-03.2	The workout area shall have exercise equipment which are sufficient enough to maintain muscle and bone mass in low gravity conditions	Yes	Inspection	Elastic restraint bands used to simulate gravity on Earth	Section 10.4
SR-SYS09-01.1	The scientific research area shall have general tools	Yes	Inspection	Accounted for in the size of the work-bench	Figure C.1
SR-SYS09-01.2	The scientific research area shall have an emergency fire extinguisher system	Yes	Inspection	Each room has a dedicated fire extin-huisher	Section 10.3
SR-SYS09-01.3	The scientific research area shall have an data acquisition system	Yes	Inspection	Accounted for in the size of the work-bench	Figure C.1
SR-SYS09-01.4	The scientific research area shall have a video recording system	Yes	Inspection	Accounted for in the size of the work-bench	Figure C.1
SR-SYS09-01.5	The scientific research area shall have a working bench	Yes	Inspection	Accounted for in the size of the work-bench	Figure C.1
SR-SYS09-02.1	The scientific research testing modules should be easy accessible and storable in racks	Yes	Inspection	Accounted for in the size of the work-bench	Figure C.1

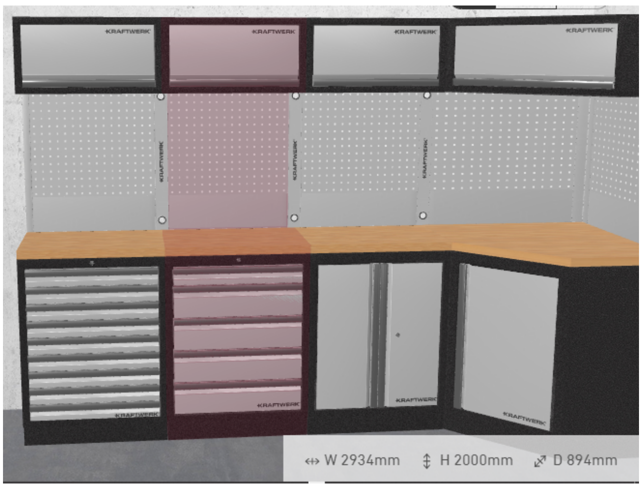


Figure 11.2: The workbench in the scientific research area with storage area on top and drawers in the bottom

12. Safe House

In case of an emergency due to catastrophic failure of the main habitat, a safe house should be available and will placed nearby the main habitat to provide shelter for the 4 astronauts. A number of requirements on the safe house are set and will be discussed in Section 12.1. From these requirements, a layout of the safe house will be detailed in Section 12.2 for the interior and in Section 12.3 for the external part of the safe house.

12.1. Requirement

One of the mission requirement is to account for a safe house to be available in case of a failure in the main habitat, which shall be capable of housing the four astronauts for 21 days. Such safe house shall be designed with a number of requirements in mind that have already been discussed in the Baseline report [38], these are the following:

SH06: The habitat shall have a 'safe house' which can be reached within 60 minutes from the habitat
MLOSYS0601: The route to the safe house from the habitat shall be easily traversable by astronauts on foot.
MLOSYS0601.1: The route shall be selected on the basis of terrain flatness and lack of harmful materials and conditions.
LSSYS0602.1: The emergency habitat shall contain water, oxygen, and food for 21 days.
STSYS0602.2: The emergency habitat shall be constructed with an additional layer of structural and radiation safety than the general habitat.

Requirements **MLOSYS0601** and **MLOSYS0601.1** are directly linked to requirement **SH06** since the safe house will be placed nearby the main habitat to allow quick access. This means that, a terrain analysis will be done before building the main habitat making sure the route to the safe house is safe and traversable.

Requirements **LSSYS0602.1** and **STSYS0602.2** will be looked at closely when designing the internal (Section 12.2) and external (Section 12.3) safe house layout respectively.

12.2. Internal Layout

The safe house should contain the necessary facilities and resources for 4 astronauts to live in during 21 days. It will contain a number of essential subsystems such as water, food and power supplies.

In addition to the essential life support supplies needed a number of facilities will have to be included in the safe house. It is important to understand that the safe house is purely used for emergency, therefore, unneeded items to survive, such as; extra space for comfort, individual rooms or space for scientific experiments won't be included. On top of that, the life support systems will mostly be open-loop meaning no recycling will take place since it requires a large number of additional equipment and such process is not necessary for short period of time. The safe house will thus be divided into 3 different floors. The first floor will be partially underground (1m deep) for safety reasons and extra radiation protection and will contain the sleeping and hygiene areas. A total number of 2 bunk beds (for space optimisation) will be available together with a hygiene and toilet area, their respective dimensions can be seen in Figure 12.1. Then, the second floor will mainly contain the living space and workout area. The entrance, through and airlock, will also be done through this floor as can be visualised in Figure 12.1. Half of the second floor will be used for the astronauts to live in or eat while a quarter of it will be dedicated to a workout area. Even if the safe house living period is very short, it is essential for the astronauts to workout in some ways, since even between 5 to 11 days without working out, the astronauts experience up to a 20% loss of muscle mass in space [1]. Such phenomenon is unacceptable for the astronauts health, and workout using body weight, elastic bands or even small machines (nothing compared to the main habitat workout machines) will be done daily by the astronauts in order to prevent muscle atrophy. The third and thus last floor, will be smaller in height than the other two and will only be used and accessed for storage purposes (food, power, water etc, ...). It will also contain all the necessary medical facilities and devices needed in case an astronaut is injured. In addition to essential subsystems for the astronauts a number of safety systems, such as radiations sensors will be added to the safe house. Such subsystems will be redundant in order to improve the safe house overall reliability and safety, as will be discussed in Chapter 16.

The safe house dimensions were estimated from the Orion spacecraft which was designed for the same

[1]NASA information on muscle atrophy, URL: https://www.nasa.gov/pdf/64249main_ffs_factsheets_hbp_atrophy.pdf [Cited June 7, 2021]

number of astronauts and a similar residence time[2]. The 4 astronauts will live and work for 21 days in a volume of 11 m^3. The safe house will have a total habitable volume (first and second floor only and excluding entrance area) of 36,97 m^3 and habitable surface area (first and second floor only and excluding entrance area) of 14,77 m^2. All the safe house dimensions can be visualised in Figure 12.1.

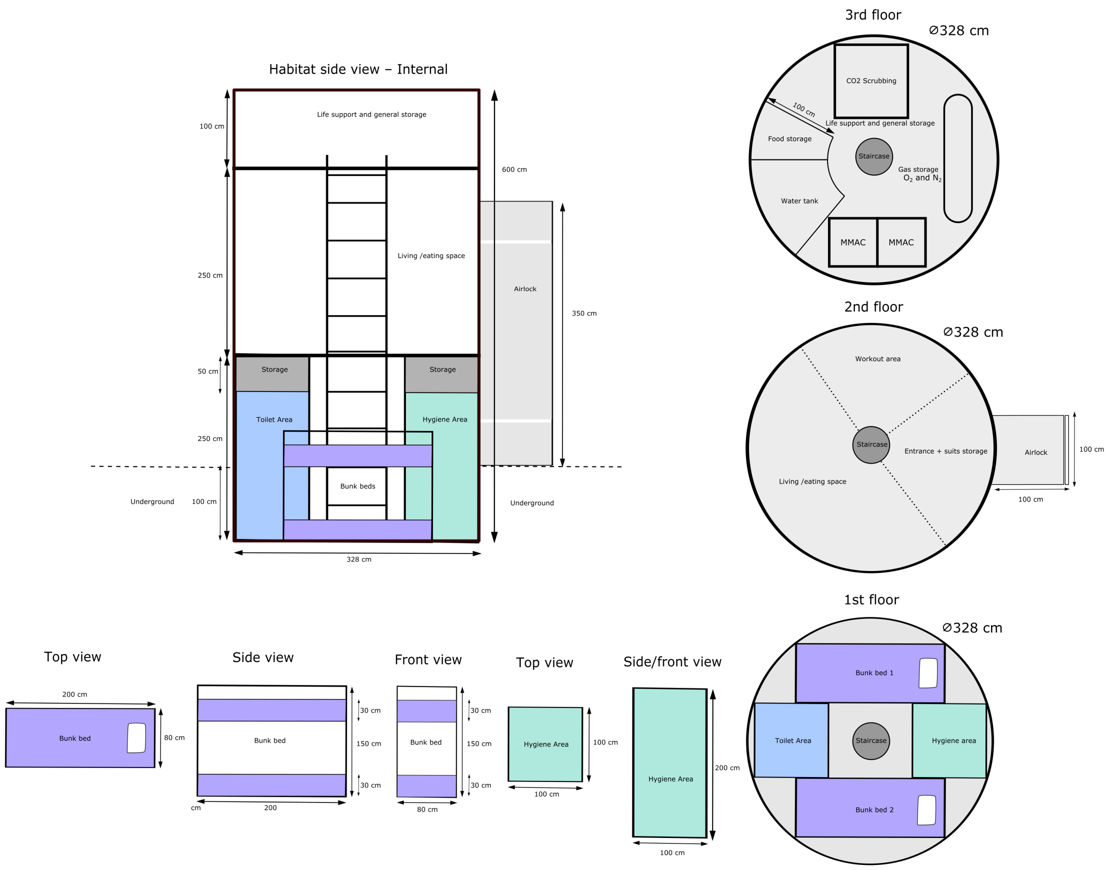


Figure 12.1: Safe house's internal layout of its different floors including dimensions.

12.3. External Layout

The safe house will be used by astronauts for emergency shelter if a catastrophic event was to happen to the main habitat. For this specific reason, the safe house structure should contain an additional safety layer regarding structural strength and radiation safety. It has thus been decided to use rigid walls made out of different materials than those used in the main habitat. Previous research on radiation event shielding indicates that the use of organic materials as well as combinations of typical aerospace materials (such as aluminium or titanium) can provide sufficient protection even with a SPE twice as large in magnitude than any observed before [62][111].

Solar particles are blocked mostly by adding additional material between the Sun and the organism being protected. NASA concluded that the dosage due to an SPE event can be reduced to below 1 [rad] using a combination of the materials presented in Table 12.1 [62].

Table 12.1: Initial wall thickness calculation

Material	**Areal Density Required [g/cm^2]**	**Density of the material[g/cm^3]**	**Thickness required[cm]**
Polyethylene	30	0.975	30.8
Aluminium	40	2.7	14.8
Carbon	37	1.9-3.52	19.5-10.5
Titanium	43	4.5	9.6
		Total thickness	65.7-74.6

Since a total thickness of over 60 [cm] would likely lead to an unfeasible heavy solution, water stored in the

[2]What is Orion?, URL: https://www.nasa.gov/audience/forstudents/5-8/features/nasa-knows/what-is-orion-58.html [Cited June 7, 2021]

habitat will also be used as radiation protection, and only the bottom floor will have the required SPE protection characteristics, while the outer layer of the habitat will be made out of aluminium and polyethylene for adequate solar and galactic cosmic ray protection. An overview of the external habitat can be found in Figure 12.2.

Moreover, part of the radiation protection can be replaced by utilising the water that is already required to keep the astronauts hydrated, extra water will thus be brought on the Lunar surface to provide protection. Water is almost as effective as polyethylene [73], which in turn is more effective than non-organic materials such as metals [111]. Hence, the walls and ceiling at the bottom of the habitat can be reduced by a factor of two while improving their radiation protection characteristics as long as an equal amount of water.

The rest of the safe-house will be made so as to minimise other risks such as puncturing and moonquakes. Magnesium and aluminium both have extensive prior use in aerospace applications, perform exceedingly well in scenarios requiring damping and meteoroid protection, and can easily be combined into an alloy. Moreover, the strength of this material would increase when subjected to low temperatures, such as the ones at the Shackleton crater [82].

In the safehouse, communication will Earth is essential, in case of emergency, or to prepare any return mission. Communication shouldn't however rely on the same transmitter used for the main habitat since, it case of failure of the latter, it might not be efficient anymore. For this reason, a separate antenna will be used for the safehouse which will have constant contact with the antenna used to communicate with Earth. More explanation on the latter will be detailed in the upcoming Chapter 14 Another external safe house feature is the airlock, it has been decided to use an airlock to easily access the safe house without letting any regolith enter and enabling the astronauts to enter the house without altering its inside environments. At this stage, the airlock is simply assumed to be a scaled down version of the main airlock of the habitat.

All in all, the safehouse is a rigid structure, that will entirely be built on Earth, brought to the lunar surface and installed in its 1m deep trench, used in case of emergency.

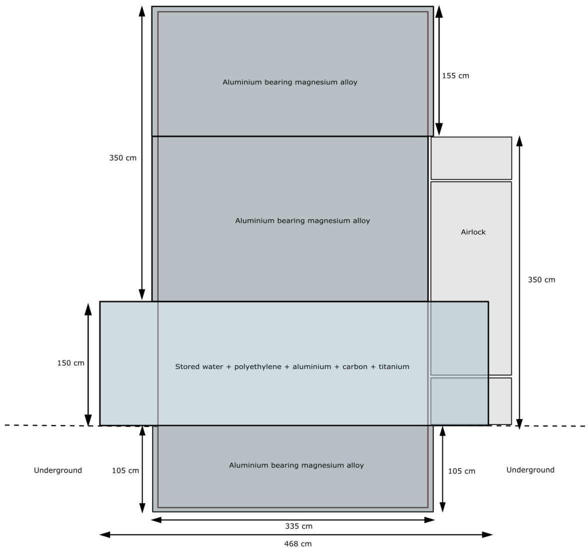


Figure 12.2: Safe house's external layout visualisation of the different protective layers.

12.4. Safe House Mass Estimation

12.4.1. Internal and Airlock Mass Estimation

The subsystems and supplies needed in the safehouse for 21 days and 4 astronauts have all been detailed in Table 12.2. A safety factor of 4 have been implemented for structural components and 1.25 for other types of subsystems. Every subsystems have been scaled from the main habitat subsystems, are find in a similar way as in Chapter 10. The detail on how all these masses have been calculated is explained in Chapter 10.

Table 12.2: Total amount of mass needed per safe house subsystems for 4 astronauts and for 21 days.

Subsytems	Total mass (kg)
Life support	1834
Toilet	80
Beds	100
Ladder	30
Living/eating	150
Airlock pump + CO2	130
Airlock structure	600
Total internal mass	**2924**

12.4.2. External Mass Estimation

The external mass can be estimated by dividing the safe house into its main two components: the bottom floor meant for sleeping, hydrating, and being protected from SPEs, the end caps, the top floor, the section underground, and the airlock. The bottom floor utilises the stored water as well as polyethylene, aluminium, carbon, and titanium in the same proportions as seen in Table 12.1, while the rest is assumed to use 5[cm] layer of an aluminium bearing magnesium alloy, with an assumed density of 2.2 [g/cm^3] (taken as an average of both metals). The bottom floor also uses some of the consumable water as radiation protection, which translates to a thickness 26 [cm]. Using the total thickness estimate of 70[cm] as seen on Section 12.3, this leads to a rigid wall thickness of 44 [cm]. The airlock mass is computed by linearly scaling down the DCIS airlock [43] and then doubling the weight to account for SPE protection, and is assumed to cover one squared meter of the bottom floor (reducing the need for the thicker SPE protection layer in that area).

Table 12.3: Summary table of the safe house structure

	Height [m]	Wall thickness [m]	Wall density [g/cm^3]	Weight[kg]
Top Floor	3.5	0.05	2.2	1263
Bottom Floor (rigid)	1.5	0.37	2.15	3105
Bottom floor (water)	1.5	0.33	1.0	1296
Underground segment	1	0.05	2.2	360
End caps	-	0.05	2.2	1860
Airlock	-	-	-	1834
			Total Weight [kg]	9717

Table 12.3 summarises this sections results, the main one being a total mass of 9.3 [t]. This results also shows that making a fully rigid habitat would not be a feasible solution due to the dramatic weight increase over that of the inflatable. The 5 [cm] thickness was also chosen by taking into account possible sublimation of magnesium over time [82].

12.5. Compliance Matrix

Table 12.4: Compliance matrix for the safehouse subsystem

Requirement Code	Requirement Text	Compliance	Method	Justification	Source
SH06	”The habitat shall have a 'safe house' which can be reached within 60 minutes from the habitat”	Yes	Inspection	The safehouse will be placed close enough to the main habitat to be reach rapidly without being too close in case of destructive disasters.	Section 6.2
MLOSYS0601	”The route to the safe house from the habitat shall be easily traversable by astronauts on foot”	Yes	Inspection	The route from the safehouse to the habitat will be on the same level, with a slope ranging between 0 and 5°.	Section 6.2
MLOSYS0601.1	”The route shall be selected on the basis of terrain flatness and lack of harmful materials and conditions”	Yes	Demonstration		Section 6.2
LSSYS0602.1	”The emergency habitat shall contain water, oxygen, and food for 21 days”	Yes	Analysis	The safe house will contain all the necessary life support for 21 days	Section 12.2
STSYS0602.2	”The emergency habitat shall be constructed with an additional layer of structural and radiation safety than the general habitat”	Yes	Testing, Analysis	T The safehouse bottom floor will have the required SPE protection characteristics, while the outer layer of the habitat will be made out of aluminium and polyethylene for adequate solar and galactic cosmic ray protection	Section 12.3

13. Orbital Mission Analysis

With all subsystems designed and sized, and all mass estimates performed, the last obstacle to overcome is to transport all required material from Earth to the Moon. The overall orbital mission will consist of 3 launches in total. Two of these will be done using the Starship for the space company *SpaceX* and will transport all material and machinery required for the habitat (construction) to the Lunar surface. The last launch will utilize the near-future Ariane64 launch vehicle from *Arianespace*, developed by ESA, and will bring the astronauts to the habitat after which the nominal mission life of the habitat will start. In total, approximately 100,000 kg will have to be brought from Earth to the Lunar surface. The transport of this enormous amount of mass is possible due to the Starship design and its ability to refuel in Low Earth Orbit.

First the relevant requirements for the astrodynamic mission design will be given, after which a short description of the launch vehicles of the mission will be given in Section 13.2. Following this, the orbital model will be presented in Section 13.3 together which the ΔV budget of the mission. The loading of the launch vehicles and the specific payload of each launch will then be given in Section 13.4. The total payload mass of each launch, together with the ΔV budget of the mission from Section 13.3 will lead to a final propellant mass budget which is given in Section 13.5. The chapter will end by briefly touching upon the return journey to Earth in Section 13.6 and giving a sustainability analysis of this particular part of the mission in Section 13.7.

13.1. Requirements

In order to present the final design of the launch and orbital segments of the mission, the mission requirements specifically important for this stage of the mission are repeated. At the end of the chapter, a compliance matrix will be given where all requirements are verified. For a more extensive list of requirements and also additional explanations, the reader can consult the Baseline report [38].

