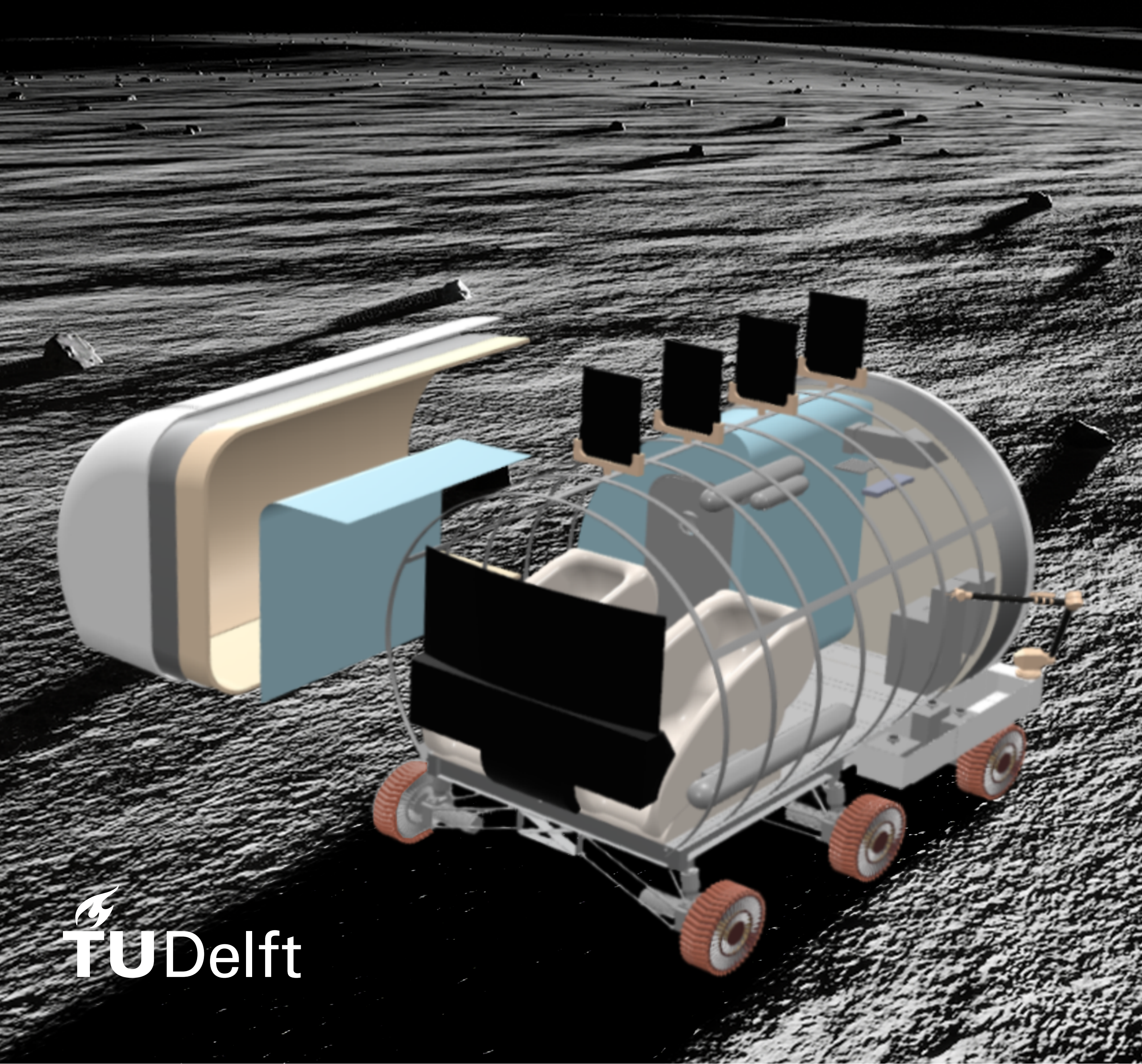
Lunar Buggy

Final Report

AE3200 Design Synthesis Exercise 2024 Spring

Group 18



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Lunar Buggy

Final Report

by

Group 18

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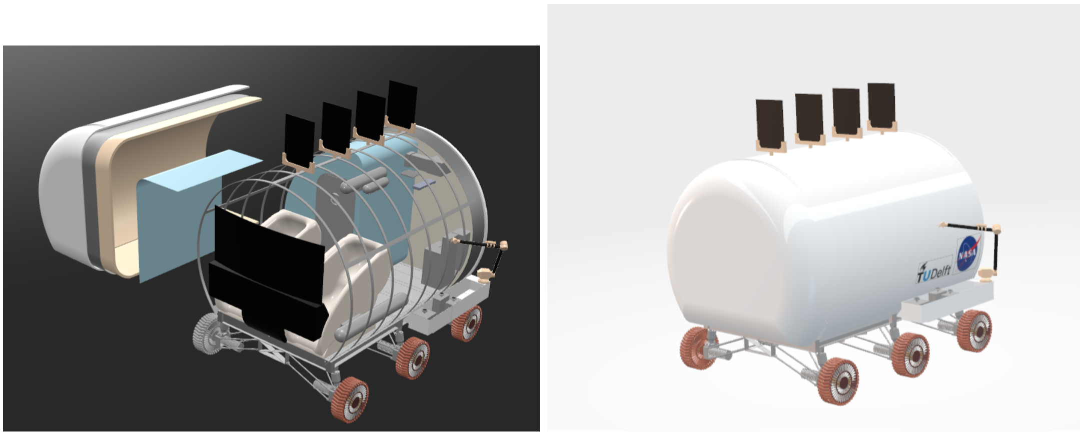
Executive Overview

Project Vision

Lunar exploration is in the focal lenses of spaceflight again. Several plans are in place to establish a permanent human presence on its surface. Lunar Industries aims to take part in these plans by producing a Lunar Transportation System (LTS), an autonomous vehicle able to transport and sustain a crew of two in exploration missions with the potential to become the preferred general transportation method of future lunar inhabitants.

Project Approach

The engineering department of Lunar Industries was tasked to create a preliminary design of the LTS. The project objective is to demonstrate the feasibility of the LTS. A systems-engineering approach was used. A stakeholder and functional analysis, proved essential in stipulating a list of system and subsystem requirements. These formed the base for the design of the single subsystems, which were then integrated together to form a complete system – the LTS. The following paragraphs give an overview of each subsystem, including the respective architectures and the design rationales, and are followed by the integration methodology and some final remarks. However, only a top-level, general outlook is provided here, while more detailed and technical information is found in the rest of the report. An exploded view of the LTS is showed in Figure 1a.



(a) 1. Exploded LTS

(b) 2.Assembled LTS

Figure 1: Final LTS assembly views

Powertrain & Mobility Subsystem

The LTS employs six wheels. Each is independently driven by electric motors and independently connected to the chassis. This allows it to be a very reliable, safe and redundant system. This subsystem is designed to allow the LTS to traverse adverse terrain without compromising on passenger comfortability. In fact, it will employ an innovative, active electromagnetic suspension system. Additionally, regenerative braking will be exploited to recover some of the kinetic energy lost in deceleration. The wheels themselves employ a specialized, cutting-edge composite technology; they are flexible, yet strong and provide sufficient traction such that they allow the vehicle to handle the roughness of the lunar terrain. Finally, four-wheel steering – excluding the middle wheels – will render the rover very manoeuvrable, without compromising on agility.

The entire powertrain and mobility subsystem will weigh 542 kg, and the power consumption of all its components adds up to about 2.4 kW. It allows the LTS to move smoothly across the Moon, with a top speed of 10 km/h, and the ability to climb over slopes of up to 20 degrees.