- **SH05**: The mission shall incorporate a return operation which can be operated at all times
  - **MLO-SYS05-03**: The method used for leaving the Moon shall be available at all times
    - **MLO-SYS05-03.2**: The return operation shall be able to be initiated regardless of the time, temperature, position of the orbiter, or scientific activities
  - **SLO-SYS05-05**: The lunar lander shall be able to reach 100 km above the Moon surface
    - **SLO-SYS05-05.1**: The lunar lander shall be able to hold 18000 kg of fuel
    - **SLO-SYS05-05.2**: The lunar lander shall have engines with an I_{sp} of at least 220 s
    - **SLO-SYS05-05.3**: The lunar lander shall weigh no more than 21600 kg at departure at Earth
  - **SLO-SYS05-06**: There shall be an orbiter on the Moon capable of returning to Earth with the astronauts and all required material for the length of the mission
- **SH10**: The distance between the landing field and the habitat shall be between 1 and 3 km
  - **MLO-SYS10-01**: The landing site approximation shall be smaller than a circle with a 1 km radius
- **SH11**: The rocket taking the astronauts to the Moon shall carry a small vehicle able to carry said astronauts
  - **SLO-SYS11-01**: The final Lunar lander shall be able to accommodate a total volume of 1100 m^3
  - **SLO-SYS11-02**: The final Lunar lander shall be able to accommodate a total mass of 100000 kg
- **SH14**: The time between the landings of the final component carrying rocket and the astronauts carrying rocket shall be shorter than 6 months
- **SH15**: The time between the first and last landing should be shorter than 24 months
- **SH16**: Regolith dust cloud formation shall be minimised
  - **ENV-SYS16-02**: Landing location shall minimize regolith dust cloud formation
  - **SLO-SYS16-03**: Landing thrusters shall minimize regolith dust cloud formation
- **SH17**: The mission shall be as sustainable as possible
  - **ENV-SYS17-01**: The mission shall be as environmentally sustainable as possible

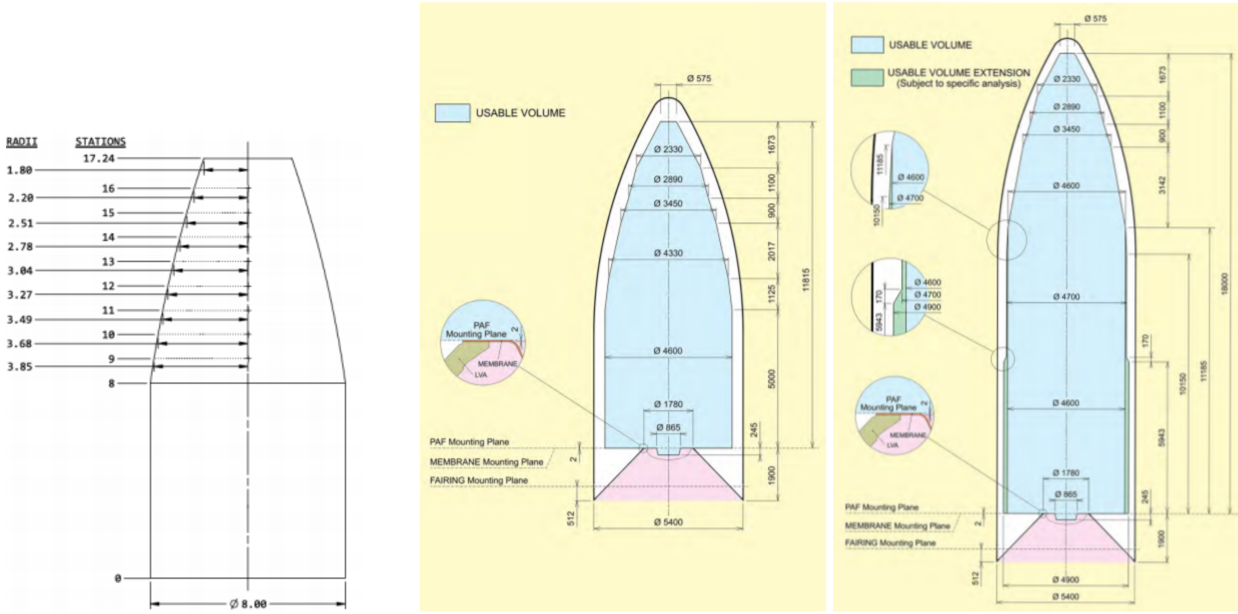
    * **ENV-SYS17-01.5**: The launcher shall use reusable boosters
    * **ENV-SYS17-01.7**: After the end of life of the habitat the material used to construct it shall be either reused, or if not possible, brought back to Earth
    * **ENV-SYS17-01.8**: No parts of the Moon which contain archaeological information about the past shall be used for landing, habitat construction or excavation

- **SH20**: All mission materials, equipment, and supplies shall be delivered to the Moon by the launch and space segment of the mission
  - **SLO-SYS20-01**: The launchers shall deliver the payload to the Lunar surface
    * **SLO-SYS20-01.1**: The launchers shall deliver all Earth materials and tools to build the habitat
    * **SLO-SYS20-01.2**: The launchers shall deliver the scientific research equipment (1000 kg)
    * **SLO-SYS20-01.3**: The launchers shall deliver a Lunar vehicle
    * **SLO-SYS20-01.4**: The launchers shall deliver the astronauts
    * **SLO-SYS20-01.5**: The launchers shall deliver all means for the astronauts to survive for a year on the Moon
  - **SLO-SYS20-02**: The launchers shall deliver a minimum thrust of $1.1 \cdot W_{LV}$ [N]
  - **SLO-SYS20-03**: The launchers shall be capable of reaching LEO orbit
  - **SLO-SYS20-04**: The Lunar orbiter shall have a pointing accuracy of 500 m
  - **SLO-SYS20-05**: The Lunar orbiter shall provide a ΔV of at least 1650 $\mathrm{m\,s^{-1}}$

13.2. Launch Vehicles

An extensive trade-off for the launch vehicle selection of the mission can be found in the Midterm report [39]. In conclusion, the Starship from *SpaceX* will be used as the main transport vehicle due to its large payload capacities and low launch costs, which are needed for the immense scale of the mission. However, as the last launch (Launch 3) transporting the astronauts (see Section 13.4) will only contain the astronauts themselves as well as some additional systems, the Ariane64 from *Arianespace* can be used to transport the astronauts to the Lunar surface instead of Starship. The most important reason for this choice is that sending an almost empty Starship in the last launch is not sustainable nor efficient, and using a launch vehicle developed by ESA alleviates political tensions.

In Figure 13.1 the cargo bay dimensions of both the Starship and Ariane can be seen, which will be used in Section 13.4 to fit all the payload inside the launch vehicle. The Ariane cargo bay has both a small and large payload cargo bay configuration, as seen in Figure 13.1b and Figure 13.1c respectively.



(a) Starship Cargo Compartment Dimensions

(b) Ariane 6 Cargo Compartment Dimensions (Small Configuration)

(c) Ariane 6 Cargo Compartment Dimensions (Large Configuration)

Figure 13.1: Cargo compartment dimensions of both the Starship and Ariane64 [102][3]

The launch vehicle payload capabilities can be seen in Table 13.1.

Table 13.1: Starship and Ariane payload capabilities into Low Earth Orbit [102] [3].

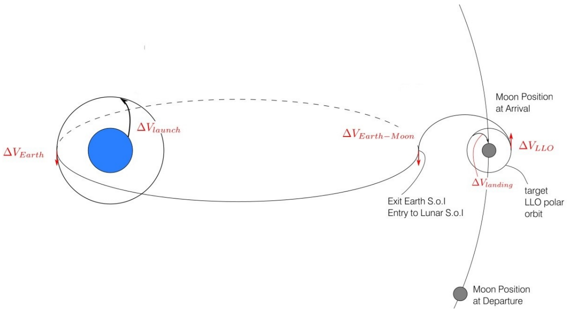
Launch Vehicle Characteristics			
Characteristic	*Starship*	*Ariane64*	
		Small Configuration	*Large Configuration*
Payload Capacity to LEO [kg]	100,000	21,600	
Max. LEO Insertion Altitude [km]	500	Any	
Max. LEO Insertion Inclination [deg]	98.9	Any	
Volume [m3]	1,100	140	250
Max. Payload Height [m]	17	11.8	18
Max. Payload Diameter [m]	8	4.6	4.7

An important distinction to make in the orbital mission analysis is between launch from Earth and transfer to the Moon. The launch from Earth will be the responsibility of the launch vehicle company/agency in question, and so no calculations will be done on the launch segment of the mission to verify that the promised numbers as seen in Table 13.1 are achievable. The payload mass, cargo bay volume and dimensions are important parameters for the logistics of the mission and will be built upon in Section 13.4. Only from the point that the spacecraft injects into Low Earth Orbit at the desired altitude and inclination does the orbital mission analysis of this particular design start. This part of the mission will be the topic of next section.

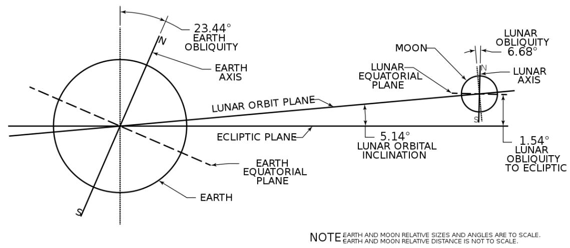
13.3. Orbital Plan

A general overview of the orbital plan of the mission can be seen in Figure 13.2a, starting from launch on Earth and ending with the landing on the Lunar surface. As already mentioned in Section 13.2, the calculations in this chapter will start from parking orbit in LEO, and will not take into account the launch from Earth to LEO.

As the mission considers two launch vehicles, an important distinction has to be made in the mission profiles regarding the two launch vehicles. Namely, the difference between Starship and Ariane, is that Starship is also the Lunar transfer vehicle, bringing the payload to the lunar surface, whereas Ariane will eject 21600 kg into LEO, after which this payload mass is on its own. This means that for the mission profile of the Ariane, at least a preliminary design of a Lunar transfer vehicle should be provided, which will be further touched upon in Section 13.4. The other big difference between Starship and Ariane, is that Starship offers refueling capabilites in LEO, meaning its range and payload capabilities increase drastically.



(a) Schematic overview of the mission orbital plan, showing both the (transfer) orbit dimensions (not to scale) as well as the required manoeuvres.



(b) Orientation of Earth's rotational axis and orbital plane with respect to the Moon's

Figure 13.2: Overview of orbit mission analysis needed for further calculations

Looking at Figure 13.2, the transfer from LEO to the Lunar surface can be split up into 5 segments. From a mission logistics perspective, only the first stage of the mission differs between the Starship and Ariane. Namely, the Starship mission will stay in LEO longer in order to refuel and tank enough propellant to get the maximum of 100 tons of payload to the Lunar surface. For the Ariane mission, this will not be the case, and thus the stay in LEO will be shorter. The exact duration of the parking time in LEO will be given in Section 13.5.

A total overview of the characteristics of the orbital segments as well as the ΔV's of said segments can be found in Table 13.2. A more detailed description of the orbital segments as well as the ΔV's will follow in the upcoming paragraphs. The ΔV of each manoeuvre will be the difference in velocity between two particular points in two subsequent orbital segments. To calculate the velocity in a particular point in both an elliptical orbit and circular orbit, Equation 13.1 and Equation 13.2 are used respectively. For orbital inclination changes Equation 13.3 is used.

$$V = \sqrt{\mu\left(\frac{2}{r} - \frac{1}{a}\right)} \tag{13.1}$$

$$V = \sqrt{\frac{\mu}{R_{body} + h}} \tag{13.2}$$

$$\Delta V_{inclination} = \left\{V_1^2 + V_2^2 - 2V_1V_2cos(\Delta\theta)\right\}^{\frac{1}{2}} \tag{13.3}$$

Where a is the semi-major axis of the orbit in question and is given by Equation 13.4. For a circular orbit this simply becomes the radius of the orbit.

$$a = \frac{r_p + r_a}{2} \tag{13.4}$$

Starting from a LEO altitude of 500 km and orbital inclination of 28.58° (as this is the inclination required to align with the Moon's orbital plane with respect to Earth's equatorial plane: $23.44° + 5.14° = 28.58°$, see Figure 13.2b[1]), the first part of the orbital plan will be the injection into a Lunar Transfer Orbit (LTO) to exit Earth's sphere of influence (SoI). The sphere of influence of the Earth-Moon system is defined as the radius of the sphere around Earth where the gravitational attraction on an object from both the Moon and Earth are equal. So as to leave Earth's gravitational well, the transfer vehicle will inject into a highly elliptical Hohmann transfer orbit to the edge of the SoI, also called *Lagrange point 1* or *EML1* [2]. This elliptical orbit will have an perigee distance equal to the LEO radius and an apogee radius equal to the radius of the Earth's SoI and corresponds to *Mission Segment II* in Table 13.2. The ΔV of this manoeuvre, depicted by ΔV_{Earth} in Figure 13.2a, will be the difference between LEO velocity and the perigee velocity of the previously described elliptical Hohmann transfer orbit.

The second stage of the mission will be capture of the transfer vehicle by the Moon's gravitational force. At this point the gravitational force from the Moon will be larger than that of Earth, and the transfer vehicle will

[1]URL: https://commons.wikimedia.org/wiki/File:Lunar_Orbit_and_Orientation_with_respect_to_the_Ecliptic.svg [cited on June 22, 2021]
[2]URL: https://solarsystem.nasa.gov/resources/754/what-is-a-lagrange-point/ [cited on June 21, 2021]

enter the Moon's SoI. When entering the Moon's SoI, the transfer vehicle is again put into a highly elliptical Hohmann transfer around the Moon, with an *aposelene* distance equal to the radius of the Moon's SoI, and an *periselene* distance equal to the Low Lunar Orbit (LLO) radius. During the same manoeuvre, the transfer vehicle will perform an inclination change of its orbit into a Lunar polar orbit at aposelene to minimize ΔV, which corresponds to an inclination change of 90° with respect to the Moon's equatorial plane (see Figure 13.2b). The Lunar capture ΔV will be the difference in velocity between the apogee velocity of the Earth Hohmann transfer orbit and the aposelene velocity of the Lunar Hohmann transfer orbit. The second ΔV is calculated using Equation 13.3, where $V_1 = V_2$ and equal the aposelene velocity of the Lunar Hohmann transfer orbit, and where $\Delta\theta$ equals 90°. Together these form $\Delta V_{Earth-Moon}$ as seen in Figure 13.2a. The characteristics of this orbital segment and its manoeuvres can be found under *Mission Segment III* in Table 13.2.

The third orbital segment will consist of circulizing the highly elliptical orbit into a polar LLO at an altitude of 100 km above the Lunar surface. For this manoeuvre, the transfer vehicle will do an engine burn at the periselene of the Lunar Hohmann transfer orbit to match the circular velocity of the LLO in question. The ΔV of this manoeuvre, corresponding to ΔV_{LLO} in Figure 13.2a, will thus be the difference in velocity between the periselene velocity of the Lunar Hohmann transfer orbit and the circular velocity of the target LLO. The orbital characteristics for this manoeuvre as well as the ΔV value can be found under *Mission Segment IV* in Table 13.2.

The final orbital segment of the mission will be the landing trajectory from LLO to the Lunar surface near the Shackleton Crater (see Chapter 6 and Section 6.2). Unlike in LEO, the transfer vehicle will only stay in LLO for 1 revolution before commencing the landing segment. To achieve this manoeuvre, the transfer vehicle will have to lose all of its LLO velocity, meaning the ΔV of this manoeuvre will be equal to the circular velocity of the LLO in question at an altitude of 100 km, and corresponds $\Delta V_{landing}$ in Figure 13.2a. The characteristics of this segment and the exact ΔV value can be found under *Mission Segment V* in Table 13.2.

A final note on this segment is that it is probably the worst estimated ΔV of the whole orbital mission analysis. The reason for this is that so called gravity loss has been neglected. What this entails, is that a spacecraft does not only have to lose its circular (or tangential) velocity in order to land, but it should also counter the Moon's gravitational force once the trajectory becomes more vertical. As this requires a more detailed analysis, it was not deemed feasible in this stage of design, as this orbital plan is more so a basis for more detailed design on astrodynamics, rather than a final solution, which is more the case for the design aspects corresponding to Chapter 7 and Chapter 8. However, to compensate for this inaccuracy to a small extent, an additional ΔV of 10% of the total ΔV has been added to the final ΔV of the mission, to account for orbital model inaccuracies as well as smaller orbital correction manoeuvres. The final value of the ΔV budget for the Moon-bound journey can be found in Table 13.2, and will form the starting block for the propellant mass calculations of Section 13.5.

A final, more aesthetic overview of the orbital mission is given by Figure 13.3[3] for the Starship. The figure also shows the refueling in LEO which is a big aspect of the mission. The exact number of refuelling tankers required in LEO will be given in Section 13.5, which is where the propellant mass required for the Lunar transfer will be calculated. However, to determine these values, first the final payload masses of the launches have to be known, which is the focus of the next section.

[3]Figure modified from SpaceX, URL: https://www.spacex.com/human-spaceflight/mars/index.html [Cited June 15, 2021]

Table 13.2: The orbital mission plan giving the parameters of the different orbital sections of the Earth-Moon journey as well as the (final) ΔV budget.

Orbital Mission Plan				
Mission Segment	*Orbital Parameters*	*ΔV [m/s]*	*Duration [min]*	*Comments*
I. Launch into LEO + 1 parking orbit	$h = 500km$ $r_p = r_a = 6871km$ $e = 0$ $i = 28.58°$	-	-	*Responsibility of LV company* *Falls outside mission design*
I. Refueling in LEO		-	95	*Duration per tanker.* *Only applicable to Starship* *Depends on final payload mass and thus propellant mass*
II. Injection into LTO	$r_p = 6871km$ $r_a = 327026km$ $e = 0.959$ $i = 28.58°$	3042	5660	*Brings the spacecraft to the edge of Earth's sphere of influence*
III. Lunar Capture + Inclination Change to Polar Orbit	$r_p = 1837km$ $r_a = 36274km$ $e = 0.9094$ $i = 90°$	271	1967	*Capture by Moon's gravity into an elliptical orbit, together with inclination change into lunar polar orbit*
IV. Circulization into LLO	$h = 100km$ $r_p = r_a = 1837.4km$ $e = 0$ $i = 90°$	620	118	*One parking orbit in LLO before landing*
V. Landing on Lunar surface	$h = 0km$ $89.54°S$ $0°E$	1633	180	*Landing at Shackleton ridge*
Total ΔV				5566 m/s
Additional Correctional Maneuvres (10%)				557 m/s
Final ΔV				***6123 m/s***

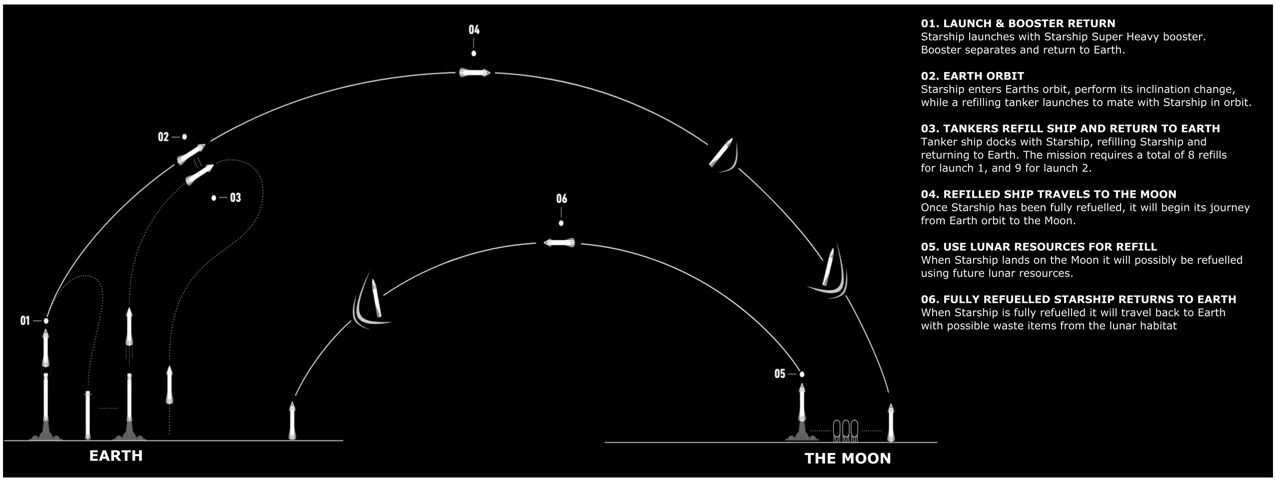


Figure 13.3: Travelling from Earth to the Moon using a Starship spacecraft, Super Heavy rocket and tankers

13.4. Astrodynamics Logistics

In order to bring all the material to the Moon with the minimum amount of launches possible, payload arrangement logistics are extremely important. The logistics behind the payload bay are directly linked with the on-site construction logistics since those will determine what should be brought in each launch for the on-site manufacturing to run smoothly. The steps of the constructions have been discussed in Chapter 8.

For this mission a total amount of 95.707 [tons] and 508.12 [m^3] will have to be brought to the Moon in the minimum amount of launches possible while still being able to meet the manufacturing schedule detailed in Figure 8.8. For manufacturing, sustainability and logistics reasons, it has been decided to use 3 launches. The first two, using Starship, will bring the integrity of the cargo needed for manufacturing while the third one will be exclusively used to bring the astronauts and their lunar rover on the lunar surface once the habitat is fully habitable. For lunar logistics reasons, it is essential for 2 cargo launches to be used, this means they won't be fully packed for our mission. The alternatives options on how the extra cargo can be used, such as ride-share, have been discussed in details in Chapter 18.

13.4.1. Payload Arrangement

The Starship spacecraft is capable of bringing a total payload mass of 100 tons to the Moon. It has a payload volume capacity of 1100 m^3 and its payload dimensions can be visualised in Figure 13.1a.[102]

Launch 1

Launch 1 will carry all the necessary material to prepare the solar farm, start digging the lunar surface, prepare the regolith bags and prepare the terrain for the habitat (arriving in launch 2) and will also contain the safe house. All these subsystems will have to be arranged in the most efficient manner in the Starship payload bay for volume efficiency. In Table 13.3, an overview of all the subsystems and safe house together with their total volume and mass included in launch 1 can be visualised. The volume found in this table are the approximated volume used for launch logistics or volume of the containers where needed. The specific volume of each subsystems will be detailed in the upcoming chapters.

Table 13.3: Launch 1 details of Starship's payload

Subsytem	Total mass [kg]	Total volume [m^3]
Solar farm (x10) (including solar panels, sun flower and cables)	10,983	35
Solar farm - habitat cable	700	2.2
Power docking station	200	0.5
Transporter base (x2)	4,680	55.2
Cylinder on top of transporter base (x2)	1,324	153.9
Excavator (x2)	198	3.6
Regolith bags (x1435)	1,849	5.7
Bagging system	700	12
Safe house	12,642	67.1
Fuel cells, electrolysis device, liquefier, vaporizer and tanks (0_2, H_2)	3,644	6.3
Total	**36,920**	**341.5**

A visualisation of how the Starship payload configuration will look like for launch 1 can in Figure 13.4. The dimensions of each subsystems together with their 3 different view can also be visualised. Starship's payload fairing is a clamshell structure, it contains a door capable of opening once the payload is ready to be deployed. When having multiple payload in the launch, a rotating mechanism (since Starship is a cylinder) can be used to deploy a specific payload at a time. Another mechanism also enables the entire payload to deploy at once, if necessary.

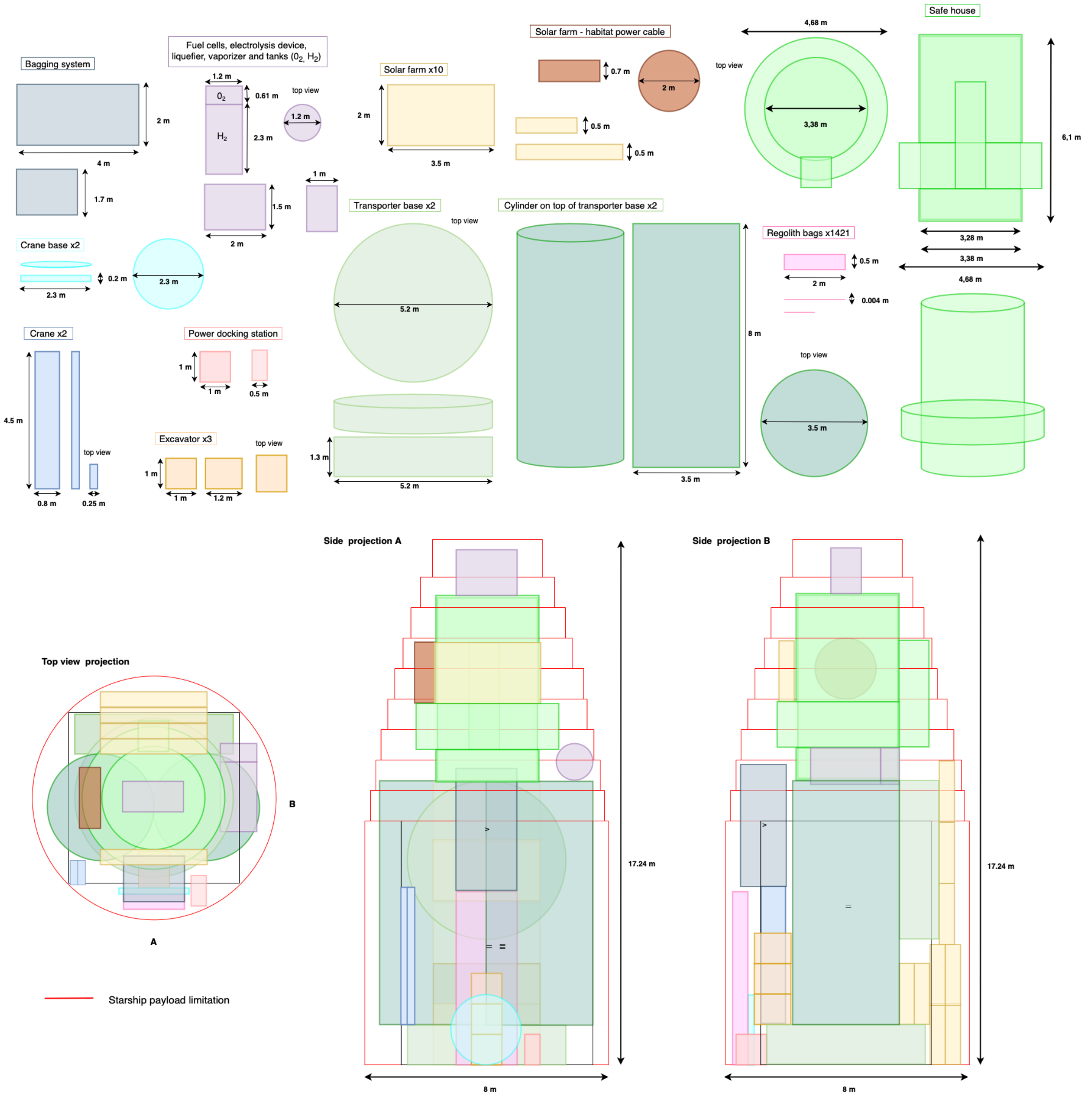


Figure 13.4: Starship payload configuration for launch 1

Launch 2

Launch 2 will carry the main habitat (airlock and inflatable part), the internal transporter needed to bring autonomously the necessary subsystems into the habitat. These subsystems (including astronauts provisions for 1 year) will all be placed into containers (cargo blocks) and will be either stored in their designated place under the habitat floors or placed where they belong inside the habitat. A large number of life support resources and additional cargo blocks will already be placed in the airlock or habitat rigid part during travel in order to gain volume. In Table 13.4, an overview of all the subsystems and habitat together with their total volume and mass included in launch 2 can be visualised. The volume found in this table are the approximated volume used for launch logistics or volume of the containers where needed. The specific volume of each subsystems will be detailed in the upcoming chapters.

Table 13.4: Launch 2 details of Starship's payload

Subsytem	Total mass [kg]	Total volume [m^3]
Airlock + habitat rigid part (including internal subsystems)	42,047	89.5
Cargo blocks (x59)	15,350	68.0
Internal transporter	200	0.4
Total	**57,597**	**157.9**

A visualisation of how the Starship payload configuration will look like for launch 2 can in Figure 13.5. The dimensions of each subsystems together with their 3 different view can also be visualised.

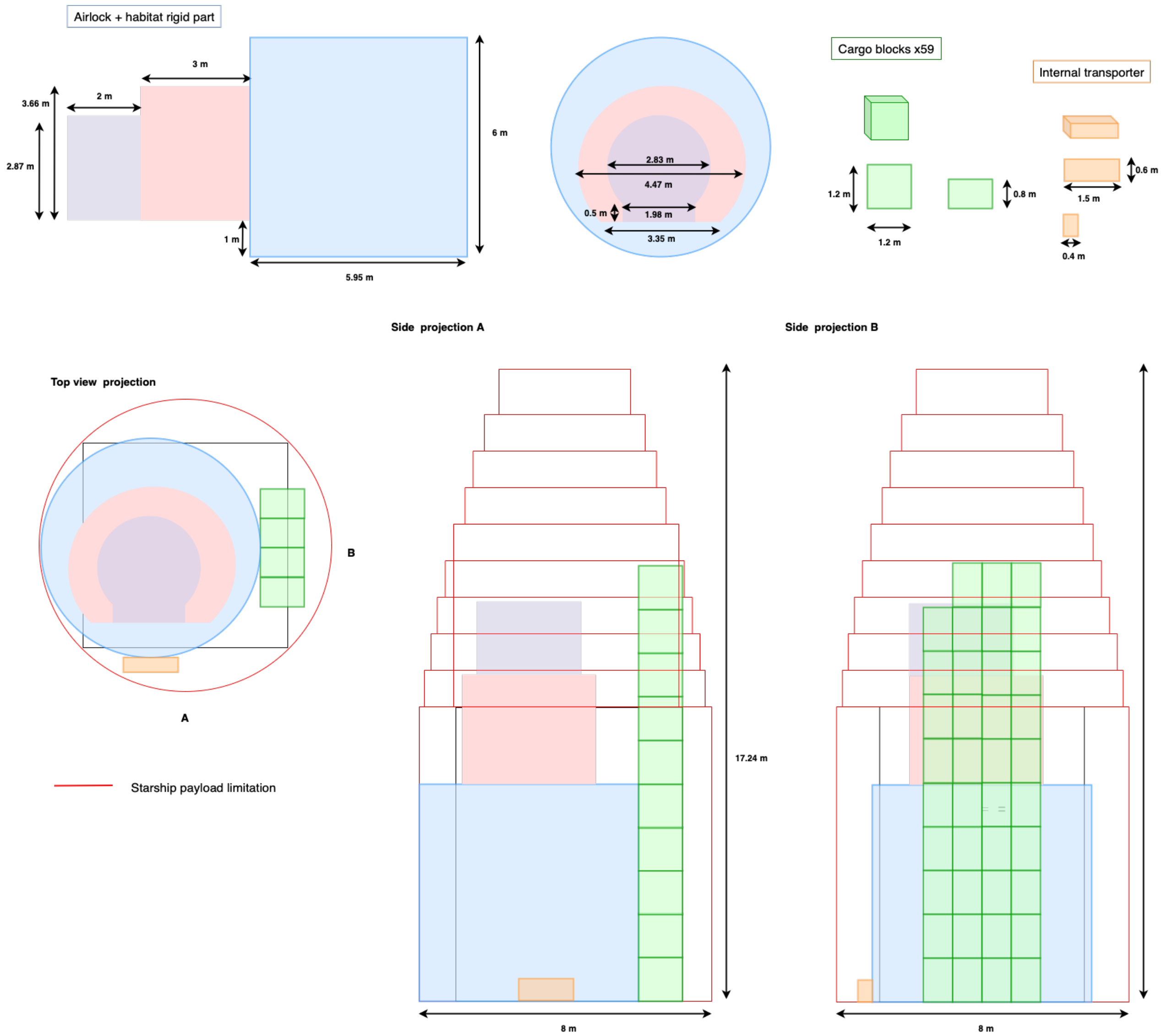


Figure 13.5: Starship payload configuration for launch 2

13.4.2. Manned Spaceflight Arrangement

The next step in the mission, once the habitat has been fully assembled and tested, is to bring the astronauts to the lunar surface safely. The detail of this launch payload configuration can be found in the upcoming subsection.

Launch 3

Launch 3 will land a few weeks after launch 2 and will accommodate the 4 astronauts (wearing their Mark III spacesuits Section 10.5), the provisions they need for their 6 days long journey [including; water (drink and hygiene), food, clothes, O_2, N_2] and their lunar rover. The reason for the discrepancy between the payload mass of Table 13.5 and the maximum payload mass of Ariane64 as given in Table 13.1 is as follows. The payload mass given in Table 13.5 is the useful payload mass that will actually be brought to the Lunar surface, which includes the astronauts, provisions etc. The payload mass of Ariane64 as given in Table 13.1 is the maximum amount of mass the Ariane64 launch vehicle is capable of inserting into LEO. As refueling in LEO for this mission is not possible, said mass will also need to include the structural mass of the Lunar transfer vehicle as well as the propellant mass needed for the transfer as given in Section 13.5. This is why the actual payload mass that will be brought to the Lunar surface is that much lower than the insertion mass into LEO.

Table 13.5: Launch 3 details of Ariane 6's payload

Subsytem	Total mass [kg]	Total volume [m^3]
4 astronauts with their spacesuits	484	4
Provisions for 6 days	496	1.19
Lunar rover	210	3.53
Total	**1,190**	**8.72**

13.4.3. Starship Lunar Landing

The first two launches, transporting all the subsystems necessary to build the lunar habitat, the habitat itself and the safe house, will all go through the same landing procedure. After leaving LLO, the lunar Starship will land vertically on the Moon's surface, on the landing site. The landing site will be located 2km away from the habitat site (requirement **SH10**) and will have a radius of 250m (requirement **MLO-SYS010-01**) as can be visualised in Figure 6.7.

The landing accuracy of Starship can be estimated to be better than Perseverance's since Starship uses the same navigation technology, but on the Moon. Starship can autonomously land with a precision on 100 m on the lunar surface, using Terrain Relative Navigation (TRN) [50]. This accuracy can potentially be increased in future landing (after launch 1) or missions when making use of an antenna on the Moon or more precise future technology.

13.5. Propellant Budget

The final step in completing the orbital mission analysis, is calculating the required propellant mass, both for Starship and Ariane64, to bring the required payload from LEO to the Lunar surface. To determine a rather simplistic initial estimate of the propellant mass required, Tsiolkovsky's rocket equation, as seen in Equation 13.5, will be used for the same reasons as mentioned in Section 13.3.

$$\Delta V = I_{sp} g_0 ln\left(\frac{M_{initial}}{M_{final}}\right) \tag{13.5}$$

To get a propellant mass estimate, an assumption is made that the transfer vehicle will have no propellant mass left after its final manoeuvre. For the Moon bound journey, this means the transfer vehicle will have completely empty tanks once it has landed on the Lunar surface, meaning Equation 13.5 becomes:

$$\Delta V_{landing} = I_{sp} g_0 ln\left(\frac{M_{structures} + M_{payload} + M_{fuel,landing}}{M_{structures} + M_{payload}}\right) \tag{13.6}$$

The ΔV has already been calculated and can be found in Section 13.3. The same is true for the payload mass of each launch, which has been extensively worked out in Section 13.4. All other parameters are parameters depending on the Lunar transfer vehicle that is being used. For the Starship and Ariane, these values are given in Table 13.6, where the structural mass of the Lunar transfer vehicle used in the Ariance mission, was determined using structural coefficient from current and near-future spacecraft [39].

Table 13.6: Transfer vehicle characteristics required for propellant mass calculations

Lunar Transfer Vehicle Overview		
	Starship	*Ariane64*
Payload [kg]	36,920 (Launch 1) 57,597 (Launch 2)	1,190 (Launch 3)
Structural Mass [kg]	130,000	$3M_{PL} = 3,570$
Propellant Capacity [kg]	1200,000	-
Propellant	CH_4/LOX	LH_2/LOX
Vacuum Specific Impulse [s]	356	455

Inputting these values into Equation 13.6 and rewriting, yields a propellant mass required for the landing procedure as seen in Table 13.7. Using this propellant mass, all other propellant masses required for the different manoeuvres of the mission can be determined be working back from Lunar landing to the insertion into LTO from LEO. To give the reader a more clear image of how this works; for the second to last manoeuvre (i.e. the one that comes before Lunar landing), which is circulization into LLO, the propellant mass can be determined using:

$$\Delta V_{circulization} = I_{sp} g_0 ln \left(\frac{M_{structures} + M_{payload} + M_{fuel,landing} + M_{fuel,circulization}}{M_{structures} + M_{payload} + M_{fuel,landing}} \right) \tag{13.7}$$

Doing this for each manoeuvre will yield a linear system of 5 equations and 5 unknowns which can be easily solved. Filling in the specific payload mass for each launch and the specific impulse value corresponding to the transfer vehicle of said launch, yields the propellant mass budget as seen in Table 13.7. For similar reason as those mentioned in Section 13.3, to account for smaller orbital corrections and to compensate for the fact that the calculations used are not the most involved, a contingency of 10% was applied to the total propellant mass, yielding a final propellant mass which can seen as well in Table 13.7. The total mass of the total transfer vehicle, which includes propellant mass, structural mass and payload mass is also mentioned, which is especially important for the Ariane64 mission, as this value should be below 21600 kg.

The final mission aspect to determine, having the final values for the propellant mass required for Lunar transfer, is the amount of times each launch utilizing the Starship, has to be refueled in LEO, and subsequently, calculating the total transfer time from LEO to Lunar surface. The number of refuels needed in LEO is simply calculated by dividing the final propellant mass as seen in Table 13.7 by the maximum payload mass of the tanker, which is the same as a normal Starship and equal to 100000 kg (its payload is comprised fully of propellant).

Knowing the amount of times refueling is required in LEO, one can calculate the total duration of the transfer vehicle stay in LEO by multiplying said number by the orbital period of the Low Earth Orbit given by Equation 13.2, add subsequently adding a minimum of one additional LEO orbit account for the first orbital revolution of the transfer vehicle after the insertion by the launch vehicle. Secondly, the transfer time of both the Earth Hohmann and Moon transfer time are both given by Equation 13.9 by filling in the appropriate values of the orbit dimensions as given in Table 13.2. For the duration in Low Lunar Orbit, the orbital period of one LLO orbit is taken for simplicity. After this one orbital revolution in LLO the transfer will commence its final descent to the Lunar surface which will take approximately 180 minutes, based on [1]. The total transfer time of the Moon-bound journey is given by adding up all previously mentioned orbital periods and can be seen in Table 13.7. The only reason the different launches differ in transfer times is because of the different times spent in LEO for refuelling (Starship missions) or not refuelling at all (Ariane mission).

$$T = 2\pi \sqrt{\frac{a^3}{\mu}} \tag{13.8}$$

$$T_{Hohmann} = \pi \sqrt{\frac{a^3}{\mu}} \tag{13.9}$$

Table 13.7: Mission propellant budget for all three launches going from Earth to the Moon

Mission Segment	***Propellant Mass [kg]***		
	Launch 1	*Launch 2*	*Launch 3*
	LV: SpaceX Starship Payload Type: Material Supplies Payload Mass: 36,920 kg Launch Date: 26/12/2029	LV: SpaceX Starship Payload Type: Material Supplies Payload Mass: 57,597 kg Launch Date: 06/09/2030	LV: Arianegroup Ariane 64 Payload Type: Astronauts Payload Mass: 1,190 kg Launch Date: 06/03/2031
I. Launch into LEO	-	-	-
II. Injection in LTO	478,172	537,405	8,439
III. Lunar Capture and Inclination Change	25,705	28,889	509
IV. Circulization into LLO	51,799	58,216	1,054
V. Landing on Lunar Surface	99,566	111,899	2,167
Total Propellant Mass	655,241	736,409	12,170
Contingency (10%)	65,524	73,641	1,217
Final Propellant Mass	**720,766**	**810,049**	**13,387**
Total Transfer Vehicle Mass	887,686	998,006	18,147
Fuel/Payload [kg/kg]	19.5	14.1	11.2
Number of Refills in LEO	8	9	x
Total Transfer Time	**8596 min 5.969 days**	**8691 min 6.035 days**	**7840 min 5.444 days**

A final thing to mention before closing off the section is the required fuel mass per kg of payload mass as seen in Table 13.7. The fact that this number seems to decrease with increasing payload mass, might seem counter intuitive. However one should remember that not only the payload mass is *dead* mass being brought to the Lunar surface, where *dead* mass means the mass can't produce any momentum for the rocket which is the case for the propellant mass. Namely, the structural mass of the rocket also needs to be transported to the Lunar surface. Because of this, with increasing payload mass, the payload mass fraction ($M_{structure}/M_{payload}$) increases which results in a lower fuel mass per kg of payload according to Tsiolkovsky's equation (Equation 13.5), as this resembles a more efficient rocket. Comparing the fuel mass against the total vehicle mass however for the different launches, results in the same mass fraction for each launch. Concluding, the counter intuitive result is more based on mathematics and the natural logarithm of Tsiolkovsky's equation rather than reality, where propellant mass scales exponentially with payload mass, due to the snowball effect of needing more fuel, thus bigger tanks, thus more structural weight, thus more fuel etc.

The previously described reasoning only holds for the two Starship launches, as the Ariane64 launch only consists of one launch. The reason the fuel mass per kg of payload mass for the crew launch is so much lower, is because of the different propellant used in this mission. Namely, the crewed mission transport vehicle uses LH_2/LOX which has a significantly higher specific impulse than methane, which results in a decrease in propellant mass needed.

13.6. Return Mission

For a return on investment (Chapter 2), as well as sustainability reasons (Section 13.7), it is important to bring both the Starships and all other material back to Earth after the 10 year mission duration as much as possible, provided it is not used in the expansion of the Lunar habitat into a Lunar village. A big problem regarding this mission aspect, is that the Starship will land with empty propellant tanks on the Lunar surface and thus will not have propellant for a return journey to Earth initially. To solve this problem, three possible solution exist of which the last in is the most feasible.

Before discussing the possible options for creating propellant mass for a return mission, it is important to know how much propellant mass will be needed for the return mission to Earth from the Moon. Important to know for the propellant mass calculation for the return journey, is that the ΔV budget will stay the same for the orbital segment. Only the order of the orbital segments will be reversed. This does however matter for the propellant mass budget as now some more energy intensive manoeuvres (like circulisation into LEO) happen at a lower total vehicle mass. To give the reader an impression, Table 13.8 shows the propellant required for a return mission for payload masses of 10, 25, 50, 75 and 100 tons of payload mass for both the standard Starship using LCH_4/LOX and a modified version of Starship, designed especially for Lunar missions, using LH_2/LOX, which will explained further in later paragraphs.

Table 13.8: Mission propellant budget for return missions to Earth using Starship, all values are in [kg] unless specified otherwise.

Starship Earth Return Mission										
Mission Segment	*Standard Starship (CH4/LOX)*					*Modified Lunar Mission Starship (LH2/LOX)*				
	10 t	25 t	50 t	75 t	100 t	10 t	25 t	50 t	75 t	100 t
I. Lunar Launch	251,887	270,544	301,639	332,733	363,828	145,184	154,562	170,191	185,820	201,450
II. Injection into ETO and Inclination Change	86,634	95,916	111,386	126,857	142,327	54,311	60,131	69,829	79,527	89,226
III. Earth SoI Entry	10,683	11,828	13,736	15,644	17,551	6,892	7,630	8,861	10,091	11,322
IV. LEO Circulization	194,611	215,462	250,214	284,966	319,718	136,824	151,484	175,917	200,350	224,783
Total Propellant Mass	543,815	593,750	676,975	760,200	843,424	343,211	373,806	424,797	475,789	526,780
Contingency (10%)	54,381	59,375	67,697	76,020	84,342	34,321	37,381	42,480	47,579	52,678
Final Propellant Mass	***598,196***	***653,125***	***744,672***	***836,220***	***927,767***	***377,532***	***411,187***	***467,277***	***523,368***	***579,458***

The first propellant option would be to fill the remaining Starship payload mass/volume with propellant for a return journey. However, with the payload masses found in Section 13.4 this propellant mass will only be about 50 tons. Furthermore, calculations show that when the Starship is fully refueled in LEO to its maximum capacity of 1200 tons, it is possible to have about 20 tons of fuel left in the tanks when landed on the Lunar surface. This brings the total propellant mass available for a potential return mission to 80 tons, which is not nearly enough when looking at Table 13.8, even for the lowest payload mass missions.