Structures Subsystem
The Structures Subsystem includes three components - a pressurized cabin, an airlock, a supporting structure, and a frame. The cabin is designed to be a cylinder housing the airlock and most of the other subsystems. The main points for the cabin are that it was chosen to not have windows to, among other things, save on structural complexity. The pressured cabin is made of AlSi10Mg, enabling the possibility of in-situ additive manufacturing, with thickness is 3 mm, ultimately concluding to a cabin mass of 445 kg. The frame on the other hand, is positioned directly below the cabin to provide structural stability and allow proper attachment and load paths from the cabin to the suspension. Surrounding the cabin is a supporting structure sustaining the insulation and radiation shielding layers. In total, the mass of the LTS's structure amounts to 1315 kg.

Life-support Subsystem
The Life-support Subsystem consists of various components: a pressure control system, an air revitalization system, a waste management system, a food and water management system and a fire detection and suppression system. The most important of them is the Carbon Dioxide Removal by Ionic Liquid System (CDRILS), which removes CO_2, humidity and contaminants from the cabin's air. The total mass and power of the Life-support Subsystem are respectively 263 kg and about 1.1 kW.

Radiation and Micrometeorite Protection
For both radiation and micrometeorite protection, which are important agents to protect the vehicle and the crew from, a three-centimeter-thick shield of regolith will be used. The regolith is held by aluminium panels, with a wall thickness of 1 mm, providing additional shielding to the cabin. Additionally, making the shielding out of panels, renders this part of the LTS highly modular. The entire shielding amounts to about 1500 kg of mass. However, about 94 % of it is due to the regolith's weight, meaning that not all of this mass has to be brought from Earth, reducing the launch costs.

Telecommunication Subsystem
The telecommunication subsystem utilises NASA's proposed LunaNet satellite system, which allows positioning and navigation services and also a low-power communication channel with Earth and other agents. Thanks to that, the communication antennas can be very lightweight and consume almost negligible power. Additionally, there will be a separate emergency antenna integrated into both the LTS and into the crew's tablets, which will allow distress broadcast even in the most dire situations, as they work from their own small battery. Lastly, the LTS will also allow WiFi connection to the crew during EVAs in the proximity of the vehicle.

Guidance, Navigation, and Control (GNC) Subsystem
The GNC subsystem consist of LIDARs, Hazcams, and IMUs. The LIDARs are implemented for long-range detection, for mapping, and path planning, while Hazcams are used for short range detection for hazard avoidance. Lastly, the IMUs are used for relative localisation by measuring orientation of the buggy. The designed GNC subsystem fully captures images of the lunar terrain from every angle of the buggy. The Simultaneous Localisation and Mapping (SLAM) algorithm integrates data from LIDARs and Hazcams to update the map and position of the buggy simultaneously, enhancing the accuracy and reliability of the GNC system. Finally, in dark areas, the LTS will have headlights which will improve visibility of the Hazcams.

User Interface Subsystem

The User Interface Subsystem consists of four main components: the chairs, that also function as beds, the dashboard, that allows the crew to control and monitor the LTS, the internal lights, which ensure the health of the crew's natural rhythms, and finally, a window mimicking system. The window mimicking system will satisfy the crew as it will act as an immersive display showing the outside of the LTS just like a window would, but without the engineering challenges brought by creating a hole in the structure for it.

Cargo Handling Subsystem

The Cargo Handling Subsystem consist of a cargo container, a cargo compartment, and a robot arm. For lightweight design and capability during lunar night, carbon-fiber composites (XN-70 and XN-80) are used for the robot arm tube and Bulk Metallic Glass(BMG) for its actuator. To enable autonomous cargo capability, the cargo container will contain RFID tags, in order to be recognised, handled and placed in the desired cargo compartment. The LTS is therefore able to autonomously handle and transport up to 50 kg of cargo.

Thermal Control Subsystem

The thermal control subsystem is responsible for controlling the temperature of the entire LTS, including the pressurized cabin, the electronics, the fuel cells, the motors and the tanks. The thermal control system allows the LTS to operate during lunar days in the Sun and in lunar shadow regions. The thermal control subsystem is designed to be as energy-efficient as possible, thus, a lot of insulation is used to minimize heat losses. The total mass, volume and power consumption of the thermal control subsystem are respectively: 1293 kg, 3.70 m^3 and 625 W.

Electrical Power Subsystem

The electrical power subsystem of the LTS can provide a nominal load of around 4900 W for the entire vehicle, with peak power loads required by the powertrain and suspensions systems. It was decided that the nominal power loads will be provided by a fuel cell system, consisting of a fuel cell stack, four tanks for hydrogen, four tanks for oxygen and two tanks to store the water produced by the fuel cells. In total, the entire EPS weighs about 430 kg. An efficient feed system allows for safe transfer of reactants from the tank to the fuel cell, and a power distribution system allows for safe transfer of power from the fuel cell to the components.

Integration and Assembly

The integration of the LTS is performed in two steps. First, before launch, the main structure is assembled into the stowed configuration. This structure will carry all the extra launch loads induced onto the vehicle during this phase. Upon arrival on the Moon, the second phase of the integration will be performed. Subsystems which are unable to sustain the launch loads when assembled in stowed configuration will be assembled here. The connections are such that replacement and maintainability can be performed easily. Figure 1b the complete assembled LTS is presented.

Cost Budget Breakdown

A parametric equation was used to determine the total budget needed for the development of the LTS and the manufacturing of one unit. These values came out to be between 2.28 and 4.57 billion USD and between 205 and 410 million USD, respectively. The cost to transport a singular LTS to the Moon using Starship is 20.3 million USD. A budget breakdown to a subsystem level was not possible, and would require the use of professionally used tools like NAFCOM. A mass breakdown to the subsystem level is also present, with the total mass of the entire LTS (when fully loaded) equalling 5734 kg.

Verification and Validation
The design was verified and validated using the V-model. Prototype testing and inspections to verify system requirements for the capabilities of the LTS. The prototype will test autonomous capabilities and mobility capabilities while also demonstrating the modularity of the LTS. The inspections in the created list will need to be done before launch and ideally before each mission where the LTS is used. Specific tests for communication systems, the pressure cabin and the radiation protection are also planned. Finally mission requirement verification was done together with the validation of the LTS following the V-model.

Risk Assessment
After the subsystem design, the failure or damage on three subsystems was considered to be most critical for the mission operation: Cabin, Power, and Communication. The failure of these subsystems is directly linked to crew safety and mission failure. The identified risks are visually represented through a risk map and Bowtie diagram.

Market Analysis
The main market gap analysed was for mid-range, highly adaptable lunar vehicles which can be easily altered to changing environments and use cases. The main stakeholders identified were project supervisors and Lunar Industries as a whole. It was decided to assume that 10 LTS should be sold to break even, resulting in a unit price between 435 million and 867 million USD, depending on the effective costs.

Concept of Operations
The main capabilities of the LTS are: a crew capacity of two, endurance of 2 days, range of 300 km, top speed of 10 km/h, teleoperable and level 5 autonomy, with exploration capabilities. Furthermore, the LTS will be transported to the Moon using Starship. After each use of the LTS, inspection will be performed to determine whether maintenance is necessary. If maintenance is necessary, one of the crew members will replace the damaged part with a new part. When the LTS has reached its lifetime, its condition will be assessed. Afterwards, the LTS will be disassembled, and all the parts that are recyclable will be recycled. The LTS has the potential to have a big societal impact. Its impact can be technological, environmental and economical.

Contents

1 | Introduction

Author: Max

The Moon is becoming the center of attention of space exploration once again, with various countries and private companies competing with each other to build lunar infrastructure first. One of the most notable missions is the Artemis campaign[1] which is a collaboration between NASA and various private companies, like SpaceX[2] and Northrop Grumman[3]. As there is a growing market for the exploration of the Moon, more and more opportunities are provided to board this (space)ship. Lunar surface transportation development is not totally new though; there is already a race to design the best system for it, like NASA's *Advance Moon Mobility* competition[4] or the pressurized rover[5] of JAXA's and Toyota's collaboration. Upon identifying an opportunity for medium-ranged, highly adaptable vehicles in the lunar transportation market, Lunar Industries' CEO, Prem Sundaramoorthy, tasked his engineering department with the design of a Lunar Transportation System (LTS). Further, the design shall incorporate the company's strive for sustainability, innovation and inspiration. To make use of then already existing infrastructure, and to potentially secure a first customer, the project will be aligned with the timeline and needs of NASA's Artemis Program. However, the design will also be highly adaptable and modular, as Lunar Industries' aim is to produce a design that will eventually become the preferred general transportation method for future lunar inhabitants.