The second option would be to bring tanker Starships to the Moon to refuel the Starships already there. As this Starship would be a tanker, its propellant payload capacity would be 100 tons. With the previoulsy mentioned 20 tons of additional fuel in the propellant tanks, this would bring the total propellant mass brought to the Lunar surface to 120 tons, which, again looking at Table 13.8, is not nearly enough for a return journey. Additionally, this would introduce an additional Starship to the Lunar surface which would have to be returned as well, only making the problem worse.

The final and most feasible option is refueling the Starship on the Lunar surface with ISRU. This would not require any additional fuel to be brought from Earth nor does it introduce additional Starships on the Lunar surface. The only difficulty is that LCH_4/LOX is difficult to produce using Lunar resources. The reason Starship uses methane as its primary fuel, is that it is easily produced from Martian resources, which is the primary mission of Starship [102]. LH_2/LOX is more easily produced from Lunar resources and thus, to make this particular option feasible, a modified version of Starship using LH_2/LOX should be encouraged. The estimated propellant mass for a return mission using this version of Starship can also be found in Table 13.8. For more detailed information on Lunar ISRU for propellant production, the reader can refer to Chapter 2. Initial estimates exist for the required mass of a possible fuel production facility on the Lunar surface, however these are still very preliminary. In any case, a production facility of this sort will not be brought in the launches mentioned in this chapter, but only during later (supply) missions when a better Lunar infrastructure has been set up.

13.6.1. Crew Return

Up until now, only the return of the Starships and (waste) material has been discussed, but returning the astronauts back to Earth safely is significantly more important. The return of the astronauts will occur much earlier on the timescale of the mission, when compared to the return of the Starships etc. Namely, after a 1 year stay on the Lunar surface, the astronauts will return to Earth. Their return will be split up into 3 segments: moving to the safehouse where a crew capsule will be stationed; launch into LLO; and return to Earth.

For the journey of the astronauts from the main habitat to the safehouse, as well as a more detailed design of the safehouse itself, the reader can refer to Chapter 12. The launch of the astronauts will be provided by a high TLR crew capsule like the *Dragon* from SpaceX[4], or the *Orion* from Lockheed Martin and Airbus Defence and Space [5]. These crew capsules contain Launch Abort Systems (LAS), which can eject these capsule far away from the rocket in case of a launch failure and can provide enough thrust to launch from the Lunar surface into LLO. The only design modification needed would be thrust throttling as these LAS often can reach accelerations of 10 g's [6], which is enough acceleration to cause bruises, breaking of bones, and fainting [65]. These high accelerations are needed on Earth as there is imminent danger for the astronauts and LAS is the last resort for the astronauts to escape death. Luckily however, this is not the case when the astronauts launch into LLO, so lower acceleration should be designed for to ensure a comfortable launch for

[4]URL: https://www.spacex.com/vehicles/dragon/ [cited June 29, 2021]
[5]URL: https://www.lockheedmartin.com/en-us/products/orion.html [cited June 29, 2021]
[6]URL: http://www.russianspaceweb.com/soyuz_sas.html [cited June 29, 2021]

the astronauts.

The final mission segment is then to bring the astronauts back to the Earth surface from LLO. Two options exist for this mission segment. Either the crew module docks with the *Lunar Gateway*, which is a project from NASA to put a space station in orbit around the Moon in the near future [7], after which they will enter a tranfer vehicle that is docked to the Lunar Gateway; or an entire new rescue mission will have to be designed which launches from Earth prior to the astronaut launch into LLO, to make sure it is in orbit around the Moon in time to pick up the astronauts. Both are interesting option to be worked out further in more advanced stages of the mission design. As the chapter has focused mostly on the technical aspects of the orbital mission so far, the next session will touch upon the sustainability of the astrodynamics aspect of the mission before closing of the chapter.

13.7. Astrodynamics Sustainability

In general, space missions are inherently environmentally unsustainable. Performing such mission to the Moon is highly polluting in a number of ways. The amount of mass that is brought to the Moon is directly proportional to the amount of propellant needed to bring such payload on the Moon's surface. For this specific reason a number of sustainable mitigation will have to be performed in order for the mission to be the most sustainable possible. Regarding the astrodynamics sustainability, some strict requirements have to be met:

ENVSYS1701.6: "The launcher shall not emit more than TBD kg of harm full gasses into the atmosphere" [38]

ENVSYS1701.5: "The launcher shall use reusable boosters" [38]

ENVSYS1701.7: "After the end of life of the habitat the materials used to construct it shall be either reused, if not possible, brought back to earth" [38]

In order for this mission to be the most sustainable possible, a great emphasis has been put on the booster's sustainability and thus reusability. By using Super Heavy booster, requirement **ENVSYS1701.5** is met; the reusable rocket doesn't discard its first stage but recover and reuse it, which significantly reduce waste link to booster destruction and minimise space debris. Using such booster for sustainability reason has been an evidence since it enables the terrestrial impact to be reduced [106], space debris to be minimised, and importantly to significantly reduce the amount of material and imported components needed for the next launch during the mission.

In order to bring 100 tons at once on the lunar surface, a large number of propellant is needed. By using Starship as a spacecraft, a total number of 8 refuelling Starship for the first launch and 9 for the second, will be needed in order to refill the spacecraft in LEO. These succeeding launches and high number of spacecraft used have a huge negative impact on the environment and mitigation for those should be taken into account. This is the reason why Starship has been used since it is fully reusable[8].

In addition to that, Starship is currently being developed and can potentially use Mars resources to create propellant. It can be assumed that SpaceX will work on a new type of Starship that can make use of lunar natural resources to create propellant and travel back to Earth. Such technology is the future of space exploration and could mean we are a step close to live on the Moon and Mars. Assuming this technology will be available before our mission launch date, requirement **ENVSYS1701.7** can be met since produced propellant on the Moon will enable a potentially large mass (including the astronauts and potential waste) to be brought to Earth; contributing to the Moon sustainability.

Another major impact on the environment is created during launch: the emission of propellant gases. During the launch of a spacecraft, the amount of propellant expelled, and thereby emissions of harmful gases, is extremely large. For this mission, the main solution to decrease this effect would be to minimise the amount of launches during the mission lifetime by choosing an adequate launcher capable of bringing the necessary payload in as few launches as possible [39]. By choosing Starship as a spacecraft, this impact has been mitigated since it can bring a total of 100tons to the Moon, making it the most performant spacecraft ever built.

13.8. Compliance Matrix

[7]URL: https://www.nasa.gov/gateway [cited June 29, 2021]
[8]Starship SpaceX, URL: https://www.spacex.com/vehicles/starship/ [Cited on June 14, 2021]

Table 13.9: Compliance matrix for the launch and travel to the Moon

Code	Requirement Text	Compliance	Method	Justification	Source
SH05	”The mission shall incorporate a return operation which can be operated at all times”	Yes	Inspection		Chapter 12
MLOSYS0503	”The method used for leaving the Moon shall be available at all times”	Yes	Inspection		Chapter 12
MLOSYS0503.2	”The return operation shall be able to be initiated regardless of the time, temperature, position of the orbiter, or scientific activities”	Yes	Inspection		Chapter 12
SLOSYS0505	”The lunar lander shall be able to reach an altitude of 100 km above the Moon surface”	Yes	Inspection	The target LLO polar orbit has an altitude of 100 km	Section 13.3
SLOSYS0505.1	”The lunar lander shall be able to hold 18000 kg of fuel”	Yes	Analysis	The lunar lander corresponding to the Ariane mission needs a total propellant mass of 13400 kg	Section 13.5
SLOSYS0505.2	”The lunar lander shall have engines with an I_{sp} of at least 220 s”	Yes	Inspection	The transfer vehicles ar power by CH_4/LOX and LH_2/LOX which both have an I_{sp} higher than 220s	Section 13.5
SLOSYS0505.3	”The lunar lander shall weigh no more than 21600 kg at departure at Earth”	Yes	Inspection	The transfer vehicle, used when launching with Ariane, will have a total departure mass of 18280 kg	Section 13.5
SLOSYS0506	”There shall be an orbiter on the Moon capable of returning to Earth with the astronauts and all required material for the length of the mission”	Yes		There will be a crew caption ready near the safe house to accomodate for that	Chapter 12
SH10	”The distance between the landing field and the habitat shall be between 1 and 3 km”	Yes		The landing field will be placed 2km away from the habitat site	Section 6.2
MLO-SYS010-01	”The landing site approximation shall be smaller than a circle with a 1 km radius”	Yes		The landing site radius will be 250m, this is achievable thanks to the landing precision we have using TRN	Section 6.2
SH11	”The rocket taking the astronauts to the Moon shall carry a small vehicle able to carry said astronauts”	Yes	Inspection	A lunar rover will be included in launch 3, using Ariane 6. It will be similar in volume and mass than the one used for the Apollo 15 mission	Section 13.4.1
SLO-SYS11-01	”The final Lunar lander shall be able to accommodate a total volume of 1100 m^3”	Yes	Inspection	Starship, which will be used for the payload launches; launch 1 and 2 has a payload volume capacity of 1100 m^3	Section 13.4.2
SLO-SYS11-02	”The final Lunar lander shall be able to accommodate a total mass of 100000 kg”	Yes	Inspection	Starship, which will be used for the payload launches; launch 1 and 2 has a payload mass capacity of 100 tons	Section 13.4.2
SH14	”The time between the landings of the final component carrying rocket and the astronauts carrying rocket shall be shorter than 6 months”	Yes		The time between landing 2 and landing 3 will be 6 months	Section 8.3
SH15	”The time between the first and last landing should be shorter than 24 months”	Yes		The time between landing 1 and landing 3 will be approximately 15 months	Section 8.3
SH16	”Regolith dust cloud formation shall be minimised”	Yes	Testing	While excavating, the machines have been programmed in such a way that dust cloud will be reduced. In addition to that, the excavating site is placed the furthest away possible from the habitat, solar farm and safe house.	Section 6.2
ENV-SYS16-02	”Landing location shall minimise regolith dust cloud formation”			The landing location will be placed the furthest away possible from the habitat, solar farm and safe house.	Section 6.2
SLO-SYS16-03	”Landing thrusters shall minimise regolith dust cloud formation”				
ENV-SYS17-01	”The mission shall be as environmentally sustainable as possible”	Yes	Testing	Even though it is a highly polluting mission a large number of steps have been made to reduce the impact; using reusable boosters and spacecraft, using lunar resources, manage waste, using a maximum of solar power etc...	Section 13.7, Section 9.9, Section 8.5
ENV-SYS17-01.5	”The launcher shall use reusable boosters”	Yes	Testing	Starship uses ’Super Heavy’ rocket which are fully reusable[9]	Section 13.3
ENV-SYS17-01.6	”The launcher shall not emit more than <TBD> kg of harmful gasses into the Earth atmosphere”				
ENV-SYS17-01.7	”After the end of life of the habitat the material used to construct it shall be either reused, or if not possible, brought back to Earth”	Yes		The improved version of Starship is assumed to be able to use lunar resources as propellant, by 2030. Using this technology, used material and waste will be easily brought back to Earth	Section 13.6
ENV-SYS17-01.8	”No parts of the Moon which contain archaeological information about the past shall be used for landing, habitat construction or excavation”				
SLO-SYS20-01	”The launchers shall deliver the payload to the Lunar surface”	Yes	Testing		Section 13.3
SLO-SYS20-01.1	”The launchers shall deliver all Earth materials and tools to build the habitat”	Yes		Launch 1 and launch 2, using Starship, will deliver all the necessary material and machines to build the habitat	Section 13.4
SLO-SYS20-01.2	”The launchers shall deliver the scientific research equipment (1000 kg)”	Yes		Launch 2 will deliver the scientific research equipment, with the main habitat	Section 13.4
SLO-SYS20-01.3	”The launchers shall deliver a Lunar vehicle”	Yes		Launch 3, using Ariane, will bring the astronauts to the Moon together with a lunar rover.	Section 13.4
SLO-SYS20-01.4	”The launchers shall deliver the astronauts”	Yes		Ariane64, in launch 3, will deliver the astronauts to the Moon once the habitat is fully operational	Section 13.4 Section 13.2
SLO-SYS20-01.5	”The launchers shall deliver all means for the astronauts to survive for a year on the Moon”	Yes			
SLO-SYS20-02	”The launchers shall deliver a minimum thrust of 110% of the total launch vehicle launch mass” N”				
SLO-SYS20-03	”The launchers shall be capable of reaching <TBD> orbit”				
SLO-SYS20-04	”The Lunar orbiter shall have a pointing accuracy of less than 500 m”	Yes	Testing	The landing accuracy of Starship will be approximately 100m, since it uses TRN. After launch 1, an antenna can be used, increasing this accuracy for future landing (including when using Ariane64)	Section 13.4.3
SLO-SYS20-05	”The Lunar orbiter shall provide a ΔV of <TBD> $m\,s^{-1}$”				

14. Data and Communications

Communication is essential to the success of this mission. Any moment without communication, could mean the continuation of an off course trajectory or a warning of a solar burst not arriving, potentially causing loss of crew. In this chapter the various roles of the communication subsystem and its requirements are addressed.

14.1. Requirements

To size the communication subsystem the following requirements are considered.

MLO-SYS05-01: The astronauts shall be able to initiate the return operation from every facility within the habitat or within the allowable distance from the habitat

MLO-SYS05-02.1: There shall be more than one method to communicate the need of a return operation to the astronauts

SH07: The astronauts shall be able to communicate with the Earth segment at all times

- **COM-SYS07-01**: The communication link with Earth shall be twice as fast as that of the ISS
- **COM-SYS07-02**: Communications shall extend to the astronauts friends and family at the astronauts request
- **COM-SYS08-06**: The astronauts in the field trip shall be able to keep communication with those on the habitat at all times
  - **COM-SYS08-06.1**: The habitat shall contain a dedicated communications system for communication between astronauts including field trips

Besides communication between the habitat and Earth, there is communication between the subsystems as well. Especially during the construction phase, communication between the robots and Earth is essential, as they are not autonomous, but require human input. In Figure 14.1 the communication between the different subsystems as well as their respective data is presented.

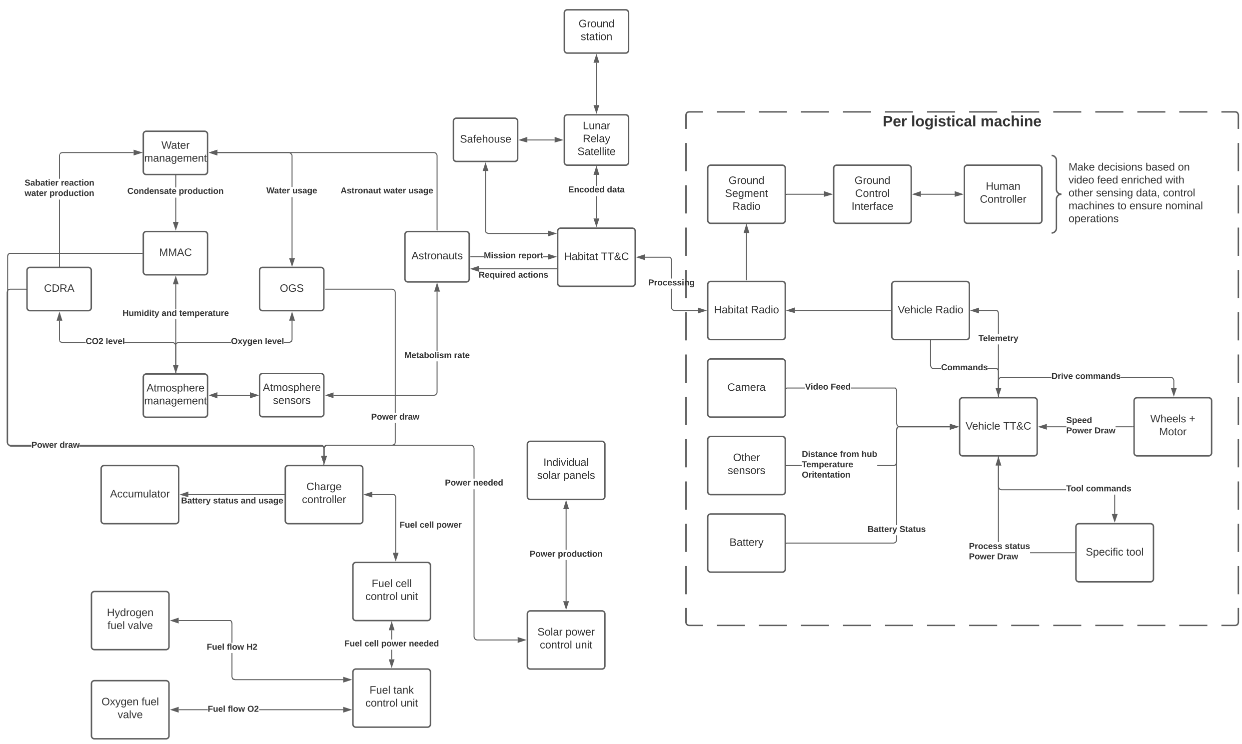


Figure 14.1: Communication flow diagram of the various subsystems and the astronauts.

As can be seen, the communication comes together at the Habitat Telemetry, Tracking and Command (TT&C)

subsystem. Ultimately, the two weakest points in the communication link are the lunar orbiter and the lunar habitat receiver and transmitter. A failure of either one, would mean the loss of mission. And if a rescue operation cannot be achieved, a loss of crew. In emergency's the safehouse will be used. Therefore to mitigate the risk of no communication, the safehouse will have a separate antenna capable of communicating directly with the lunar relay satellite. In addition, the safehouse will also have a direct connection to the habitat, to communicate the sensor data and system activity. Furthermore, astronauts will have connection at all times as well, both inside and outside the habitat during EVA's.

Starting with the connection between the habitat and the satellite, an estimation for the link budget can be made. The two key characteristic of a link budget analysis is the signal to noise ratio (SNR) and the data rate. Calculating the SNR can be done using the equation below, where P is the transmitter power, L_l the transmitter loss, L_r the receiver loss, L_a the path loss, L_s the space loss, L_{pr} the pointing loss, G_t the transmitting antenna gain, G_r the receiving antenna gain , R the required data rate [bits/s] and T_s the system noise temperature.

$$SNR = \frac{P \cdot L_l \cdot G_t \cdot L_a \cdot G_r \cdot L_s \cdot L_{pr} \cdot L_r}{R \cdot k \cdot T_s} \tag{14.1}$$

The main inputs for this link budget estimation are the power and the data rate. As requirement COM-SYS07-01 says, the data rate should be twice that of the ISS. Currently, after recent upgrades, the ISS has a data rate of 600 megabit-per-second (Mbps) [1]. Considering that the ISS has much less space loss with respect to a communication system near the Moon (60 dB roughly), a data rate twice as high seems like a steep requirement. Knowing the required data rate, the power can be determined based on the desired SNR. For clear images a minimum SNR of 8 dB is found, and for error-free communication a SNR of 30 dB [36]. Using an upper limit of 30 dB a transmitting power of 40 W is found. The exact calculations for each parameter in Equation 14.1 will not be given for the sake of conciseness, but the values have been based on the Lunar Reconnaissance Orbiter.

Table 14.1: Downlink link budget between the habitat (transmitter) and the lunar relay satellite.

Parameter	Symbol	Unit	Value
Transmitter power	P	W	40
Transmitter power	P	dBW	16.02
Downlink frequency	f	GHz	8.4
Transmitter antenna gain	G_t	dB	36.30
Transmission path losses	L_a	dB	0 [(a)]
Space loss	L_s	dB	-170.93 [(b)]
Total antenna pointing loss	L_{pr}	dB	-2.04
System noise temperature	T_s	K	135
Required data rate	R	bit/s	$1200 \cdot 10^6$
Received SNR	SNR	-	30.59

References
[a] The Moon has no atmosphere, so no path losses
[b] Based on a circular orbit around the Moon with altitude 1000 km.

A similar link budget analysis was made for the communication between the lunar orbiter and the ground station, but the efficiencies and losses were found to be unreliable. Instead a comparison should be made with existing lunar orbiters, and their communication capabilities.

However, currently the Lunar Reconnaissance Orbiter, only has a data rate of 100 Mpbs [2]. At the moment most of the communication done in space is within radio-frequencies, mainly because they are the most reliable. However, with the increase of complexity and resolution of measurement equipment, the amount of data needed to send back will increase throughout the years. A new promising alternative RF communications is laser communications. NASA has performed the Lunar Laser Communications Demonstration with this technology, and managed to achieve a data rate of 622 Mbps from the Moon to Earth[3]. Considering this was in 2013, it is reasonable to assume breakthroughs have been made to achieve a bit rate of 1200 Mbps. For the communication on the ground, RF communication is still the best choice as it is more reliable.