The purpose of this report is to present the engineering department's progress on the design of the LTS leading up to a 10-week first-order detail design mark. Over the course of ten weeks, the engineering department performed a thorough stakeholder and market analysis to understand the necessary functionalities, and characteristics, of the final system; this process led to formation of stakeholder, mission and system requirements. These requirements helped constrain the design process in a fashion that previously identified needs and goals are met. A quantitative trade-off between five different design concepts was then performed, which ultimately led to the selection of a buggy as the LTS design concept; the buggy was defined as a pressurized cabin resting on top of wheels. Having selected a concept, a deeper dive into the integral functionalities and characteristics of, and the risks associated with, the buggy was performed. This helped identify the necessary subsystems and the requirements these should comply with. This led to the preliminary design process of each subsystem and to a subsystem requirements compliance verification. Further, the entire integration of all subsystems was verified against the system requirements. Moreover, future design steps, verification and validation actions, a production plan and a return on investment analysis were performed. Effectively, the engineering department of Lunar Industries developed a first-order detailed design for the LTS, taking into account all necessary systems engineering methods and providing insights into future steps, and this report aims to convince the board of Lunar Industries to support the continuation of the project past the 10-week mark.

In Chapter 3, an analysis of the lunar setting and considerations of major importance for the design of each subsystem will be featured in Chapter 3. Subsequently, an overview of the LTS'

[1] https://www.nasa.gov/humans-in-space/artemis/
[2] https://www.spacex.com/humanspaceflight/moon/
[3] https://www.northropgrumman.com/space/nasas-artemis-program
[4] https://www.nasa.gov/news-release/nasa-selects-companies-to-advance-moon-mobility-for-artemis-missions/
[5] https://global.toyota/en/mobility/technology/lunarcruiser/

necessary functionalities will be presented in Chapter 4. In Chapter 5 through Chapter 15, the design process for each subsystem will be detailed. Besides, the methodologies employed to obtain the selected subsystem design, a failure modes analysis, a subsystem requirements compliance verification and an overall design analysis will be presented. In Chapter 16, the projected production plan will be estimated. In Chapter 17, the total mass, power and cost budgets will be given. In Chapter 18, the integration of all system requirements will be discussed, while an outlook into future verification and validation steps will be detailed. In Chapter 19, an overview of the top level risks and their mitigation, and contingency, strategies will be provided. In Chapter 20, a market analysis culminating in a return on investment estimation will be featured. Finally, in Chapter 21, the concept of operations will be discussed.

2 | Project Summary

Author: Dani

The engineering department of Lunar Industries, by the request of the CEO, Prem Sundaramoorthy, is working on a sustainable, innovative and adaptable LTS for exploration and for the ferrying of future inhabitants on the Moon. As the project arrives at a major milestone, the Preliminary Design Review, this is a good opportunity to have an overview of the project's goals, phases, the current state of affairs and an outlook on the future developments in the design process of the LTS. Presenting this will build trust and unity in the design and the approach of the engineering department by showing the necessary steps that have been (and will be) taken. The hope of the engineering department is that the Preliminary Design Review convinces the board of Lunar Industries to support the continuation of this project past the 10-week mark.

The motivation for the development is characterized by the main stakeholders; they identified two market gaps within the lunar transportation market and are determined to leverage these opportunities. Doing so, could not only prove very profitable, but it could also greatly further human exploration while having meaningful innovative and inspiring implications. From the stakeholder needs, the following mission statement and project goal were proposed in their respective order:

"The Engineering Department of Lunar Industries will demonstrate an autonomous and adaptable system for ferrying passengers and cargo by 2040, thereby enabling safe and sustainable exploration and transportation of future lunar inhabitants on the Moon's surface."

"To design an autonomous and adaptable Lunar Transportation System (LTS) that enhances human mobility on the lunar surface and allows agile and safe exploration of the lunar environment."

Additionally, as one of the project's main objectives is to allow sustainable exploration of the Moon with the LTS, the following sustainability goals were established under the Lunar Sustainability Initiative. The goals' main purpose is to *lead, take responsibility and demonstrate viability*.

1. Promote sustainable values among team members.
2. Learn from, lead, educate and inspire the aerospace industry, the public and the stakeholders.
3. Take social responsibility through ethical supply chain management.
4. Take environmental responsibility through emission, waste and resource management.
5. Demonstrate the return on investment at sustainability for stakeholders.

6. Demonstrate safe and comfortable usage in the long term.
7. Demonstrate the possibility of designing a new circular and modular lunar transportation system.

The project was divided into five main phases. Phase 1, the Preliminary Concept Design, is an accelerated first-order design phase, which needed to be done in 10 weeks in total. This can be broken down to three sub-phases. The first sub-phase is the project scope, during which the main needs and goals of the project are defined and a thorough market analysis is performed. The second sub-phase is the conceptual trade-off between potential design candidates, where the buggy, a pressurized cabin on top of wheels, came out winning out of five concepts. The other possible vehicles were a snake that propels itself forward with a screw-like outer body, a sledge that pushes a ship-like bottom hull and a cylinder that has two big wheels and a small motorized one. The buggy quite clearly outscored the other concepts in sustainability, simplicity and practicality while scoring second best in mobility in a trade-off. Hence lastly, during the third sub-phase the first-order design of the buggy was concluded. Moreover, it should be noted, the principal supporting infrastructure development is outside the scope of this project, as largely the Artemis architecture is going to be used for this. Overall, the Phase 1 objectives were indeed achieved within the first 10 weeks of the project with the Preliminary Design Review (PDR) coming up.

The first-order design of the buggy addressed all the crucial top-level choices and options to provide a clear picture of the LTS and its approach to sustainability. Each of the subsystems were designed to such a degree that it proves its feasibility, safety, reliability and capabilities. The in-depth technical designs are presented throughout this report, with additional focus on the future outlook of the project. Regarding sustainability, to ensure easy adaptability and decommission of the LTS, the vehicle was designed in such a way that it allows quick and accessible repair and maintenance through modularity.

Phase 2 is the full detailed design of the system which runs in parallel with some testing and verification already. After that, in Phase 3, the full design can be manufactured and assembled to verify and validate all the requirements and performance characteristics. Then, during Phase 4, a demonstration will be performed on the surface of the Moon before the end of 2040. If all of these are successful and the qualification tests are passed, then in Phase 5 the production of the system can be started based on demand. Supply chains and facilities are set up before production. The timeline for this is shown in Figure 2.1. This is an optimistic timeline, which allows a safe margin until 2040 to complete the project in case of delays.

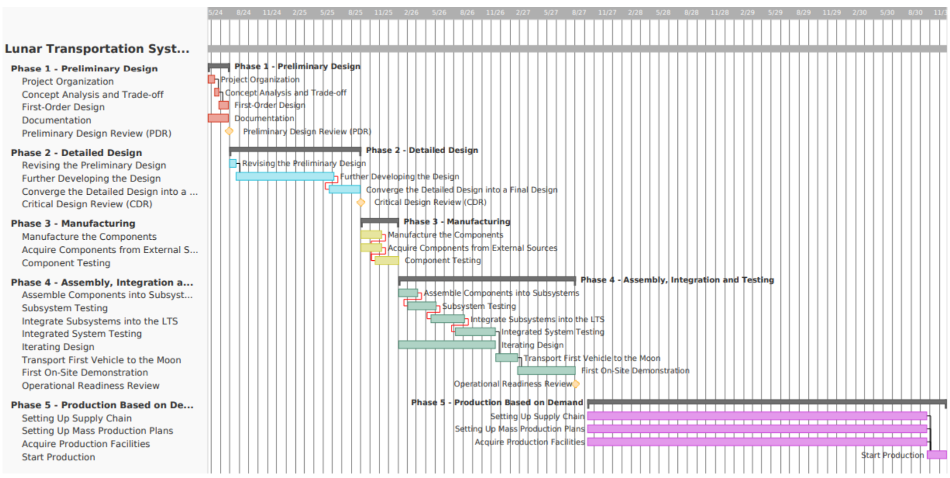


Figure 2.1: Project Timeline after PDR

In the following diagram the **Project Design & Development Logic** can be found. After this DSE, the LTS will go into the detailed design phase. Then the prototype will be manufactured and tested. Based on this, a design iteration might be made. After enough confidence in the design, the final version of the LTS can be manufactured, followed by the assembly. Then the LTS will be prepped for launch, where the stowed configuration will be designed, manufactured and tested. This is followed by a launch readiness review. Then before the launch the LTS will undergo an operational readiness review after which it is launched. Once the LTS arrives on the moon it will be assembled and deployed. Then it will perform its mission. At EOL the LTS systems and materials will be assessed, and recycled/repurposed where possible.

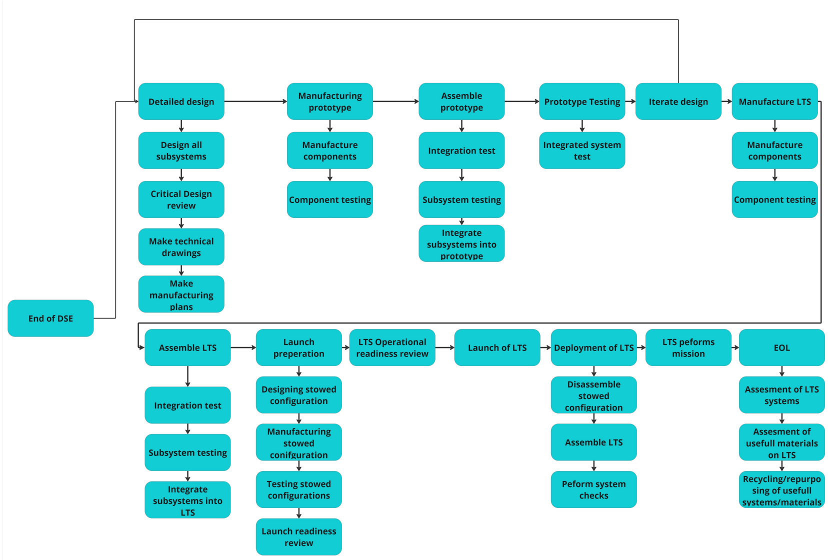


Figure 2.2: Project Timeline after PDR

3 | Lunar Environment

Authors: Luca, Dani, Ishaan, Aleksei, Max

3.1. Artemis Program

The LTS will rely on the Artemis Program. It represents NASA's first step in their Moon to Mars campaign, which aims to bring humans to the Moon first, and eventually to the Red Planet. As part of the Artemis program, NASA is planning to build permanent infrastructure on the Moon [1], which the LTS will exploit during its use. NASA has defined an Artemis Exploration Zone (AEZ), a circular region of roughly 150 km radius up to 84° S around the Basecamp, located next to the South Pole, presumably near the Shackleton crater (Figure 3.3). This is the region that the LTS is going to be developed for.

LunaNet

LunaNet is a lunar communication satellite system in development by NASA. With Delay/ Disruption Tolerant Networking (DTN), it allows near-instantaneous telecommunication between assets on the lunar surface and Earth [2]. The system comprises the Lunar Segment, a constellation of satellites in lunar orbit, and the Earth Segment, satellites in terrestrial orbit (Figure 3.1). LunaNet is essential for operations in the AEZ, as not all locations can be in direct communication with Earth, (the LTS cannot "hear" Earth). Finally, LunaNet will also provide navigation and guidance to the LTS, similarly to GPS systems [2].

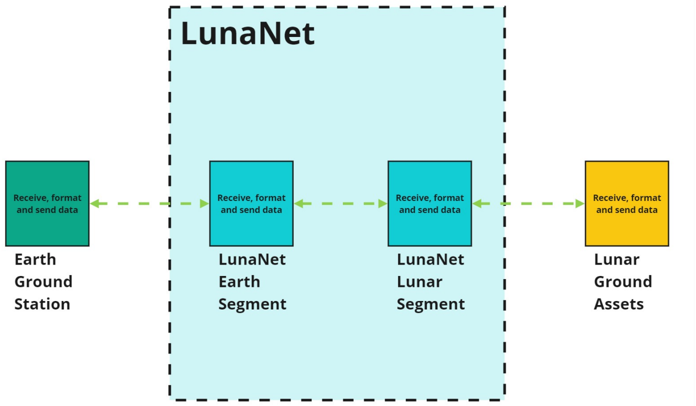


Figure 3.1: LunaNet system architecture

Basecamp

Basecamp is the transfer point for humans and equipment to and from the lunar surface, to be put in place by NASA as part of the Artemis Program. It will be the base of human operations on the Moon, and it will also function as a habitable hub for visiting humans. Naturally, Basecamp will also function as the base for any LTS operation.

Basecamp will also be able to provide assets like the LTS a source of energy through its own energy generator, a nuclear fission reactor (Figure 3.2) [1]. Due to the long periods of dark in the AEZ, nuclear fission is not only the most sustainable energy source, but also the most feasible and technologically ready option. Nuclear is not always the preferred option in terms of sustainability, but in this case, the benefits outweigh the costs [3]. It is very reliable and safe (the restricted zone from the reactor is only a 500 m radius circle), and it can sustain both Basecamp, and systems like the LTS during the long night periods.

NASA demonstrated the high technological readiness of the reactor experimentally (based on the Kilopower project [4]), and is planning on performing on-site tests as well [5]. Even though decommissioning a nuclear reactor is far from easy, it is expected to be able to produce 40 kW of power for ten years without any human involvement, and requiring only $\sim$50 kg of Uranium [3]. Therefore, the reactor is able to sustain missions on the Moon for a long period, with no considerable impact left behind. Lunar Industries considers this a sustainable and viable option to sustain the LTS during the night periods, although, supplying the LTS with energy during these periods is the responsibility of the customer.

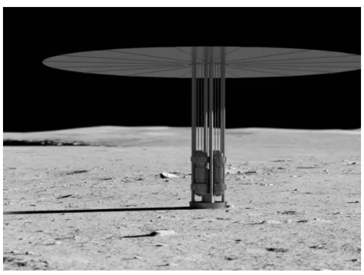


Figure 3.2: NASA's planned nuclear fission reactor [1]

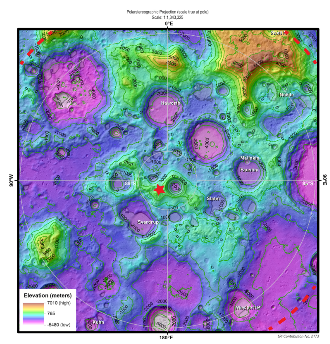


Figure 3.3: Topography of the Moon's south polar region, where the Artemis Exploration Zone is shown with red dashed lines, and the Basecamp with a red star. [6]

3.2. Lunar Environment

To correctly design the LTS, the present situation in its future operational region need to be carefully assessed. The LTS will be operational within the AEZ, and thanks to previous satellite missions like the Lunar Reconnaissance Orbiter[1] or the Lunar Orbiter Laser Altimeter[2], the terrain conditions are fairly well known. The topography of this area will be analysed in this section.

Figure 3.3 shows elevations in the expected area of operations of the LTS and is observed to range from -5480 m to 7010 m. On the other hand, Figure 3.4 portrays the slopes in the same area. It can be observed that the majority of slopes are below 20°, and the slopes exceeding this value are those of large craters, where the LTS is not expected to be operational. It can therefore be expected that the LTS will not be used in areas with slopes exceeding 20°.