[1] URL: https://www.nasa.gov/feature/goddard/2019/data-rate-increase-on-the-international-space-station-supports-future-exploration
[2] URL: https://directory.eoportal.org/web/eoportal/satellite-missions/content/-/article/lro2
[3] URL: https://www.nasa.gov/sites/default/files/llcdfactsheet.final_.web_.pdf

However, with only 1 lunar relay satellite, requirement **SH07** (communication with Earth at all times), cannot be fulfilled. Any orbit will always have a period of time in which the line of sight is blocked by the Moon. For communication with ground stations at high latitudes on Earth, the so called Molniya orbits are used. These are highly elliptical orbits, which maximise the amount of time in contact. This can be seen in Figure 14.2, where only 1 out of 12 hours is unavailable for communication. Because the habitat is located on the lunar south pole, similar highly elliptical orbits are suitable for communication. The orbit specifications are summarised in the table below. Roughly 14 hours out of the total 16 are spent in contact with the habitat. So if two satellites are used, an overlap of 2 hours can be created. Not only do multiple connections increase redundancy, but also the bit rate. To fulfill the requirements, at least 2 satellites have to be used, but using more is also possible and provides additional benefits.

Table 14.2: Specifications of the lunar relay satellite orbit for communication to Earth.

Parameter	Symbol	Unit	Value
Semi-major axis	a	km	7400
Eccentricity	e	-	0.75
Perigee altitude	h_p	km	100
Apogee altitude	h_a	km	13060
Period	T	h	16

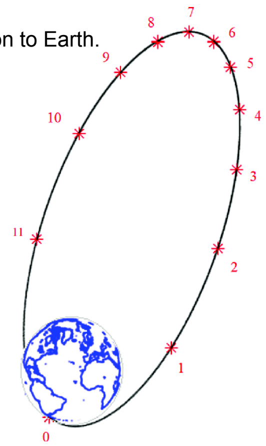


Figure 14.2: Schematic drawing of a Molniya orbit around Earth, in which the majority of the orbital period is spent in contact with the ground station. [31]

For requirement **MLO-SYS05-02.1** (There shall be more than one method to communicate the need of a return operation to the astronauts) multiple connections have to be possible. If the main habitat receiver/transmitter antenna were to fail, a connection can still be made through the direct connection with the safehouse. If the astronauts have to leave the main habitat, the need for a return operation can be communicated directly through the safehouse antenna. And depending on the time of day, 2 satellites might be available for communication. Therefore, if no defects were to occur, there are three ways to communicate the need for a return operation.

14.2. Compliance Matrix

Although not every requirement has been treated, the justification in Table 14.3 elaborates on why at this point in time the requirement complies.

Table 14.3: Compliance matrix for the communications subsystem

Requirement Code	Requirement Text	Compliance	Method	Justification	Source
MLO-SYS05-01	The astronauts shall be able to initiate the return operation from every facility within the habitat or within the allowable distance from the habitat	Yes	Inspection	Every important key point will have access to the main TT&C. During EVA communication is already accounted for.	-
MLO-SYS05-02.1	There shall be more than one method to communicate the need of a return operation to the astronauts	Yes	Inspection	There are three ways: habitat to satellite, safehouse to satellite and habitat to safehouse to satellite	-
COM-SYS07-01	The communication link with Earth shall be twice as fast as that of the ISS	Not yet	Analysis	In the future it is safe to assume it will be met, but not at this point in time.	-
COM-SYS07-02	Communications shall extend to the astronauts friends and family at the astronauts request	Yes	Inspection	Already accounted for in existing space communication systems like in the ISS	-

Part III
Additional Considerations

15. Subsystem Testing

The main facility where ESA tests its hardware is located at ESTEC (European Space Research and Technology Centre) in Noordwijk, Netherlands, and is run by the European Test Services corporation. This facility can perform tests in a variety of fields, including, but not limited to: vibration, shock, acoustic noise, mass properties, thermal vacuum, sun radiation simulation, electromagnetic compatibility, and infrastructure-level tests [1]. These capabilities unfortunately do not cover the full spectrum of required tests for H.O.M.E. systems, but already account for a majority of them.

The tests required for the validation of the H.O.M.E. architecture are driven by the main requirements and constraints placed on its subsystems and the potential failure modes they can experience. Due to the low product series of habitat systems and the high cost of development and manufacture, it is preferable that the system be tested non-destructively if possible (Non Destructive Testing or NDT), and therefore no tests like punctures or heavy impacts should be conducted. The exception for this is components small enough to be replaced in a cost-effective manner (for example, a small sheet of the inner lining of the inflatable could be punctured).

The four kinds of tests below depend on the completion of certain phases of the manufacturing and integration process - information system testing cannot happen before the information systems have been built, for example. Because of this cadence of the availability of tests, they must be completed in a specific order as shown in Figure 15.1.

15.1. Operation Readiness Testing

Operation readiness tests are used to demonstrate that all elements of the ground segment, including the people, the hardware, the software, and the facilities, can accomplish the mission, using a real timeline. Since this is not dependent on the completion of hardware manufacturing and assembly, this kind of test can be performed concurrently with the other types. Two main objectives will be met through operation readiness testing: verification that the ground element can successfully operate an accelerated version of the nominal mission flow, and verification that all crises/contingencies are dealt with successfully by this same ground element.

Concerning the accelerated mission profile, the most crucial phase to ensure the mission control personnel is familiar with is the unmanned, automatic construction of the habitat and supporting infrastructure. Assuming each moving machine will need at least one operator monitoring or outright controlling it at all times, it can be estimated that, in order to ensure adequate human control for a single machine 24 hours a day for sustained time periods, a total of five people are needed overall if weekends, holidays, and sleep are taken into account. This requires operators to be proficient not only in executing but ensuring continuity of control over the hands of several colleagues. The crewed phase of the mission will require less manpower at ground control due to the less intensive or mission-critical use of machinery, and much of the monitoring will fall to life support vitals or pressure levels inside the inflatable, which are easily adjustable algorithmically.

Elsewise, the importance of ground control is greater in the case of emergencies and unplanned outcomes, as they will be needed to minimize losses to infrastructure or potentially human lives.

15.2. Stress Testing and Simulation

Stress testing and simulation assesses the system's robustness to changes in performance and fault conditions. This can be performed at any level of system assembly and complexity, but it is the only sort of test that can be conducted at the component level - additionally, simulations can always be run before any

[1] URL: https://www.european-test-services.net/services-overview.html

physical production has begun. For this reason, it is the immediate feedback mechanism for V&V after the procurement/construction of components, which, if passed, allows them to proceed to the subsystem level.

At this stage, the verification of requirements is completed by testing the components and subsystems to the diversity of nominal and extreme operating conditions the H.O.M.E. architecture will face. Without considering the risky/catastrophic scenarios treated in Chapter 5, the most notable conditions are Earth launch (from a mechanical/structural/vibrational perspective), and unprotected lunar surface operations (from a thermal/radiation/electrostatic/abrasive perspective). Risk contingencies should also be tested for according to the risk mitigation strategies in Chapter 5.

The most important tests to conduct at ETS in order to ensure that these environmental challenges can be met are: vibration, shock, thermal vacuum (for the pressurization of the inflatable), radiation simulation, electromagnetic (regolith dust electrostatic) interactions, and thermal conditions.

15.3. End-to-end Information System Testing

After the components and subsystems have been manufactured, assembled, integrated and validated, the information systems connecting all the different machines can be connected and tested. The main functions and situations that need to be tested for are:

- Radio silence, interruptions in downlink
- Data errors and corruption
- Security of information and hacking prevention through encryption (who knows how sensitive H.O.M.E.'s telemetry data might be)
- Data demodulation and distribution for machine command
- Telemetry and communication standards
- Sensors on all machines and environments
- Control feedback mechanisms (for e.g. ECLSS)

15.4. Mission Scenario Testing

After all of the subsystems have been integrated and their communications and data processing abilities validated, it becomes necessary to test their capacity to carry out tasks foreseen by the production plan. In order to sufficiently model the conditions in which our machinery will be driving, excavating, bagging, and assembling, the lunar surface must be recreated as closely as possible in order to capture the main limitations on the hardware's performance. Some of the critical qualities of our operational environment include a strong vacuum, potentially low-visibility conditions (from the perspective of a human controller), polar-like thermal fluctuations, abrasive regolith covering everything, and a large enough regolith simulant testbed. Hypogravity conditions are difficult to replicate outside of Earth's gravity well, and therefore all forces experienced should be scaled in later analysis to lunar gravity conditions. This will allow all of the principal logistical machines to simulate their own tasks, the most important pertaining to the manipulation of- and maneuvering over regolith.

The inflatable habitat itself will also be tested for nominal operations at this stage alongside the safe house. After the entire assembly process has been recreated here on Earth, the habitat will be subjected to vacuum, lighting, and heating conditions that imitate the chosen ridge near Shackleton crater in order to validate the structural integrity of the Twaron holding the inflatable together, and the regolith's ability to thermally insulate the habitat sufficiently. Another important phase of habitat operations that must be tested is the entire landing and unpacking sequence: its removal from Starship's payload bay with the crane, transport to the habitat site, setup of the solar towers using stored liquid hydrogen, excavation of the inflatable's ditch, and the final unrolling and inflation of the pressure vessel. Getting this phase of construction right is paramount due to the most critical systems all being contained inside of the inflatable.

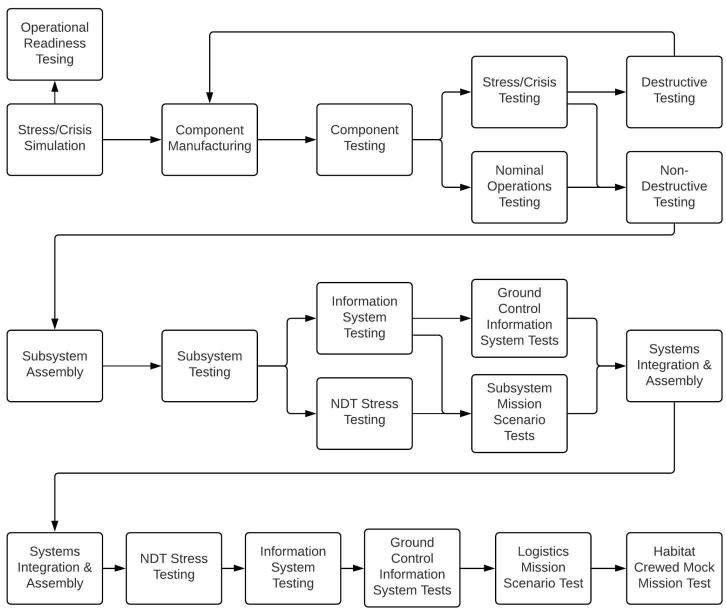


Figure 15.1: Flowchart of the top-level testing procedures to be executed during H.O.M.E.'s manufacturing process

16. RAMS

This chapter uses the results from the detailed design along with the additional considerations issued from the habitat testing chapter to detail the reliability, availability, maintenance activities, and general safety of the mission.

16.1. Reliability

As the first mission of its kind, the astronauts will be put in hazardous situations for which a reliability estimate is necessary. Moreover, identifying the reliability of the mission is necessary to asses the financial risk of the mission. Hence, this section is devoted to quantifying the reliability of the mission and how it is increased. Figures presented in this chapter are computed using mostly conservative estimates, and assume that no improvements in the field take place unless stated otherwise.

The reliability of the mission is defined as the probability that the mission operates successfully during a certain period. In practice, this means ensuring that the requirements whose compliance is necessary for the mission to operate are adhered to. This also means that the reliability of the mission can be estimated by dividing it into various segments, each representing a chronological step in the operation of the mission, and then combining the individual re liabilities found. Since each segment needs to operate successfully for the mission to be considered a success, individual re liabilities are multiplied to obtain a final reliability. These segments can be seen in Figure 16.1.

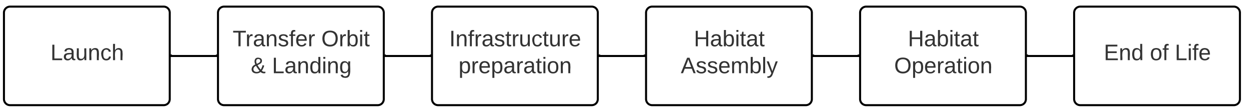


Figure 16.1: Simplified reliability block diagram of the entire mission

For the launch, transfer orbit and landing, reliability cannot be simply extracted from the past. Only a small number of landings have taken place on the lunar surface, which are not comparable to this mission in mass and size. Fortunately NASA has done a study into the probability of mission failure for different scenarios. For the most similar scenario with respect to this mission, the probability of mission failure is found to be 5.5 %, which means a reliability of 0.945.[1]. Given that lunar orbit insertions will become more common as evidenced by ESA, NASA, and CNSA, this figure is likely to increase substantially. The report assumes that the reliability will increase to 0.96 by the time the mission would launch. Launch reliability is estimated using that of the most similar launcher (Falcon 9), which is 0.98[2].

For both the infrastructure preparation and habitat assembly, multiple robots are needed. At this stage the reliability of each robot cannot be estimated. Therefore the reliability of planetary rovers are considered. When multiple robots are used to complete a certain task the estimated reliability becomes 0.995 for both blocks.[3]

For the operation of the habitat to be successful, the following subs systems should all be in operation during the entire length of the mission:

- Life Support
- Structural
- Power storage (batteries and fuel cells)
- Electrical distribution
- Solar array operation
- Communication

[1] URL: https://www.nasa.gov/pdf/140639main_ESAS_08.pdf
[2] URL: https://www.spacelaunchreport.com/log2019.html
[3] URL: https://citeseerx.ist.psu.edu/viewdoc/download?doi=10.1.1.72.8609&rep=rep1&type=pdf

Life support system reliability

Starting with the life support system, the failure rates were modeled using data from the ISS, as it is the closest mission in terms of required technologies to keep astronauts alive. Assuming a constant failure rate, the reliability can be calculated using Equation 16.1. [52] , where λ is the failure rate and t the life span. For the ISS the most optimal predicted failure rate is $3 \cdot 10^{-5}$ (1 hour).

$$R = \text{exp}(-\lambda \cdot t) \tag{16.1}$$

For the duration of one year, the reliability would be 0.77, which is far too low to be acceptable. However, sufficient redundancy and recent improvements in the field can provide the necessary boost in reliability [58][110]. Assuming life-saving elements have one layer of redundancy, and that the systems are only at risk of failing if they are operational, a new reliability can be calculated. The conditions correspond to an exponential distribution, which satisfies the memoryless property.

Since the backup system will only start being used after the first one fails, the entire system will only fail if the sum of the failure times of each system is lower than one. Mathematically, this can be translated to:

$$p(t > T) = 1 - p(t < x) \cdot p(t < y) \tag{16.2}$$

Where $x + y = 1year$. Hence, if x lasts 100 days, then y has to last 265 days (and not 365 days, as simply multiplying the reliabilities would entail). This assumes that the failure events are independent of one another, and that failure doesn't become more likely as the mission goes on. Hence, on average, each life system only needs to be able to last 6 months rather than a year. The probability that they both fail in fewer than 6 months is calculated to be 0.985.

However, a period of 10 years yields a reliability of 0.465, which needs improvement. The latter can be fixed either by bringing a redundant system in the next mission after one fails, or by using improved systems. Using a system with a failure rate of $1.5 \cdot 10^{-5}$ (1 hour) would already increase the 10 year reliability to 0.85. Assuming a combination of those factors takes place, a redundant system is brought when one fails in the previous mission, and the new systems have a lower failure rate after the second year, a 10 year reliability of $0.985^2 \cdot 0.995^8 = 0.93$ is found.

Reliability of other sub-systems

For other subsystems within the operations block, the failure rate is assumed to increase as the mission length increases. A Weibull distribution is used to take into account how the lunar environment affects system failures.

$$R = \text{exp}\left[-\left(\frac{t}{\theta}\right)^{\beta}\right] \tag{16.3}$$

The shape parameter β and the scale parameter $theta$ can be taken from literature [9], and are presented in Table 16.1.

Table 16.1: Shape and scale parameters for the reliability modeling of systems for use in operations using Weibull distributions.

System	Shape parameter β	Scale parameter θ [years]
Structural	0.36	21,308,746
Power Storage	0.75	7,733
Electrical Distribution	0.50	169,272
Solar Array Operation	0.40	19,655,868
Communication	0.39	400982

For the End of life, the astronauts have to return home. Therefore the same launch reliability of 0.98 is used. Using all these values the final mission reliability can be calculated. For both a 1 year period and a ten year period the (non-constant) reliability values for each system can be seen in Figure 16.2.

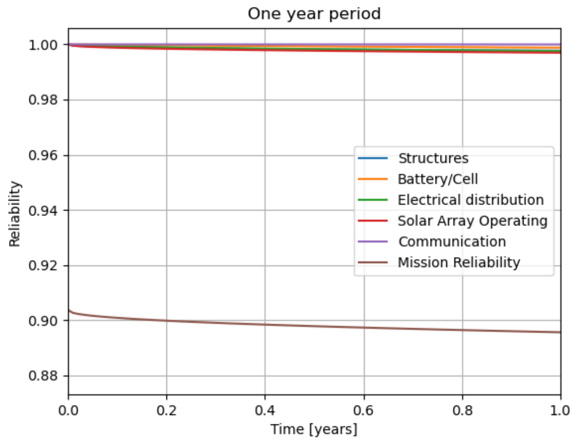


Figure 16.2: Reliability of the systems and the entire mission for a one year period.

The graph shows a minimum reliability of 0.895, which is mainly a factor of launch reliability, the orbital segment, and the life support. Improvements in all of these areas are therefore recommended by the team. In particular, oxygen generation equipment for interplanetary missions was found to be insufficiently reliable[53], and any improvements in that area would greatly benefit mission reliability as a whole. Halfing the failure rate would boost the reliability beyond 0.9, which would be necessary to justify the mission as a whole. The team therefore recommends a full redesign of the equipment currently use in the ISS, and the necessary R&D costs are included in this report. Moreover, a particular focus on life support maintenance would also boost overall reliability, which is why life support systems are further discussed in Section 16.2 and Section 16.3.

The 10 year reliability is of 0.85, with the drop mostly being due to the life support systems. The main recommendation would therefore be to bring redundant life support systems in subsequent missions even if the ones in use have not yet failed. Moreover, bringing extra parts to reduce the failure rate of such systems would also be necessary. Adding another layer of redundancy would already boost the total mission reliability to 0.89, and any improvements in launcher or orbit insertion due to added experience in the field would easily increase it beyond 0.92.

All aspects considered, this mission would be the first of its kind and reliability estimates based on previous missions will likely overestimate rates of failure. The scale of the mission justifies additional manpower to be placed on ensuring that launches, orbit insertions, and maintenance are all performed with extreme care.

16.2. Availability

Availability can be defined as the ability of an item to perform according to its purpose at a given time. This is obviously related to the system's reliability, as a system's failure implies that it is unavailable. This type of availability is the *inherent availability*, and can be calculated using Equation 16.4.

$$A_i = \frac{MTBF}{MTBF + MTTR} \tag{16.4}$$

Assuming that the failure rate is constant, the Mean Time Between Failures is found with:

$$MTBF = \frac{1}{\lambda} \tag{16.5}$$

Since Life Support systems are the most prone to failure, and that their failure may result in mission failure if proper maintenance isn't issued, it is their availability that is treated.

Using the previously discussed failure rate of life support systems a MTBF of 1,389 days is found. Considering that the repair of life support systems may take 2 days this gives an inherent availability of $A_i = 0.998$. When considering the maintenance activities the operational availability is a useful metric to consider. Instead of the MTBF and MTTR, the Mean Time Between Maintenance and the Mean Time To Maintain are considered.

$$A_o = \frac{MTBM}{MTBM + MTTM} \tag{16.6}$$

For life support systems, as explained in Section 16.3, maintenance is scheduled each day taking an hour on average. This gives an operational availability of 0.96. This figure is likely to increase with improvements in life support system design field, but is still elevated enough to be considered sufficient,

16.3. Maintainability

Maintainability is the ability of activities capable of ensuring that an item performs its function to take place. This can be divided into subsystem accessibility, maintenance actions required, duration of maintenance actions required, and frequency of these actions.

For accessibility, the main factor is whether the subsystem is within the inflatable, as any activity outside would require cumbersome spacesuits, and would render all other aspects of the activity much more complicated, and therefore also riskier. This is one of the main reason why life support systems were chosen to be fully included within the boundaries of the inflatable. However, manufacturing equipment, power systems, communications systems, and the safe house all have external components, and therefore have significant detrimental effects on the maintenance of the habitat as a whole. Table 16.2 showcases how decreased accessibility leads to a reduction in frequency of maintenance activities.

Table 16.2: Maintainability analysis of habitat subsystems taking into account both scheduled and unscheduled maintenance activities.

Subsystem	*Inside Habitat?*	*Accessibility*	*Scheduled Maintenance*			*Unscheduled Maintenance*
			Maintenance Action	*Frequency*	*Duration [hrs]*	*Duration [days]*
Life Support						
Oxygen Supply	Yes	Easy	Tank Inspection Contaminant Detection (methane from Sabatier process)	Daily	1	2
CO2 Scrubbing	Yes	Easy	Tank Inspection Concentration Analysis Filter Scrubbing/Replacement	Daily	1	2
Water Supply	Yes	Easy	Biological Contaminant Prevention	Daily	1	2
Food Supply	Yes	Easy	Biological Contaminant Prevention	Daily	1	2
Pressure Control	Yes	Moderate	Compressor Cleaning/Checkup	Weekly	2	2
Waste	Partly	Moderate	Leakage Detection and Cleanup Disposal	Daily	1	2
Humidity Control	Yes	Easy	Sensor Checkup Seal Inspection	Daily	1	2
Contaminant Control	Yes	Hard	Filter Scrubbing/Replacing Seal Inspection Viral/Micro-organism Detection	Daily	1	2
Thermal Control	Partly	Moderate	Isolation Repairs Sensor Checkup	Daily	1	2
Power System						
Photovoltaics	No	Hard	Dust Cleaning Electrical Connections Mechanism Check	Weekly	10	5
Fuel Cells	Yes	Moderate	Membrame Replacement	Weekly	4	5
Fuel Storage	No	Hard	Tank Inspection Pipeline Inspection	Weekly	4	5
Power Distribution and Management	Partly	Moderate	Fuse Replacement Thermal Management	Twice a week	2	2
Manufacturing Equipement						
Excavators	No	Not possible	Wear Detection Dust Cleaning Part Replacement	Once	-	28
Crane	No	Not possible	Wear Detection Dust Cleaning Part Replacement	Once	-	28
Transport	No	Not possible	Wear Detection Dust Cleaning Part Replacement	Once	-	28
Task Specific	No	Not possible	Wear Detection Dust Cleaning Part Replacement	Once	-	28
Safehouse	No	Hard	Life Support Checklist Structural Integrity Checklist Radiation Shielding Checklist Lunar Launcher Readiness	Twice a week	6	As short as possible
Communications	Partly	Moderate	Ground Control Check Dish Cleaning	Weekly	8	4
Structure and Shielding	Partly	Hard	Crack/Lining Damage Inspection Bladder functioning tests Restraint Layer Jamming Inspection Regolith Bag Stability Inspection Airlock Attachment Inspection Other Joint Inspections	Weekly	6	14

Frequency and maintenance duration scores were attributed through literature reviews as well as engineering judgement, and are subject to change in more advanced design stages [52]. While individual maintenance actions are likely inaccurate, their orders of magnitude are correct enough so that the combination of all maintenance activities is a good representation of the time and effort required for the ensemble of the habitat.

Most outlined maintenance activities are observation based, but are necessary to ensure that damages are found early. This however does raise the challenge of ensuring that the astronauts receive proper training and equipment to properly identify structural damages.

As it stands, each astronaut would need to perform 3.7 hours of maintenance daily to comply with the matrix, which is only a 10% increase over what is expected in large space infrastructure such as the ISS [94]. Proper training and equipment can employed to reduce the duration of various maintenance tasks, especially for the structures and life support systems. Moreover, the team recommends that ground control also participates in the monitoring of subsystems and their behaviour such that maintenance needs diminish throughout the length of the mission. Halving the life support maintenance time leads to a reduction of one hour per day per astronaut, which in turn gives the settlers more time and flexibility to deal with unscheduled maintenance.

16.4. Safety

Reliability, availability, and maintenance have the primary goal to secure that the astronauts are safe throughout the mission. However, some safety measures can be taken outside of that realm. Measures can be taken to improve the prospects of the astronauts through practiced responses and monitoring of the habitat. The following measures are recommended to be implemented:

- Maintaining a pressurized cabin environment
  - Pressure Monitoring and Control (ground and Moon station)
  - In-Orbit Leak Isolation and Repair Hardware
  - Leak Response Drills
- Providing a habitable atmosphere: contamination of the vehicle, uncontrolled microbial growth in the water or air, a fire, or failures in the systems that control the levels of CO2 or the generation/delivery of O2 and nitrogen (N2)
  - Air Quality monitoring (ground and Moon station)
  - Contaminant Release or Toxic Spill, preferably in a controlled manner
  - Microbial Overgrowth control
  - On-board Fire Drills
  - Life Support
- Sufficient consumable availability for long missions and periods longer than the planned missions in case of unplanned delays
- Providing ground support through the Mission Control Centers and otherwise ensuring astronauts are well trained and equipped with documentation regarding protocols in case of emergencies/contact loss with Ground Control
- Ensuring astronaut safety even in the event of structural collapse through EVA suit redundancies such as consumable gases, pressurization, power, and communications. These are the main things astronauts need and can survive a few days on
- Cross-checking sensors and detection systems with handheld versions of the same technology for regular inspections of the safety systems, allowing for malfunctions in anomaly detection to be discovered in time
- Bringing along enough spare parts to ensure any failures over the long mission durations do not go un-maintained. Contrast this with a high-accessibility station like the ISS, which receives spare parts every few months. This rate of re-supply cannot be achieved, and must thus be superceded by large resource redundancies (spare parts, tools, etc.)

17. Verification and Validation

In this chapter, the verification and validation of the requirements, the code, and the system are discussed.

17.1. Requirement Verification and Compliance

The verification and validation of requirements was performed on a chapter-by-chapter basis, since the Compliance Matrix at the end of each chapter inherently indicates whether or not requirements have been met (verified) and whether it is even possible to comply with them (validated). Additionally, the method of verification is included for each requirement, as is the reason for invalidating it in case it is not V.A.L.I.D. (Verifiable, Achievable, Logical, Integral, and Definitive).

17.2. Model Verification and Validation

Several different models were used in the sizing of subsystems and calculation of parameters, most of them based off of theoretical results from previous studies, our current knowledge on technologies that would require some adaptation for use in H.O.M.E., and the input of several industry experts.

Due to the highly non-linear nature of the relationships between the different models used as seen in Table 17.2, a centralized data management system was implemented to ensure that all inputs and outputs were stored in a single .csv file, and accessed equally by each script written. This attempted to eliminated the risk of faulty data transfer or human error in communication, and served as a kind of preemptive system code verification.

17.2.1. Code Verification

Thankfully, much of the calculations inside the individual models did not surpass simple arithmetic operations (often performed over arrays) in complexity, allowing for unit tests to be written simply using known solutions for chosen inputs. These unit tests were performed at the function level and tested for known values as well as singularities, boundary values, and other potential special cases.

17.2.2. Code Validation

The code validation process was similar to a loop since several models and equations used were based on the same sources that were then used for validation of results. This means that errors inherent in the sources would propagate throughout our models undetected, but due to a lack of resources, empirical experience with many of the subsystems being designed, and a lack of personal experience, these auto-referential validation sources were used. The usual procedure is to use the source's equations with values relevant to H.O.M.E., and check whether its output matches ours. The relation between our models and their origin can be seen in Table 17.1.

Table 17.1: Sources of validation comparisons for each model used

Code Block / Model	Validation Source	Notes
Robot Scaling	[12] [32] [42] [64] [77] [78] [1]	Sources used on performance of the machines
Fuel and fuel tank sizing	[61]	Sources used for sizing reference
Safehouse mass estimate	[62] [111]	Sources for thicknesses
Regolith mass calculations	[8] [10] [82]	Sources for bag sizes and stacking
Thermal flow calculations	[55]	Sources used for internal heat gain calculations and heat produced by astronauts

17.3. Sensitivity Analysis

To check the robustness of the main calculated system outputs, namely payload mass and volume, a sensitivity analysis was conducted. This measures the effect of changes in key inputs on output calculations, and allows one to estimate for what range of input values the resulting design complies with requirements. Another added benefit of performing such an analysis is to address the uncertainty inherent in the design values found at a

[1] https://www.nasa.gov/pdf/390539main_Athlete%20Fact%20Sheet.pdf

Table 17.2: N2 Chart of the inputs and outputs of different calculation blocks/models used for the design of H.O.M.E.

Landing Site/ Habitat location	Incident radiation	Temperature gradient; incident thermal energy	Target requirements, orbital profile delta v, trajectory, landing accuracy			Distance from rocket	Available regolith mass at site	Sunshine amount and direction, maximum allocated area for power, T° gradient						Direct contact time	
	Radiation	radiation heating effects			Regolith thickness, cosmic radiation layer										
		Thermal Flow			Regolith thickness, thermal layer			Power draw		Temperatures of certain habitat parts					
			Astro-dynamics												
		Air temperature, hydrogen fuel tank temperature control		**Life Support**	Tank sizes for the gases/liquids, volume of the system, weight of the system	Total Weight		Power Draw	Power draw for each system; hydrogen, oxygen, and water generation	Atmosphere values, Contaminant concentrations, power draw data,	System costs, R&D costs	Payload mass	Volume and dimensions of each system.		
		Temperature losses	S/C characteristics: Weight	Oxygen losses, thermal requirements	**Structures**	Total Weight Regolith mass and volume	Regolith mass, regolith volume				Structure costs, R&D costs	Payload mass	Support structures		
			S/C characteristics: Weight			**Logistics**		Power Draw		Location of robots, robot telemetry and control, ground control	Logistics system costs, R&D costs	Payload mass			
			S/C characteristics		Defects	Total time taken, number of robots required	**Manufacturing**	Power Draw		Location of robots, robot telemetry and control, ground control	Manufacturing system costs, R&D costs	Payload mass	Machine/vehicle storage		
		Heat generation from fuel cell	S/C characteristics, weight			Total weight; power components (Fuel tank, fuel cell, Electrolysis device)	Solar panel (size)	**Power**	Wiring	Power draw	EPS system costs, R&D costs	Payload mass, Output power, Payload volume	Power storage		
						Total Weight			**Electrical**			Payload mass			
								Power draw		**Data**				Data amount to be sent	
											Costs				
			Total launch masses									**System Characteristics**			
					habitat radius	Logistics of laying out the internal equipment, Total Weight	floor depth			Experiment Data requirements			**Internal Layout**	Connection placement	
						Weight, Satellite deployment		Power Draw	Wiring				Antenna placement	**Communications**	
		Advice on modelling thermal conduction			Feedback on inflatable load calculations	Robot mass, driving speed, action speed, efficiency, charging time	Inflatable manufacturing feedback								**Literature/ Experts**

preliminary stage. Reported uncertainties at these initial phases of design hover between 10% and 20%, and considering the amount of purely theoretical, conflicting, potentially outdated, scarce, or unclear data that was found in literature concerning H.O.M.E.'s subsystems, it is likely that the assumptions and parameters our team designed around are in a 40% uncertainty range (i.e. 20% above and below design values).

Monte Carlo sampling, the method chosen for this analysis, turns each risky/uncertain variable into a probability distribution and samples from it many times in order to produce an output space that corresponds to likely real-life scenarios. The total outputs of the system are then calculated with this input space, yielding a likely probability distribution of real-life scenarios for total payload mass and volume. The sampled variables and respective probability distributions' properties can be found in Appendix B, and the results of the Monte Carlo sampling for mass and volume are shown in Figures 17.1 and 17.2.

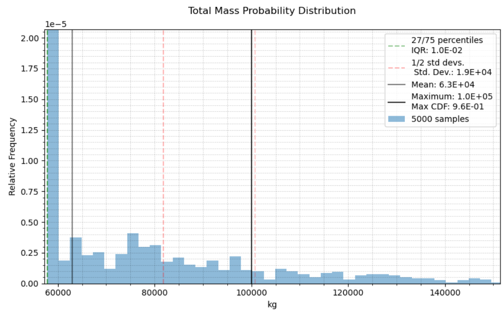


Figure 17.1: Payload mass distribution for habitat construction and the first crewed mission

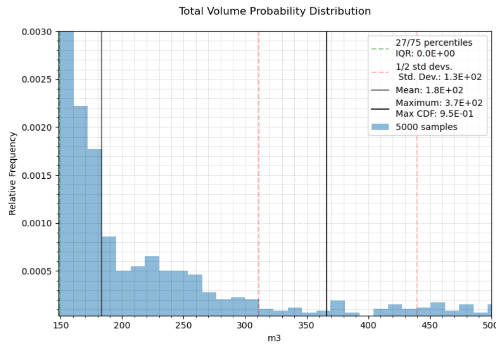


Figure 17.2: Payload volume distribution for habitat construction and the first crewed mission

Both distributions had significantly more likely values near the bottom end of their range while quickly tapering off and showing low counts of very high values, some of which are not displayed in the histograms for clarity. This is a surprising result considering the probability distributions for each variable were all normal, and is likely an artifact of the dynamics of how inputs and outputs affect each other in the aggregate model developed by the authoral team.

A limitation of the Monte Carlo approach chosen lies in its reliance on an internally cohesive model to calculate outputs from by using sampled inputs. In this specific case, many of the models/blocks of code mentioned in Table 17.1 were either not written as code initially (and thus had to be ported to Python code imperfectly) or were not well integrated with the rest of the models through their inputs and outputs. In other words, the actually used model did not reflect all of the connections displayed in Table 17.2, meaning that the incomplete propagation of results might have limited the impact of the variations in inputs on the outputs. Since some of the variables being sampled were masses already (these values were intermediate results from earlier calculations, such as the sum of all internal/layout masses, for example), it is possible that the sum of their lowest values may have created a lower bound for the total mass estimation. Potentially, some uncaught bug or error in other more integrated models may be causing their resulting masses to go to zero frequently, leading to a comparatively more common payload mass near the lower bound of the histogram. The same process likely occurred for payload volume.

More work and time on this analysis should yield much more precise demonstrations of risk and the propagation of uncertainties, but as a first level estimation, the above results allow for the tenuous conclusion that the actual mass and volume of the entire payload will likely fall below the estimates documented elsewhere in this report. This is demonstrated by the cumulative probability below a mass of 100 tons and volume below 366 cubic meters (this value was chosen for being one third of the volume of Starship's payload bay, 1100 m^3) being 96% and 95%, respectively.

Further work in the sensitivity analysis should also take the total system power draw for construction and habitat operations and the time taken for construction into consideration as outputs. This should also integrate directly with the resulting cost of the mission.

18. Return on Investment

In this chapter, the results of the Market Analysis in Chapter 2 (i.e. that the only market worth capitalizing on is lunar ISRU propellant production) will be combined with the total estimated mission costs in order to estimate the return on investment that ESA will have with H.O.M.E. between 2030-2040.

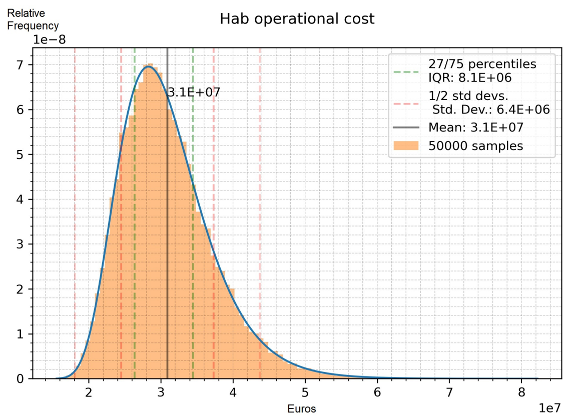
18.1. Cost Breakdown Analysis

The expenditure sources for H.O.M.E. are mainly the R&D cost of novel technologies that need to be developed and integrated, and the launch costs of the initial payload and resupply missions. Other costs, like the unit (manufacturing) costs and operational costs, factor in as well, but do not make as significant an impact as R&D and launch. Note that the budgetary impact of launch costs is heavily dependent on launch prices from Earth to the Moon, and on the lower end of the range of possible values for this parameter, launch costs make a comparatively small difference in the total price of a lunar habitat.

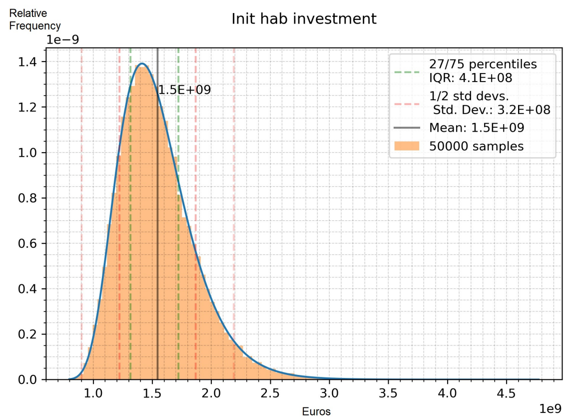
Developing a subsystem-level estimate of costs proved impossible due to the sparse information on analogue technologies (on which our design was based) and the sensitive nature of financial disclosure for high technology. However, mission-level (including all costs) estimates were found in an analysis published in 2016 [18] based on Paul Spudis' work on lunar infrastructure. The estimate for the total development of the habitat and surrounding system was estimated to be 1 billion euros based on Spudis' analysis, and the corresponding yearly operational costs, 200 million euros. It is likely that these were optimistic for the current state of wide technological gaps waiting to be filled, doubly so considering the accelerated timeframe in which H.O.M.E.'s R&D and manufacturing is planned to happen in.

Therefore, the costs of this mission were given progressively smaller likelihoods of running much higher than lower, as can be seen in Figures 18.1a and 18.1b. This is an attempt to capture the all-too-likely scenario of budget overruns and unexpected expenses. These two distributions are log-normal with a shape factor of 0.3, and starting points corresponding to the respective estimated value mentioned earlier.

The ratio between these three costs (R&D, manufacturing, and yearly operational) was assumed to be 1:3:15 i.e. unit costs were one third of development costs, and yearly operational costs were one fifth of unit costs. This is a general heuristic which applies to advanced technologies with low product series. [18] Yearly resupply mass for systems e.g. spare parts (excluding consumables for astronauts) were assumed to be five percent of the initial system mass.



(a) Probability distribution for operational costs



(b) Probability distributions for initial R&D and manufacturing investment

Figure 18.1: Probability distributions for H.O.M.E.'s operational costs and the initial R&D and manufacturing investment

18.2. Profitability Analysis

With all of the estimated mission costs accounted for, the business case of a lunar propellant production facility attached to the main H.O.M.E. habitat can be analysed.

18.2.1. Overview of the Model

The developed model is a simple calculation of the mission's Net Present Value, which is a measure of total future returns on an investment which takes into account that future money has attached risk and has been diluted through inflation, and is therefore worth less than money in the present. This is calculated using a yearly discount rate which captures the combined effects of inflation and operational risk, represented in Equation 18.1 as i, whereas the net cash flow of some period in the future is represented as R_t. [100]

$$NPV = \sum_{t=1}^{n} \frac{R_t}{(1+i)^t} - \text{Investment}_{\text{Initial}} \tag{18.1}$$

Every projected year in the model sums up the revenues from the sale of lunar propellant at several locations in cislunar space, the mission's operational costs and resupply launch costs, as well as the cost savings from refuelling the resupply rockets on the lunar surface. The year's revenue is multiplied by the equity stake in the propellant operation and the projected market share. This yields the year's net cash flow, and this value is then discounted according to how far it is in the future (starting from the first year of operations) as per Equation 18.1. Summing up the NPV of all years and subtracting the initial investment made in the habitat and propellant facility yields the final NPV of the entire project. Results were calculated separately for the use of aerobraking when reaching LEO since the effect of this one change is significant on the system's outcomes.

18.2.2. Assumptions

There are numerous assumptions being made in this model, some qualitative and some quantitative.

Qualitatively, the market for propellant in cislunar space is assumed to extend only to LEO, EML1 (Earth-Moon Lagrangian Point 1, as explained in Chapter 2), and the lunar surface. This is because accurate figures for demand projections could not be found for GEO and LLO, but also because the customers of propellant are well understood at the three considered destinations. A great deal of propellant will be needed at LEO to raise satellites into GEO, perform inclination changes, service operational satellites, and generally access the rest of cislunar space at a fraction of the price that would be paid by launching straight from Earth. EML1 is the ideal place for a propellant depot with equal access to most cislunar nodes, but also serves as an excellent staging point for future interplanetary missions, and projections for demand at this Lagrangian point captures these two demand drivers. Propellant will be needed on the lunar surface to resupply any missions which touch down, greatly reducing the mass one needs to land with in the first place (this also applies to H.O.M.E.'s resupply flights). [63]

Concerning the production and sale of propellant, H.O.M.E.'s profit is made from the difference between the price of Earth-based propellant (EBP) and that of Moon-based propellant (MBP). The assumption is made that undercutting the price of EBP at any cislunar location by at least 25% provides consumers enough savings to secure a significant share of the market for MBP. This undercut value takes operational risks into account e.g. a launch failure and the propellant never reaching the customer. The larger the difference between this discounted price (EBP minus 25%) and the the cost of producing and sending MBP is the profit made per cislunar location.

Table 18.1: Main assumptions made for the profitability model of propellant sale [101] [63] [18]

Earth Launch Decrease Rate	5%	**Moon Launch Decrease Rate**	5%
EBP Production Cost	1 $/kg	**MBP Production Cost**	500 USD/kg
EBP Prod. Cost Decrease Rate	0%	**MBP Prod. Cost Decrease Rate**	5%
Mine Percent Equity (init.)	100%	**Avg. Discount Rate**	25%
Initial Market Share	100%	**Market Share Decrease Rate**	5%
Plant Production Efficiency	25 kg/yr / kg	**Plant Development Cost**	50k USD/kg
Plant Operational Costs	3k USD/kg / yr	**Effect of aerobraking on LEO**	2-3x cost reduction

Additionally, it is assumed that launch costs from Earth and Moon, the cost of producing lunar propellant, and the demand for propellant in cislunar space, have fixed rates of yearly change i.e. any one of these parameters is assumed to exponentially increase or decrease consistently (with a certain yearly change rate) over the 10 years of operations. Partial ownership of the mine by ESA (implying construction and operations to be contracted out possibly) is also taken into account, and the reduction in market share due to the rise in competition from other lunar operations and asteroid mining operations in also factored in. These are all shown in Table 18.1 alongside the main assumptions of the propellant plant's main parameters concerning costs and outputs. The use of aerobraking to bleed off excess speed when returning to LEO from the Moon is also considered in Table 18.3 to reflect the decrease in fuel needed to transport one unit of propellant.