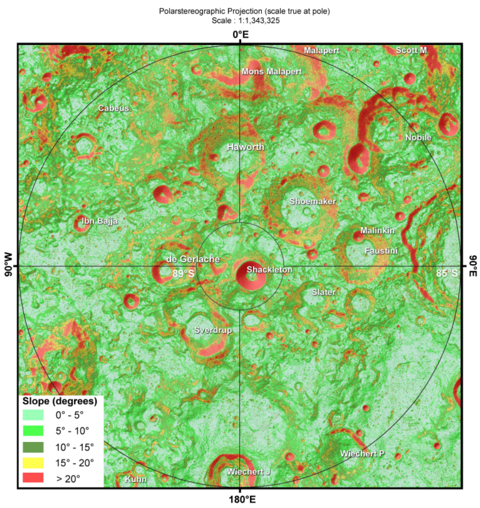


Figure 3.4: Slope Map of the Moon's South Pole (85°S to Pole) [7]

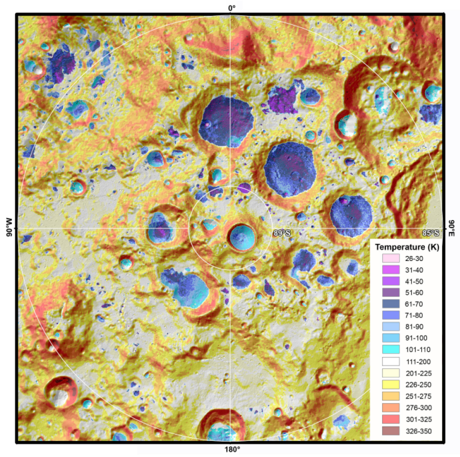


Figure 3.5: Near-Surface Temperatures Modeled for the Moon's South Pole (85°S to Pole) [7]

In terms of temperature, the slope of the landscape has vastly more importance than the altitude. Because of the absence of atmosphere, altitude has little effect on local temperature. On the other hand, slopes create shadows on the surface, and light/shadow fluctuations have a large impact on temperature changes. In Figure 3.5, the observed maximum near-surface temperature is represented. The highest temperature within the AEZ amounts to 350K. It can also be observed that some regions, notably craters, have extremely low temperatures, in the range of 26K to 40K. These are referred to as Permanently Shadowed Regions (PSRs), which, due to the low elevation of the Sun with respect to the horizon, are never in Sunlight.

Because of the absence of a proper atmosphere and magnetic field on the Moon, the LTS and its crew will be exposed to high radiation levels compared to the levels on Earth. Besides the exposure to galactic cosmic rays, exposure to Solar Particle Events (SPEs) may also occur [8]. SPEs have very high variability in flux and energy spectra, and are directly correlated to the occurrence of sunspots, as more radiation is emitted from the Sun on those occasions. SPEs depend on two cycles: solar cycles, and grand solar cycles. A solar cycle lasts about 11 years and is nearly periodic. During this cycle, radiation levels varies from a period of minimum activity to one of maximum activity. On the other hand, the magnitude of the maximum activity

[1] https://science.nasa.gov/mission/lro/
[2] https://science.nasa.gov/mission/lro/lola/

depends on the 300-year-long grand solar cycle [9].

In Figure 3.6, levels of solar activity from 1980 to 2040 are shown. The presence of solar cycles is evident. The periodicity of the activity can also be observed, although the magnitude of the maximum levels seems to be decreasing, expectedly due to grand solar cycles. Figure 3.7 presents the grand solar timeline [9]. For the LTS's design, it is useful to note that in the period between 2031 and 2043, the *Modern Minimum 1* is expected to occur, and the Sun's activity is expected to be reduced by 70%. Although, after this period, solar activity is expected to increase again, and the LTS will need to be designed for these levels as well.

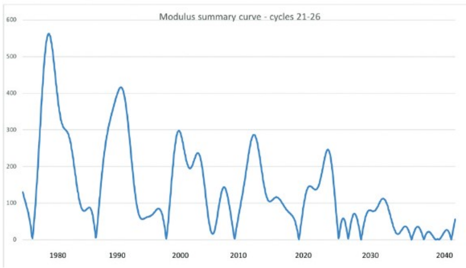


Figure 3.6: The modulus summary curve associated with the sunspot numbers derived for cycles 21–23 [9]

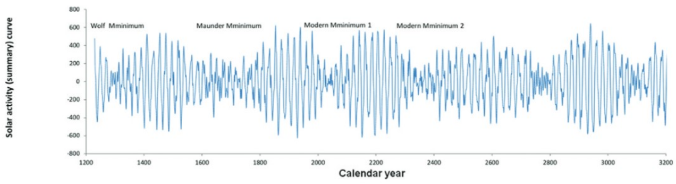


Figure 3.7: Solar activity for 1200–3300 AD [9]

4 | Functional Analysis

Authors: Luca, Max

Before delving into the design of the LTS, it is essential to analyse the system's and mission's desired capabilities. This can be done through a functional analysis, in which all top-level functions are analysed, then broken down into lower-level ones, until a satisfactory level of detail and depth is reached. Then, once all functions have been identified, they can be converted into functional requirements, which, together with requirements stemming from stakeholder needs, complete the requirement list of the LTS.

In Figure 4.1, the Functional Breakdown Structure (FBS) is shown. An FBS is useful to group functions by type, emphasizing their similarity. On the other hand, Figure 4.2 and Figure 4.3 show the Functional Flow Diagram (FFD). In the FFD, the same functions of the FBS are shown, but organized in such a fashion to emphasize their chronological order, dependencies, and sequencing.

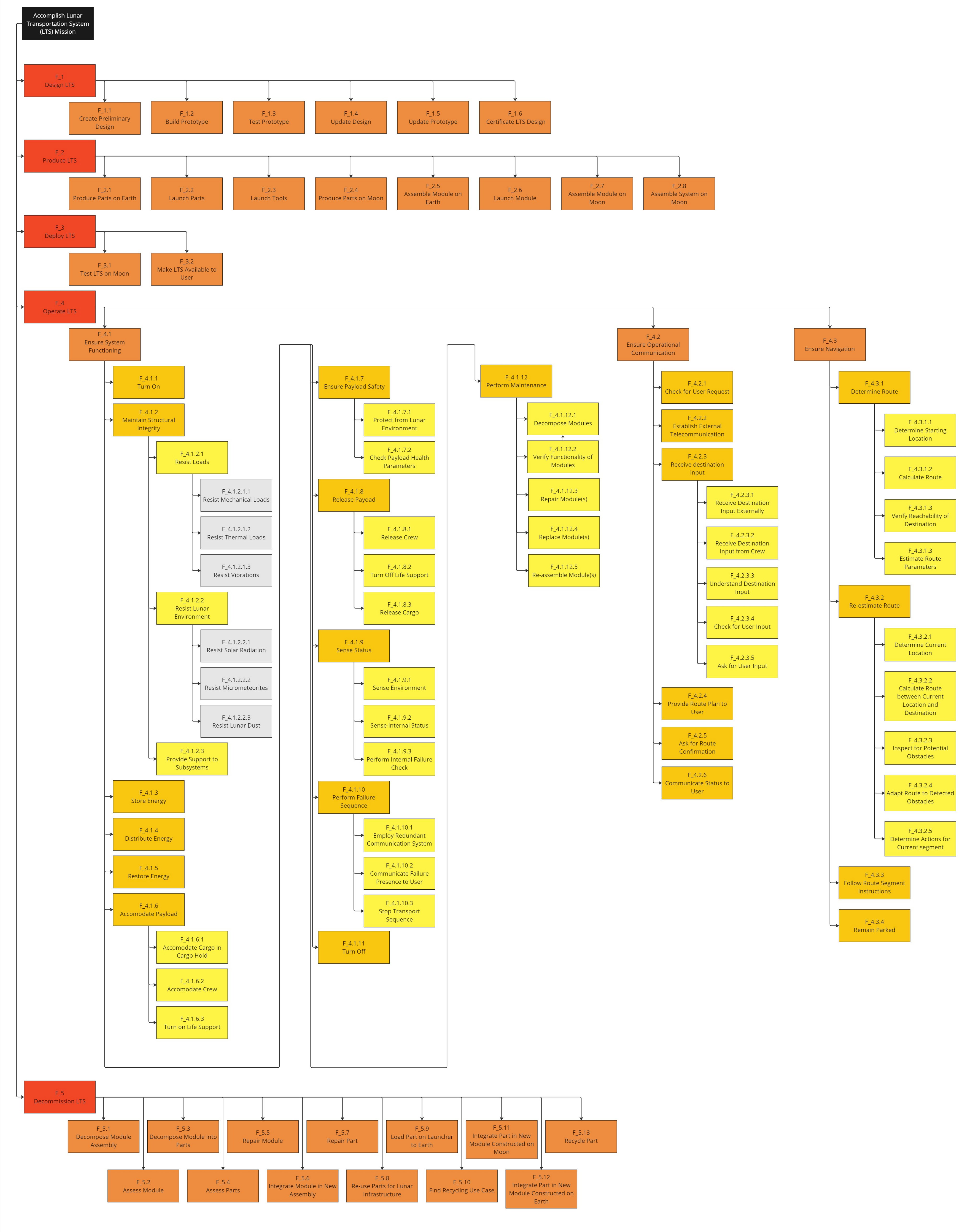


Figure 4.1: Functional Breakdown Structure

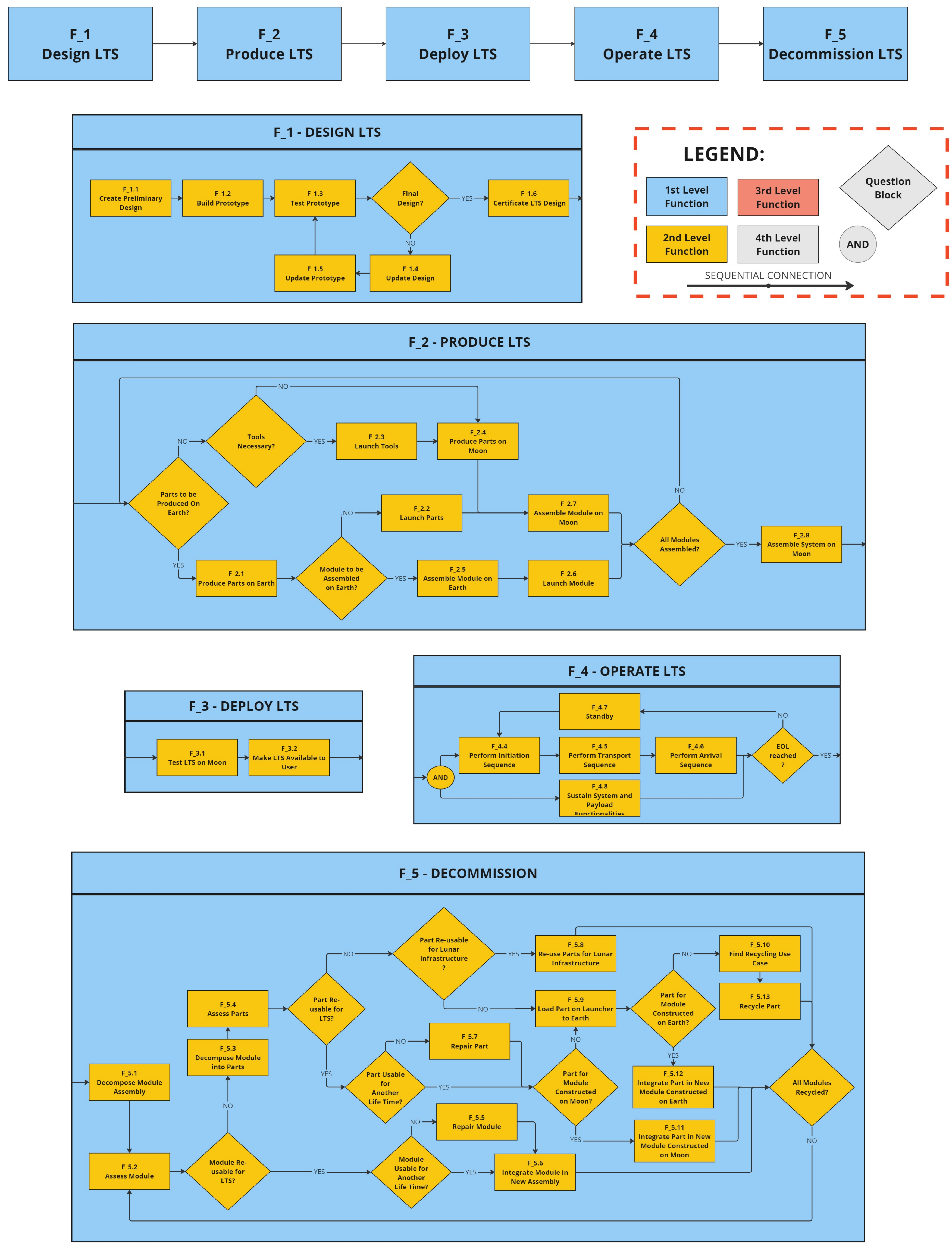


Figure 4.2: Functional Flow Diagrams Part 1

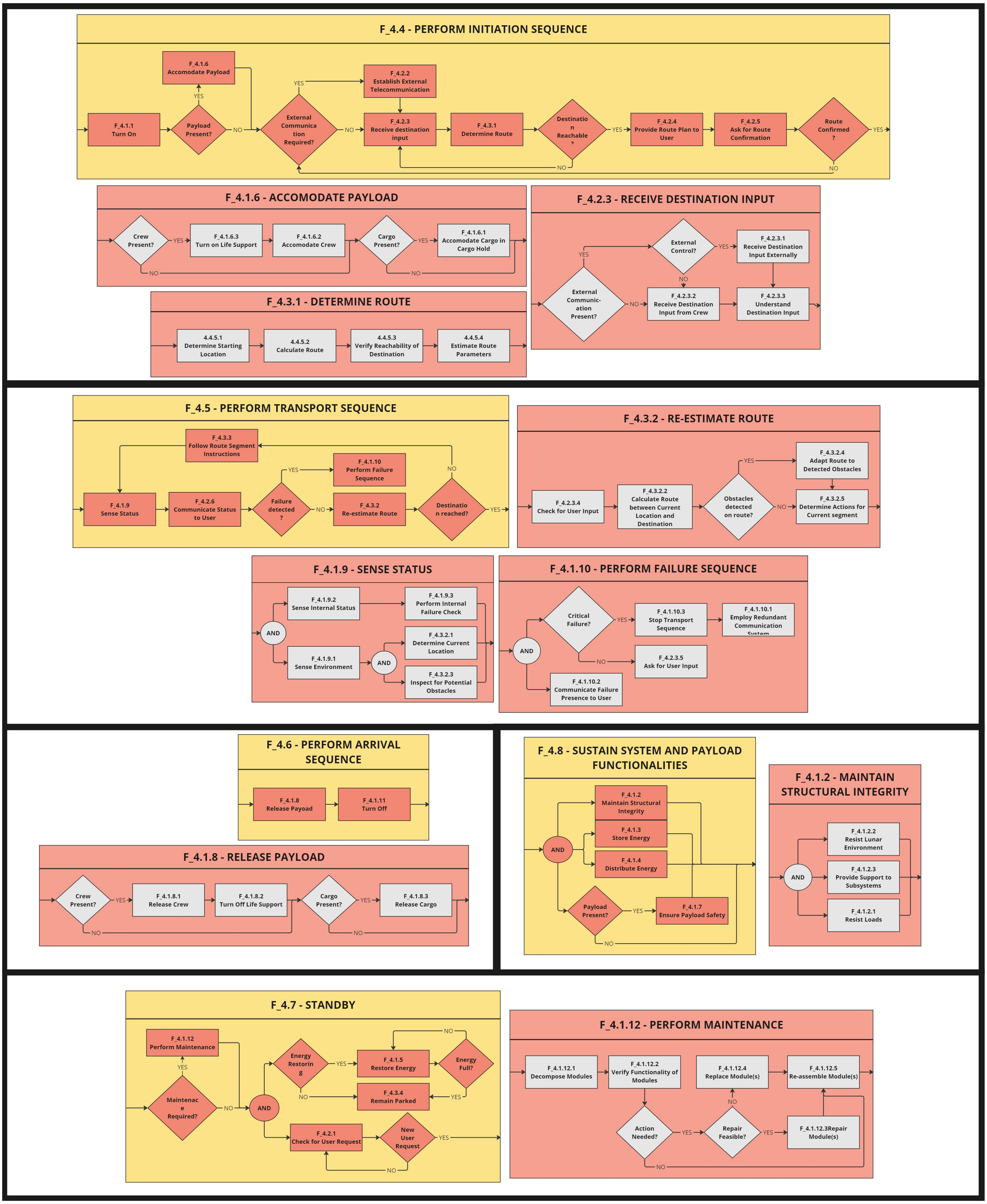


Figure 4.3: Functional Flow Diagrams Part 2

5 | Powertrain & Mobility Subsystem

Authors: Luca, Max

In this chapter, the powertrain and mobility subsystem design will presented. The rationale behind every design decision will also be explained. In Section 5.1, the general architecture of this subsystem is presented. Later, a dynamic analysis is presented in Section 5.2; here, the use cases of the LTS are analysed and design forces and torques are derived. Furthermore, in Section 5.3, the design processes of the different components of this subsystem are described. Additionally, Section 5.4 presents the final mass and power budgets of this subsystem, while Section 5.5 lists its possible failure modes. Finally, in Section 5.6 and Section 5.7, the compliance of the designed subsystem with the sustainability objective and the aforementioned subsystem requirements are described, respectively.

5.1. Overview

The powertrain and mobility subsystem consists of the motors, the suspension, braking and steering systems, the wheels and all connecting parts. It ensures efficient movement and handling of the LTS. The motors convert electrical energy into mechanical energy, while the braking, suspension and steering systems and the wheels work together, delivering a smooth ride and precise control.

It was decided to equip the LTS with six wheels, and Michelin's Moon Wheels were identified as the most viable option. The DC WEG MT-040 drive motor, which will also function as a regenerative braking system, was selected to be integrated into every wheel, while the WEG ML1601 steering motor was chosen to be integrated into the front and back two wheels to enable four-wheel steering. Further, a trade-off yielded an active electromagnetic suspension system as the best option to absorb the ride's shocks. An overview of the entire powertrain and mobility subsystem except for the steering motor is portrayed in Figure 5.1 – the wheel, driving and braking motor, shock-absorber, and structural rods are shown. The following sections will elucidate each design choice.

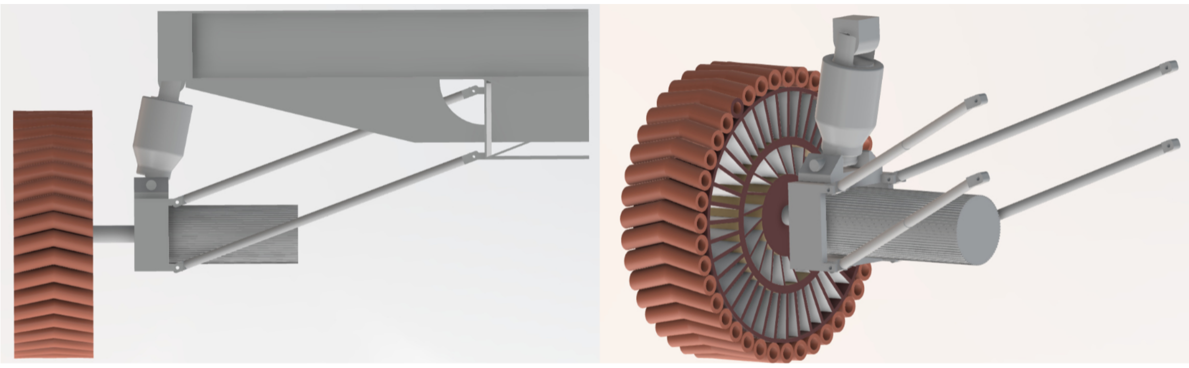


Figure 5.1: Front view (left) and isometric view (right) of wheel assembly with wheel diameter 72 cm and width 23 cm.

5.2. Dynamic Analysis

The powertrain's main objective is to provide enough driving power to the wheels. A dynamic analysis was performed to estimate how much power each wheel needs in all of the LTS' use cases. In addition to that, this analysis will help obtain load ranges in all directions, which will lay a base for the design of the suspension system and the chassis.