For the three propellant markets taken into account, the estimates for the coming decades shown in Table 18.2 were generated by ULA.

Table 18.2: Propellant and water projections in cislunar space for the next few decades [63]

Propellant & Life Support Water per Year (MT)	2010	2025	2040	2055	2070
LEO Depot	2	433	2385	3096	23320
EML1 Depot	0	425	3133	5534	43158
Moon Surface Depot	0	13	482	771	4665

Calculating the CAGR between the 2025 and 2040 projections yielded both the annual rates of growth and the assumed initial values in 2030, shown below in Table 18.3, which also shows the location-specific cost of EBP and MBP due to the amount of propellant units needed to deliver one at that location as well as other costs associated with launch like refurbishment and operations (assuming all vehicles used are considerably reusable). Note that the values for the lunar surface do not include the propellant needed for H.O.M.E. resupply flight refueling - this is modelled as a cost saving, not revenue, but is taken into account when sizing the lunar water ice mine.

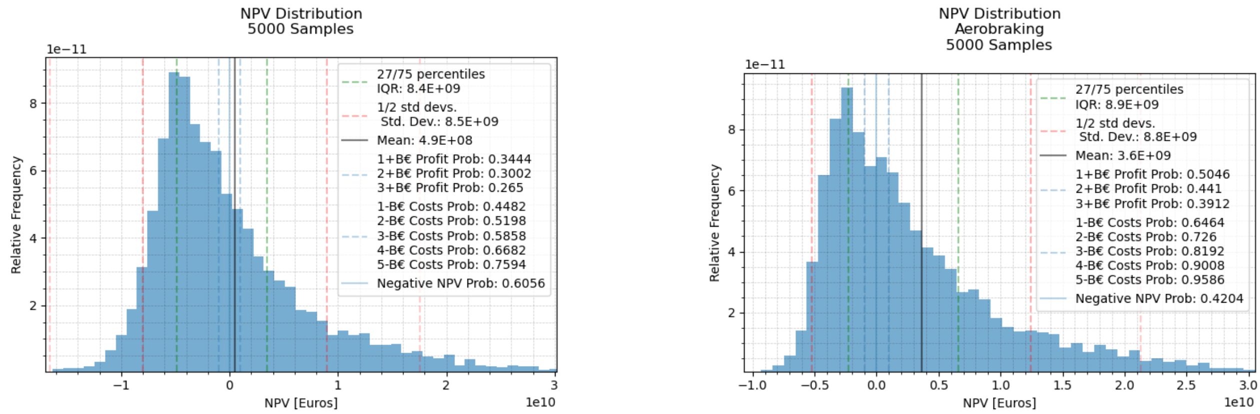
Table 18.3: Demand and cost projections for the main cislunar propellant markets over the mission time frame [63]

Location	Initial Propellant Demand	Yearly Demand Growth Rate	Propellant Cost from Earth	Propellant Cost from Moon
LEO	510	0.12	4000	1000-3000
EML1	552	0.14	12000	1000
LS	44	0.27	36000	500

18.2.3. Risk Modelling with Monte Carlo

There is a considerable amount of uncertainty surrounding the estimated values reported in the previous sections (e.g. the demand for propellant at any cislunar location), both for the costs and also for general assumptions made concerning the cislunar propellant market. In order to take this uncertainty and other sources of risk into account, these variables will be turned into probability distributions which model the likely values they will take, and produced at random from these distributions. For example, costs cannot be negative, and so cannot be modelled using a normal distribution - a log-normal distribution skewed positively more accurately captures the likely outcomes for costs, which will likely hover around the estimate, and potentially be much higher than expected. A similar logic was used in estimating the distributions for other parameters in the attempt to capture the risk inherent in each. A table with the properties of the probability distributions used can be found in Appendix B.

18.2.4. Net Present Value Calculation Results



(a) NPV distribution of project without the use of LEO aerobraking

(b) NPV distribution of project with the use of LEO aerobraking

Figure 18.2: Distributions of potential NPV's for the H.O.M.E. mission and propellant production

Calculating the total mission NPV using randomly sampled parameters several thousand times allows for the generation of a probability distribution for the NPV itself, as shown in Figures 18.2a and 18.2b, which covers all possible scenarios. This process of randomly sampling variables in order to capture their risk is called Monte Carlo sampling.

As can be seen from Figure 18.2, the NPV for both cases approaches a log-normal distribution. Table 18.4 shows the aerobraking scenarios's probabilities of having a certain net loss *at most* or *at least* a certain level of profit i.e. the right-tail cumulative probability. ESA's budget for 2020 was 6.68 billion euros, 645 million of which was allocated to Human and Robotic Exploration (HRE). Assuming for this analysis that this budget does not increase in terms of real purchasing power i.e. only tracks inflation, the total loss can be amortized over the 10 years of planned operations (paid in equal parts every year) and expressed as a percentage of both the total and HRE budgets, also shown in Table 18.4.

Table 18.4: Budgeting scenarios and right-tail likelihoods (for aerobraking case) for maximum expenditure/minimum profit cases and the respective needed percentage of ESA's budget

Scenario	Probability [%]	Yearly Amortized Cost	% of Budget	% of HRE Budget
+3B Net	39.12	NA	NA	NA
+2B Net	44.1	NA	NA	NA
+1B Net	50.46	NA	NA	NA
Break Even	57.96	0	0	0
-1B Net	61.82	100 million €	1.497	15.5
-2B Net	70.28	200 million €	2.994	31
-3B Net	80.06	300 million €	4.449	46.5
-4B Net	89.1	400 million €	5.988	62.0
-5B Net	95.26	500 million €	7.485	77.5

It is evident that the brunt of these expenditures would occur in the first year, as the costs of habitat R&D, launch, and construction are assumed to coincide in the first year of operations, as shown in Figure 18.3. Every year after the first actually has an average positive NPV in both scenarios, leading us to the possibility of ESA taking a loan to cover the high initial capital expenditure and paying it back over the next 10 years, or undergoing a special funding round separate from its usual budget in order to pay for the setup costs of the habitat and propellant mine. This analysis does not take the several ESA funding cycles that must be applied for every 3 years, which further distributes the total costs (specially R&D) over an even longer timeframe.

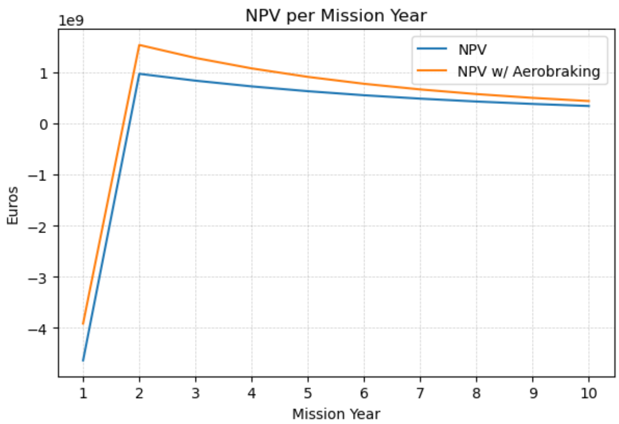


Figure 18.3: Averaged Net Present Value of each mission year with and without aerobraking

While no outcome can be predicted to occur with absolute certainty, the probabilities reported in Table 18.4 point to a feasible and realistic budgetary increase or funding round. There is an almost 95% likelihood of ESA not spending more than 500 million euros a year for H.O.M.E.'s 10-year lifetime, and an almost 40% chance of profiting at least 3 billion euros in total. The case can be made that the proposed architecture is not potentially quite profitable, but also reasonably payable in the case that it does not turn a profit.

19. Conclusion

The Moon is the key to unlocking the immense potential of space development. Setting an outpost on the lunar service not only allows for its resources to be exploited, it also establishes an important central node for many future space activities. Throughout the report, the risks associated with establishing such an outpost were analysed, and it was concluded that the mission is indeed possible and profitable. The main challenges to tackle were those of radiation shielding, assembling a structure autonomously, generating sufficient power for both the robots and the habitat, establishing reliable life support systems, establishing a reliable communication link, and designing a safehouse.

The radiation shielding aspect was solved through the use of a hybrid system that uses in-situ res sources as well as inflatable layers to protect from different types of radiation. The habitat also tackled other hazards of the lunar environment by incorporating additional material layers into the structure. The habitat is an inflatable structure, which required a specific deployment mechanism that was also designed here.

The calculations done on the inflatable structure confirmed that the habitat can handle the loads that will be present during assembly and normal operations phase. The thicknesses of inflatable layers were designed so that the habitat is well insulated and minimise oxygen loss to the lunar environment.

The habitat itself, along with the systems required for it to function, all have to be assembled by a set of robots. The logistics of the mission along with the required robots were all established. The functioning and purpose of the robots is also explained, and the assembly was designed to be finalised with ample time.

The power systems had the main challenge of requiring upright solar panels that turn to face the Sun and don’t obstruct other systems. The challenge was solved by optimising solar panel spacing and cabling, and by using a hybrid fuel cell system for recharging the robots.

The life support systems could all be incorporated into the habitat itself, where they would be continuously maintained and inspected. The systems are capable of reliably providing all the necessary resources to the astronauts so that they can remain healthy and perform all the necessary work.

The safehouse was designed to be radically different from the main habitat, so that a failure of one would most likely not correlate to the failure of the other. It was designed to have all the necessary systems to keep the astronauts healthy during emergency situations, which the current design ensures.

The mission as a whole was also concluded to be profitable in its current state, and one that opens the door to a plethora of new industries. The H.O.M.E project shows that the construction of the first lunar outpost is feasible and lucrative. The technology to make it achievable exists for the most part, but some aspects that were found lacking or in need for further study are presented in Chapter 20.

20. Recommendations

This chapter describes the aspects that could be further improved with a new iteration of the design. While the requirements were met, additional time and resources could help the design reach a higher level of optimisation.

20.1. Project Design and Development Logic

In order to actualize all of the concepts and architectures proposed in this report for the intended customer, ESA, this group's preliminary design will have to pass through all the phases that the space agency's real space exploration missions had to overcome in order to be funded and implemented. Since ESA's execution of programmes is limited by the financial and political capital it can rally from its member nations, and constrained by the decision-making cycles that inevitably inject politics into scientific affairs, it is important for any stakeholder attempting to promote a specific mission to learn the rules of the system and surf the waves of bureaucracy.

All ESA missions go through several implementation phases over their lifetimes [1], as can be seen in Table 20.1. It must also survive successive rounds of funding allocation and maintain political momentum throughout its lifetime. These rounds follow the 3-year funding allocation cycle at the end of which all the member nations convene at a plenary session and allocate funding with renewed perspectives, national interests, and available data. From the date of publication of this report, June 2021, the next plenary session will be in 2022, followed by more in 2025, 2028, 2031, 2034, and 2037. These are the sessions of interest for the H.O.M.E. project because their cycles cover the time frame between the present and eventual decommissioning or re-purposing of the habitat infrastructure in 2040.

Table 20.1: Phases of ESA missions

Phase 0	Mission analysis and identification
Phase A	Feasibility
Phase B	Preliminary Definition
Phase C	Detailed Definition
Phase D	Qualification and Production
Phase E	Utilisation
Phase F	Disposal

It is important to keep in mind that at every plenary, there will be varying levels of support for different initiatives from different nations e.g. Luxembourg is heavily interested in the utilization of Space Resources and might be a stronger supporter of H.O.M.E. than France or Germany [2].

[1]URL: https://www.esa.int/Science_Exploration/Space_Science/How_a_mission_is_chosen

[2]URL: https://space-agency.public.lu/en/space-resources/the-initiative.html

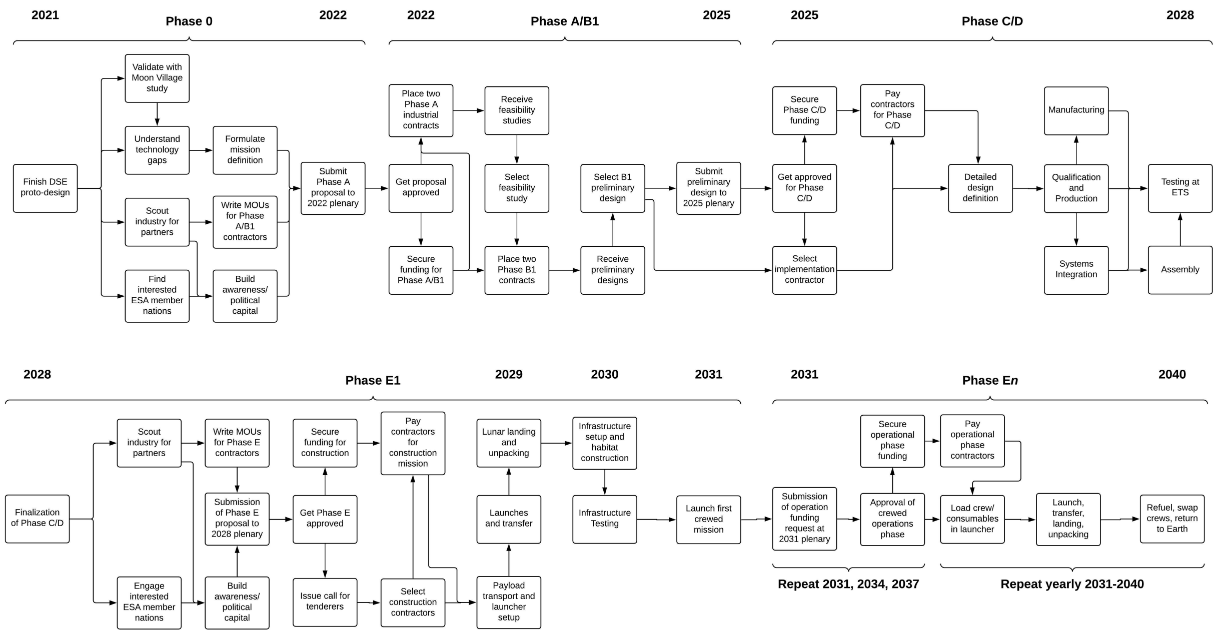


Figure 20.1: Flowchart of post-DSE activities to take H.O.M.E. from a student concept to a real ESA programme

It is also important to keep industry players aware of one's planned proposals so they can prepare their tenderer offers i.e. their offers to execute the contract for the execution of the proposal in question. Phases A/B, C/D and E each have competitions for contractors, and high-quality offers can only be ensured through proper information of the potential commercial partners. These two dynamics imply that every plenary meeting requires some networking with both nations and corporations in order to cement one's interests and increase the likelihood of receiving funding and contractor offers, a step which is reflected in Figure 20.1.

The first step is to format this report's content to ESA standards for a Phase 0 proposal, and deliver it in time for the 2022 plenary in order for it to be considered as a graduate to the A/B1 preliminary design phase as can be seen in Figure 20.1. Since the planned beginning of crewed operations for H.O.M.E. is around the year 2030 and construction begins two years before then, this leaves 6 years for all of the detailed design of the architecture, production and integration of systems, testing, and assembly in the launch vehicles - not to mention the closure of the various technological gaps which prevent certain aspects of the proposal to be executed at present. The aforementioned preparation and research for the construction for the required systems will happen mainly in phase C/D. This is because Phase A/B1 is predominantly about preliminary design, and might not include the budget needed for R&D. Each of these phases, A/B1 and C/D, correspond to one ESA funding cycle.

Once all the systems have been designed, built, tested and packed into payload bays, construction and assembly of the habitat can begin in Phase E1 around 2028 after funding is secured for another 3 years. Once these activities are complete around the year 2030, the first crewed mission can be sent up, and the program enters the En Phase where yearly missions to swap crews and resupply consumables are sent to the lunar South Pole. Funding for this last phase is secured in 2031, 2034, and 2037.

20.2. Power

Future design iterations, or missions coming after Exodus, could incorporate an oxygen generation system that use lunar soil [82]. This could lead to up to 4 tonnes of savings, as less oxygen would need to be carried from Earth, and would also facilitate the expansion of the habitat. This step would need to be incorporated at some point if a lunar colony were to become self-sufficient.

20.3. Policy Recommendations

It is clear from the scope and consequences of this proposal that ESA's decision to implement it or not will greatly affect it, its geopolitical context, and space exploration in general. Both the costs, financial and technological, but also the rewards, are significant for H.O.M.E. Building crewed lunar infrastructure must align with ESA's goals and needs while also being technologically feasible within the timeline presented in Section 20.1.

20.3.1. Space Resources Strategy

As stated the 2019 document titled "ESA Space Resources Strategy" [13], the agency's main goals by the end of the 2020's are demonstrating and maturing lunar ISRU in order to enable and cheapen further in-space operations. The desired outcomes by 2030, as listed in the report, are (emphasis mine):

- Identification and characterisation of at least one non-polar deposit. Identification and characterisation of at least **one deposit of polar ice**.
- All technologies critical to the production of oxygen, and material bi-products, from lunar regolith or pyroclastic material matured to TRL6 or above including flight demonstration for all technologies that require it.
- New terrestrial technologies and processes derived from space resources activities.
- End to end demonstration of the **production of water or oxygen at the lunar surface** from locally sourced materials.
- Space resources integration planned for in reference international exploration architecture with early demonstrations defined.

Some of these goals align perfectly with the assumed starting point of the H.O.M.E. architecture, specially concerning lunar propellant production and how this enables the financial scenario envisioned in Chapter 18: having lunar propellant production capabilities available as soon as possible after H.O.M.E. operations begin. ESA has a clear understanding of the importance of ISRU as the key to accelerate the pace of cislunar development, but in order to fully reap the benefits of lunar resources, many supporting technologies will have to be matured. As can be seen in Figure 20.2, ISRU is not an end, but a means to enable a sustained crewed presence in space.

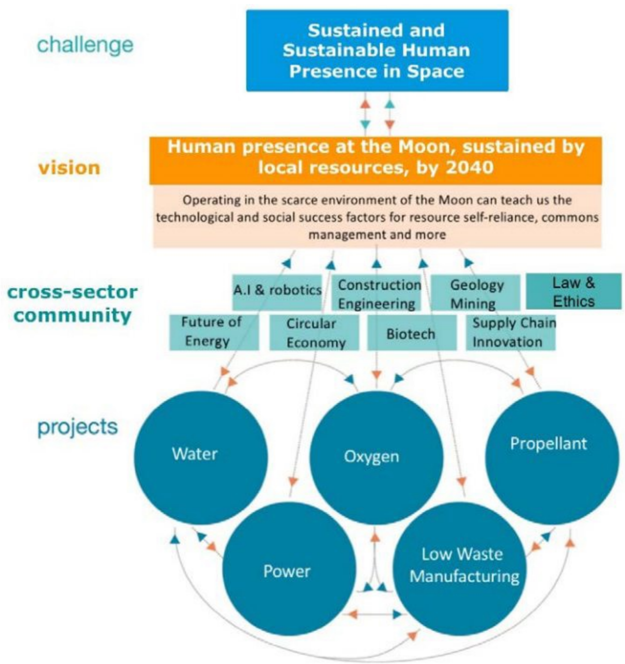


Figure 20.2: Flowchart of ESA's integrated industrial and mission-oriented strategy of investment and development [13]

However, ESA is not alone in its endeavors. Until the recent past, the role of ESA in international space missions like the ISS or scientific orbiters sent to other planets has been a supporting one for NASA - usually, around 10% of the payloads/systems was supplied by ESA, which is in proportion to the difference in budgets between NASA and ESA (i.e. ESA's budget is 10% of NASA's in size). Recently, however, ESA has been vying for larger budgets from its member states, more programs and initiatives, and a larger degree of participation in ventures with the USA with eventual space capability independence in mind. This means, for example, to provide mission critical subsystems or to have a majority of delivered payloads be delivered by Europe.

As mentioned in Chapter 2, Europe will have to invest aggressively in R&D to ensure it does not get left behind in the coming governmental and commercial expansion in to cislunar space. Figure 20.3 shows the origin of the majority of cislunar companies around the world in 2020: it is clear that the USA dominates in all fields, and if ESA does not expand its budget for cislunar exploration to match American levels, it will forever be a secondary actor in the cislunar theatre, as it is today.

H.O.M.E. represents the flagship program that can drive all of the needed investment and the creation of more European space industry to fulfill the demand for in-space services.