The dynamic analysis consists of analysing torque requirements and resulting forces on the rover's wheels in each of the following use cases: nominal cruise condition, accelerating at an incline, braking at a decline, turning with terrain tilt, and traversing terrain with both incline and tilt and with both decline and tilt. These were deemed to be the limiting dynamic cases of the LTS. Table 5.1 lists the mobility parameters involved in this dynamic analysis. Further, this analysis was also based on parameters like the LTS' mass and geometry. These were continuously iterated over, in parallel with the design of the other subsystems as well. Their final values are listed in Table 5.2. The subsystem includes six wheels, which was deemed an appropriate amount for both redundancy and manoeuvrability, including two front wheels, two middle wheels, and two back wheels.

Table 5.1: Mobility requirements and corresponding parameters

Requirement ID	Parameter	Value	Unit
REQ-SSYS-PM-1.6.1.4.1	Cruise Speed	7	km/h
REQ-SSYS-PM-1.6.1.8.1	Turning Speed	5	km/h
REQ-SSYS-PM-1.6.1.5.1	Turning Rate	20	°/s
REQ-SSYS-PM-1.6.1.1.1	Incline Slope	20	°
REQ-SSYS-PM-1.6.1.2.1	Tilt Slope	20	°
REQ-SSYS-PM-1.6.1.1.1	Acceleration at Incline	0.5	m/s^2
REQ-SSYS-PM-1.6.1.7.1	Braking Distance	5	m

Table 5.2: Estimated geometry parameters

Parameter	Value	Unit
Maximum Mass	5797	kg
x location of CG	1.95	m
y location of CG	1.55	m
z location of CG	1	m
LTS length	3.9	m
LTS width	2.5	m
Wheel lateral position	30	cm
Wheel Radius	36	cm

Another key parameter of this analysis is the rolling friction coefficient of the lunar surface. In other publications, a value of 0.04 is used [10], and it was deemed an appropriate estimation for this scope as well. It should be mentioned, however, that this parameter is hard to estimate due to the little information available about the Moon's composition, and its estimation could introduce an error in the analysis.

The cases listed above are composed of longitudinal and lateral manoeuvres. For longitudinal manoeuvres, total forces and torques were computed for wheel pairs (back, middle, and front), while, for lateral ones, what was computed was the lateral force, and the share of forces and moments acting on the left and right wheels. Let's take the following use case as an example: turning to the right while traversing some terrain with an incline of 10 degrees. The analysis of climbing an incline in a longitudinal manoeuvre will output a torque requirement for each wheel pair and forces in the X (longitudinal) and Z (vertical) direction, again for each wheel pair. In the absence of lateral manoeuvres, both wheels in the pair would share the same torques and forces, but analysing the lateral manoeuvre of turning, the share of the total force and torque is larger on the "internal" wheels . Lateral manoeuvres also result in forces in the Y (lateral) direction, which can be computed too.