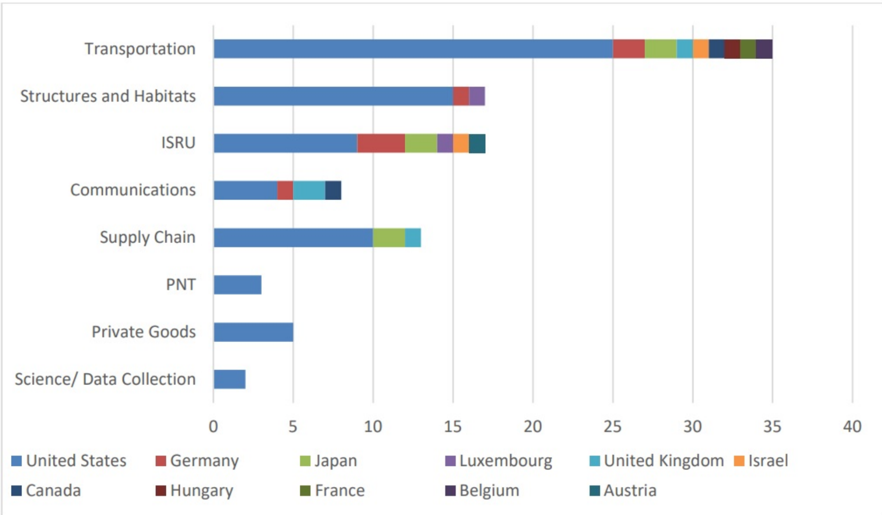


Figure 20.3: Survey results of all 89 current lunar and cislunar companies in 2020 [18]

ESA is also a member in the International Space Exploration Coordination Group (ISECG), a consortium of national space agencies which currently boasts 26 member agencies from around the world. Even though it is one of the big players within the Group budgetarily, ESA has nothing to lose from coordinating its strategy with the rest of the world. The ISECG released a report in 2020 [45] specifically outlining its goals for human lunar surface exploration, the phases of development, and the necessary research to be done in order to progressively expand our capability on the Moon. Figure 20.4 shows some of their conclusions concerning the technological gaps that need to be closed - the fact that this is the space agencies' consensus is indicative of the relative difficulty the authoral team had in designing some of these very same subsystems, some of which do not even exist yet.

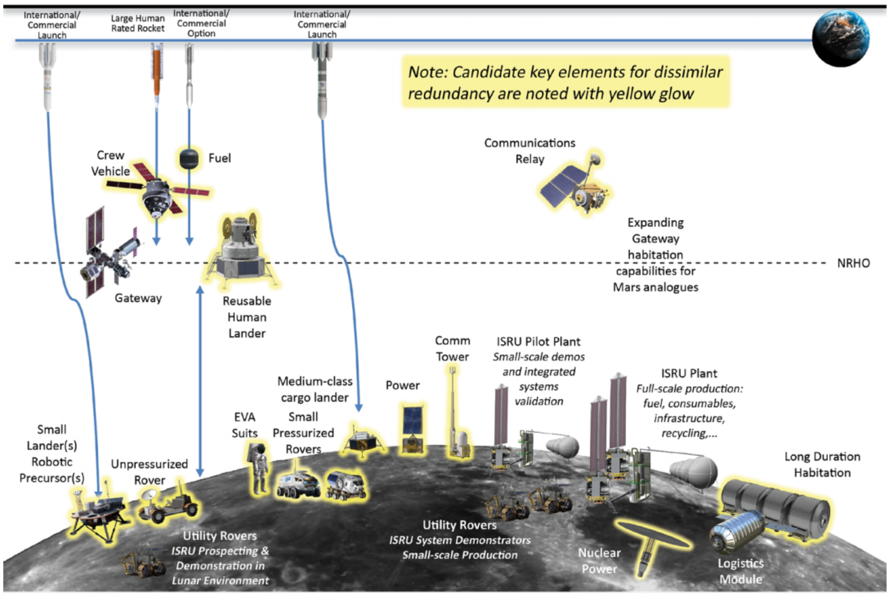


Figure 20.4: Diagram of some key technologies that crewed lunar habitation will require over the next decade

20.4. Technological Gaps and Research Focus

Several technological gaps exist that should be covered in further research. The connection and inflation of the tensairity flooring is possible, but more details have to be confirmed regarding its precise parameters. The anchoring system needs further study, especially regarding anchoring in the lunar environment with weaker gravity, performed remotely.

20.5. Materials

Testing of fabrics should be done extensively to ensure that the numbers attributed to all layers of the habitat are accurate. Testing new weaves would also allow for further optimisation, as a basket weave could potentially be more advantageous.

The bladder layer could also be further optimised, mainly to make it thinner rather than lighter. Typically, bladders in inflatable structures are complex and made out of various layers of materials themselves [23].

20.6. Reliability

As detailed in Chapter 16, reliability of life support systems needs to become better for the mission to be feasible. A redesign of a life-support system adapted to the mission, and specifically the conditions within the habitat, would be very advantageous. Moreover, reliability figures of Starship are merely estimated through those of similar launchers. Future data related to reliability of the launcher would therefore make the reliability figures more accurate.

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A. Functional Diagrams

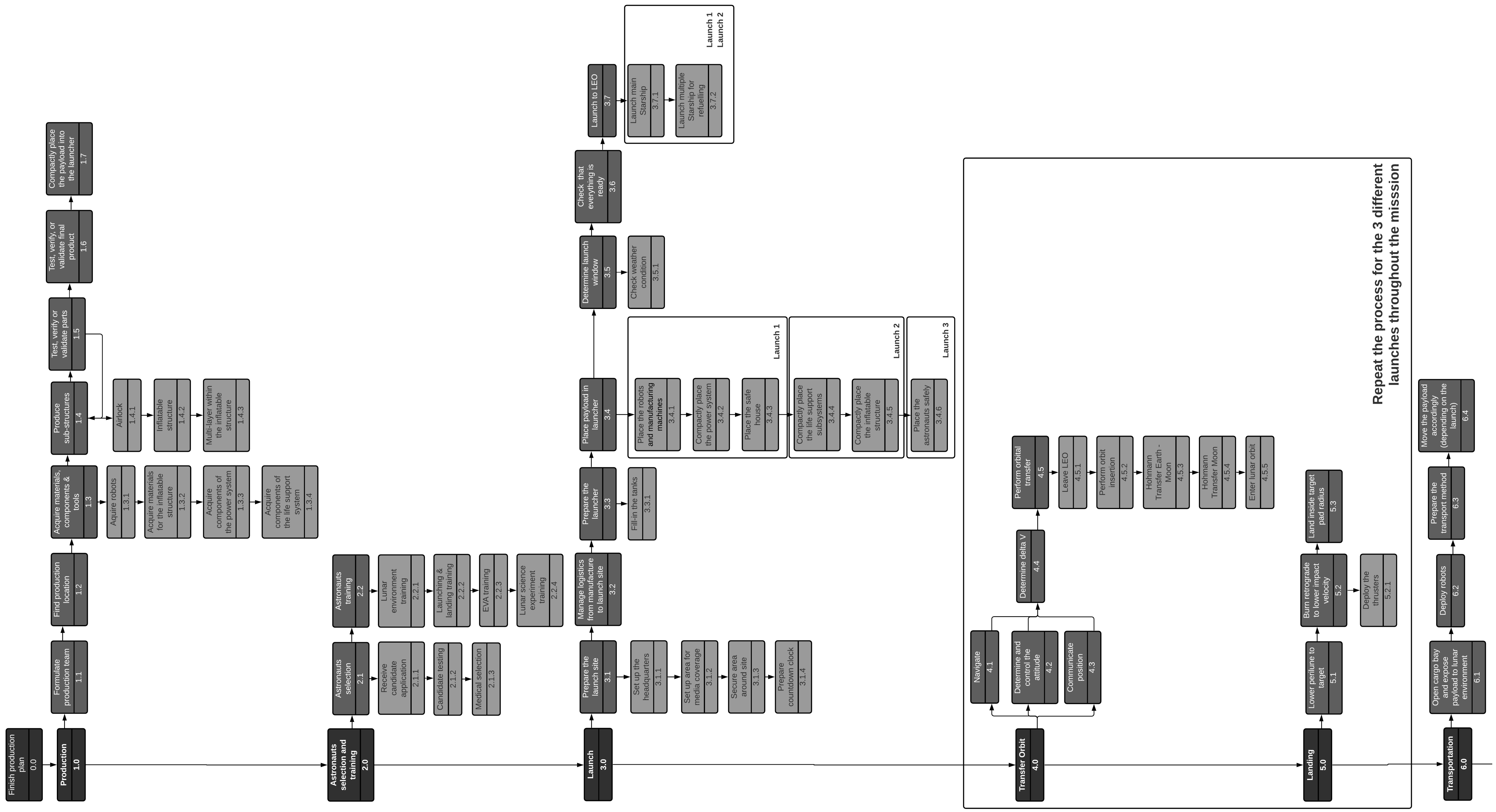

Figure A.1: Functional Flow Diagram part 1(Colour key: the different levels are in shades of grey, darker being a higher level)

Figure A.2: Functional Flow Diagram part 1(Colour key: the different levels are in shades of grey, darker being a higher level)

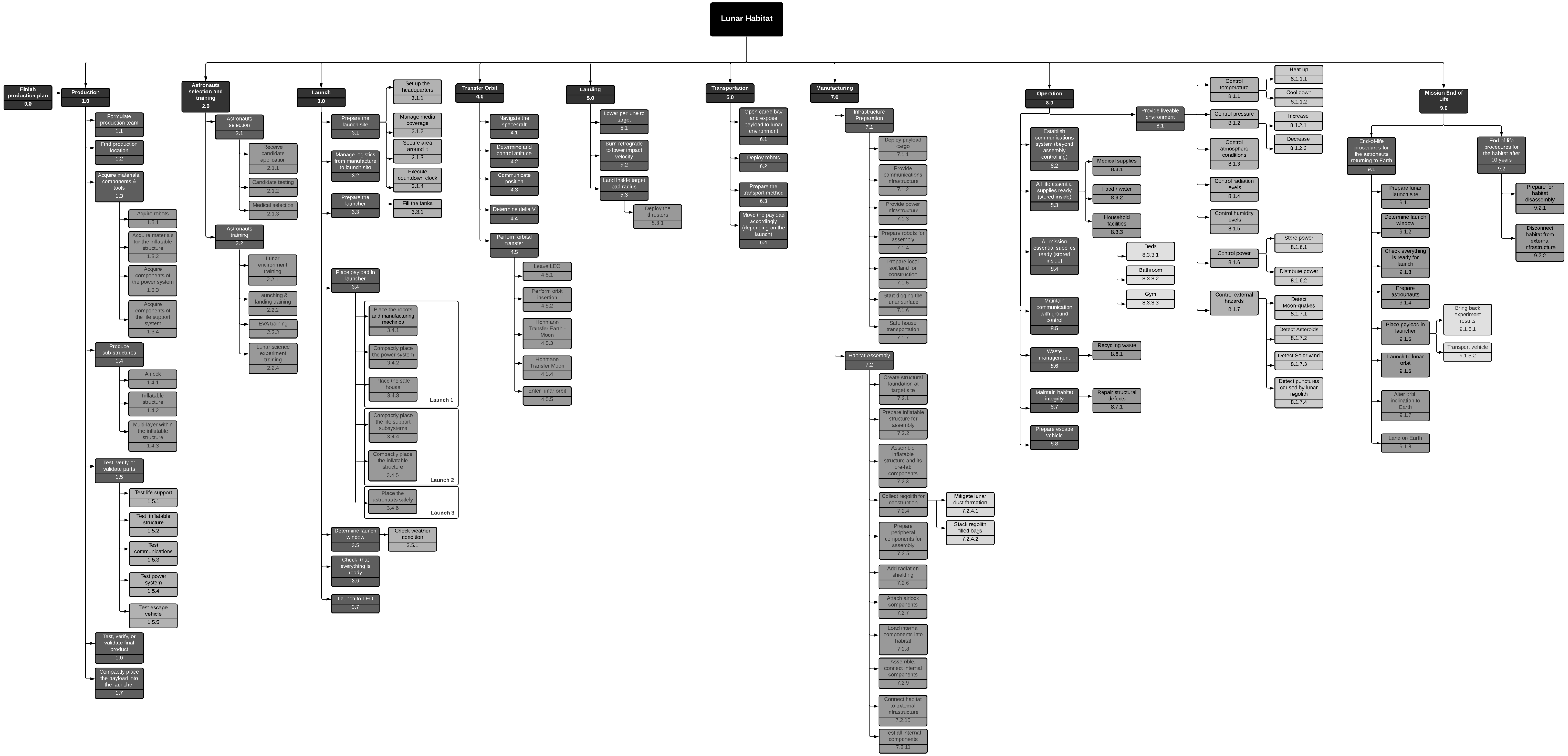

Figure A.3: Functional Breakdown Structure (Colour key: the different levels are in shades of grey, darker being a higher level)

B. Profitability Variable Probability Distributions

Net Present Value Calculations

Table B.1: List of variables used in NPV calculations and the properties of their respective probability distributions. Normal distributions have their standard distributions reported, lognormal ones have their shape factor reported, uniform ones have their range reported.

Variable	Mean	Probability Distribution	Standard Deviation / Shape Factor / Range
Init Hab Investment	1.5E9	Lognormal	0.3
Hab Operational Cost	3.1E9	Lognormal	0.3
Resupply Payload	5.0E4	Normal	5E3
Resupply Payload Refuel	2.5E4	Normal	2.5E3
Return Propellant	2.0E4	Normal	2.5E3
LEO Init. Demand	510	Normal	51
LEO Annual Demand Growth	0.12	Normal	0.02
LEO Annual Launch Cost Decrease	0.05	Normal	0.01
EML1 Init. Demand	550	Normal	55
EML1 Annual Demand Growth	0.14	Normal	0.03
Lunar Surface Init. Demand	440	Normal	44
Lunar Surface Annual Demand Growth	0.27	Normal	0.05
Lunar Surface Annual Launch Cost Decrease	0.05	Normal	0.01
Propellant Facility Specific Development Cost	7.7E4	Lognormal	0.3
Propellant Facility Specific Operational Cost	4.6E3	Lognormal	0.3
Propellant Facility Specific Propellant Output	250	Normal	25
Propellant Facility Resupply Payload	1.2E3	Lognormal	0.3
Equity Percentage in Prop. Facility	0.5	Uniform	0.0-1.0
Initial Market Share	0.75	Uniform	0.5-1.0
Market Share Change Rate	0.05	Normal	0.01
NPV Discount Rate	0.3	Normal	0.09
Market Undercut Percentage	0.75	Uniform	0.5-1.0
EBP LEO Cost	2.4E3	Beta	a=1.5, b=3
EBP Production Cost	1.6	Lognormal	0.3
MBP Production Cost	770	Lognormal	0.3
MBP Annual Production Cost Decrease	0.05	Normal	0.01
MBP LEO Cost Aerobraking Reduction Factor	2.5	Uniform	2.0-3.0

C. Drawing of the habitat

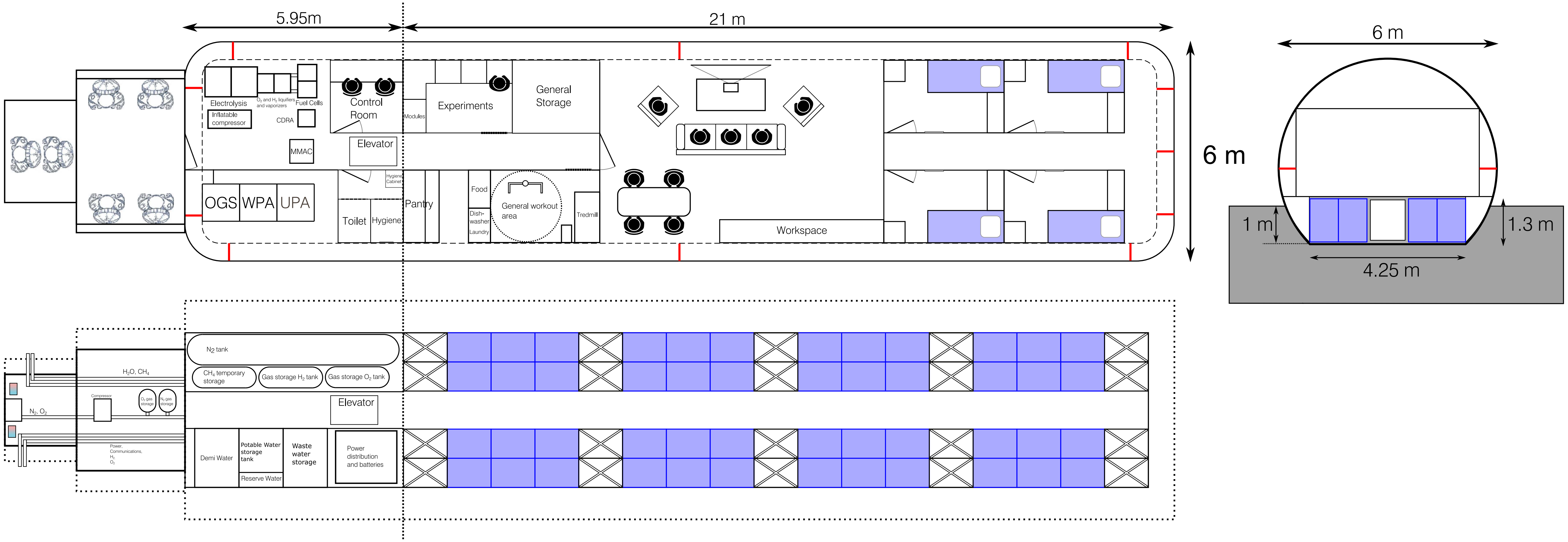

Figure C.1: Overview and main dimensions of the final habitat. It contains the top view of the main floor on top, the top view of the storage on the bottom and the cross section of the habitat on the right. The red lines in the top view of the main floor represents the uninhabitable volume indicated with red in the cross section. The cross section shows the storage placement, floor height and excavation depth. Note that that there is no ceiling inside the habitat, the rectangle inside the cross section is to indicate the habitable volume (2.4 m height). The storage layout shows the gas and water storage in the pre-assembled part, along with the necessary systems to operate the airlocks. Furthermore, the cargo storage in the other part shows the regular cargo boxes in blue, and load bearing cargo elements inscribed with diagonals.

Nomenclature

Abbreviations

Abbreviation	Definition
ADCS	Attitude Determination and Control System
AGV	Automatic Guided Vehicle
ALARA	As Low As Reasonably Achievable
ARS	Acute Radiation Syndrome
CAGR	Compound Annual Growth Rate
CDRA	Carbon Dioxide Removal Assembly
CFPR	Carbon Fiber Reinforced Polymer
CME	Coronal Mass Ejections
DNA	Deoxyribonucleic Acid
DSE	Design Synthesis Exercise
EU	European Union
EML	Earth-Moon Lagrange point
ESA	European Space Agency
ETO	Earth Transfer Orbit
EVA	Extravehicular Activities
FBS	Functional Breakdown Structure
FFD	Functional Flow Diagram
GaAs	Gallium-Arsenide
GEO	Geostationary Orbit
GCR	Galactic Cosmic Ray
H.O.M.E	Habitat On Moon: Exodus
HPS	High Performance Fuel Cell Stack
ICRP	International Commission on Radiological Protection
ISRU	In-Situ-Resource-Utilisation
ISS	International Space Station
LAS	Launch Abort System
LCH_4	Liquid Methane Fuel (cryogenic)
LEO	Low Earth Orbit
LH_2	Liquid Hydrogen Fuel (cryogenic)
LLO	Low Lunar Orbit
LOI	Limiting oxygen index
LOX	Liquid Oxygen (cryogenic)
LV	Launch Vehicle
LRO	Lunar Reconnaissance Orbiter
LTO	Lunar Transfer Orbit
MMAC	Membrane Microgravity Air Conditioner
MULASSIS	Multi Layered Shielding Simulation
NASA	National Aeronautics and Space Administration
NP	Non pressurised
OGS	Oxygen Generation System
P	Pressurised
PAN	Polyacrylonitrile
PBO	poly(p-phenylene-2,6-benzobisoxazole)
PBI	Polybenzimidazole
PE	Polyethylene
PLSS	Portable Life Support System
PV	Photovoltaic
PVDC	Polyvinyl Dichloride
REE	Rare Earth Elements
RF	Radio Frequency
RFC	Regenerative Fuel Cell
SI	International System

Abbreviation	Definition
SPE	Solar Particle Event
SPENVIS	Space Environment Information System
SoI	Sphere of Influence
SOFIA	Stratospheric Observatory for Infrared Astronomy
SNR	Signal to Noise Ratio
STAB	Solar Tower Assembly Box
TA	Teaching Assistant
TBD	To Be Determined
TRL	Technological Readiness Level
TRN	Terrain Relative Navigation
TRR	Technology Readiness Level
TT&C	Telemetry, Tracking and Command
UHMWPE	Ultra-High Molecular Weight Polyethylene
ULA	United Launch Alliance
UPA	Urine Processing Assembly
USD	U.S. Dollar
WBS	Work Breakdown Structure
WFD	Work Flow Diagram
WPA	Water Processing Assembly
WRS	Water Recovery System

Symbols

Symbol	Definition	Unit
A	Surface area	m^2
C	Thermal conductivity	$W\,m^{-1}\,K$
c	Crimp	-
E	Young's Modulus	GPa
F	Force	N
g	Acceleration due to gravity	$m\,s^{-2}$
l	Length	metre
M	Mass	kg
O	Atomic weight	kg
p	Cover factor	-
p_i	Internal pressure	Pa
q	Heat flow	W
r	Habitat radius	m
σ	Tensile strength	Pa
T	Temperature	K
t	Inflatable layer thickness	mm
V	Velocity	$m\,s^{-1}$
W_h	Weight of habitat	N
W_i	Weight of internal systems	N
W_r	Weight of regolith	N
W_R	Radiation weighting factor	-
W_t	Tissue weighting factor	-
Z	Atomic number	-