Table 5.3 shows the results of the described dynamic analysis. For simplicity, reliability, and manufacturability, all wheels, suspensions and motors will be sized based on the worst-case loads, which are highlighted in bold. As mentioned before, F_x, F_z and T are the total forces and moments on a wheel pair, and L and R represent the shares of those forces and moments acting on the left or right wheel of the pair respectively, although, due to symmetry, the two values can be inverted. Additionally, a positive torque is a driving one, while a negative torque corresponds to a braking one. Finally, the use case for a terrain with both maximum tilt and maximum incline is not considered, as it would correspond to a terrain with an overall slope of more than 20°, which is only rarely encountered in the Artemis Exploration Zone, near craters, and will be avoided.

Table 5.3: Required wheel torques for key use cases, and resulting forces on the wheels (extreme values are highlighted in **bold**)

USE CASE		**LONGITUDINAL MANOEUVRE**			**LATERAL MANOEUVRE**		
		Fx (N)	**Fz (N)**	**T (Nm)**	**L (%)**	**R (%)**	**Fy (N)**
Nominal cruise condition	Front	123.8	3096.1	44.6	50	50	0
	Middle	123.8	3096.1	44.6	50	50	0
	Back	123.8	3096.1	44.6	50	50	0
Accelerating at max incline	Front	1334.1	**2059.5**	372.6	50	50	0
	Middle	2014.5	3109.9	572.6	50	50	0
	Back	**2695.0**	**4160.3**	**752.6**	50	50	0
Braking at max decline	Front	**-2410.9**	4127.6	**-650.4**	50	50	0
	Middle	-1802.2	3085.4	-486.2	50	50	0
	Back	-1193.5	2043.3	-322.0	50	50	0
Turning at max tilt	Front	123.8	3096.1	44.6	**66.9**	33.1	± **1522.2**
	Middle	123.8	3096.1	44.6	**66.9**	33.1	± **1522.2**
	Back	123.8	3096.1	44.6	**66.9**	33.1	± **1522.2**
Accelerating at 10° incline and 10° tilt	Front	1630.2	2516.6	455.3	55.7	44.3	± 537.6
	Middle	2014.5	3109.9	562.6	55.7	44.3	± 537.6
	Back	2398.9	3703.2	669.9	55.7	44.3	± 537.6
Braking at 10° incline and 10° tilt	Front	-2146.0	3674.1	-578.9	55.7	44.3	± 537.6
	Middle	-1802.2	3085.4	-486.2	55.7	44.3	± 537.6
	Back	-1458.4	2496.8	-393.4	55.7	44.3	± 537.6

The results obtained in this section will be referred to during the explanation of the design choices of all components of the powertrain and mobility subsystem. It is worth mentioning that, as evident from Table 5.1, this analysis is based on the LTS' requirements, and for this reason, the design that will follow this analysis is expected to comply with the aforementioned subsystem requirements.

5.3. Component Design

Wheels

Several factors need to be taken into account when designing wheels that are to be used in the South Polar region of the Moon. Firstly, the wheels will need to be able to operate throughout a large range of temperatures – from 50K to 200K approximately. Furthermore, the wheels will need to resist the abrasive nature of the lunar regolith and the increased lunar radiation levels. Moreover, they will need to provide enough traction and flexibility to handle the sandy and rocky terrain. In addition, shock absorption is particularly crucial in such a low gravity environment in order to reduce bounciness.

The ideal wheel for the lunar terrain and conditions, and for the goals of Lunar Industries, would be able to balance strength, traction, flexibility, durability, temperature resilience, dust mitigation, radiation hardening, energy efficiency, potential for modularity, potential for future in-situ production, low weight and low complexity. Investigation concluded that the following materials could satisfy at least some of these properties and could potentially play a role in the wheel design: metals, elastomers, advanced ceramics, advanced elastomeric and polymeric composites and shape memory alloys. A trade off analysis evaluated each material on a scale of one to ten based on the criteria. Every criteria was deemed to have the same importance, as all the criteria represent essential features for proper functioning and an economically, and environmentally, sustainable use of the LTS. As portrayed in Table 5.4, composites outscored all the other options by a clear margin of 16 percent with regards to elastomers, the second best option. Moreover, composites consistently score well in all criteria with the exception of the *Potential for Future In-Situ Production* criterion and the *Low Complexity* criterion in which it still scores satisfactorily. It is, therefore, quite evident that composites offer the best alternative when directly compared to the other material options without having to conduct a thorough sensitivity analysis.

Table 5.4: Wheel Material Trade-Off

Criterion	**Metals**	**Composites**	**Advanced Ceramics**	**Elastomers**	**Shape Memory Alloys**
Yield Strength	8.0	7.0	6.0	4.0	6.0
Traction	7.0	8.0	6.0	6.0	7.0
Flexibility	5.0	7.0	3.0	9.0	8.0
Durability	8.0	7.0	7.0	5.0	6.0
Temperature Resilience	7.0	8.0	9.0	3.0	7.0
Dust Mitigation	6.0	8.0	7.0	8.0	7.0
Radiation Hardening	6.0	7.0	9.0	3.0	5.0
Energy Efficiency	5.0	7.0	4.0	6.0	5.0
Potential for Modularity	6.0	8.0	5.0	7.0	6.0
Potential for Future In-Situ Production	4.0	6.0	3.0	7.0	5.0
Low Weight	5.0	9.0	7.0	8.0	6.0
Low Complexity	7.0	5.0	3.0	9.0	6.0
Total	**6.2**	**7.3**	**5.8**	**6.3**	**6.2**

The possibility of combining several materials as different wheel parts was also explored. The idea was to leverage a material's strong suits for a part that does not expose its drawbacks

and to assemble several such parts into a wheel – for instance, using titanium for spokes or advanced ceramics, such as silicon carbide, as a coating protection on critical surfaces of the wheel. Doing so, the engineering department attempted to produce its own wheel design. For instance, one consisted of a core of 16 titanium spokes encompassed by a titanium shell, of an outer elastomeric layer and of an encasing titanium mesh.
The engineering department of Lunar Industries has not been able to find commercially available composites tailored to the lunar environment, thereby rendering the use of composites for self production difficult, though previously identified as potentially having the most benefits during the trade-off. Lacking the for-decades-accumulated composite research expertise for this specific Moon application, which companies like Michelin or Goodyear have developed, the wheel designs produced by the engineering department of Lunar Industries are unable to compete in terms of their mass. In fact, all self-produced designs ended up weighing in the range of 50 to 100 kg, while the mass of a Michelin Moon Wheel with greater dimensions and made out of a patented composite material with sufficient capabilities only has a mass of 15 kg [11].

Concurrently, research was conducted to explore the wheel designs of LTS' competitors – Toyota-JAXA, Intuitive Machines, Lunar Oupost and Venturi Astrolab – and that of NASA's 1971 Lunar Rover Vehicle. In addition, third party companies, such as Michelin, Goodyear or Bridgestone, which are developing lunar wheels, were investigated. In accordance with the above trade-off, Michelin, showcasing wheels made out of an unpublished patented composite material with a titanium core, provides the best commercially available option. Establishing itself in the lunar wheel market as a dominant force for the past decades, Michelin has produced two lunar wheel models. The first Michelin Wheel model dates back almost twenty years ago, but provides ideal properties for the LTS. Weighing 15 kg, the non-pneumatic 71 cm diameter, 23 cm wide wheel is made out of an unpublished patented lunar regolith-resistant elastomeric composite; specifically tailored to the lunar terrain, it can resist a maximum load of 4500 N and a top speed of 10 km/h [11], thereby satisfying REQ-SSYS-PM-1.6.1.4.3, REQ-SSYS-PM-1.6.1.1.1, REQ-SSYS-PM-1.6.1.2.1, REQ-SSYS-PM-1.6.1.5.1 and REQ-SSYS-PM-1.6.1.8.1. Further, it can operate in cryogenic temperatures down to 40K. The wheel's treads provide firm traction and great material flexibility as depicted in Figure 5.2, improving shock absorption. In addition to a distinct titanium core, which has the potential to be made out of replaceable spokes, the composite component is clearly made out of several distinct layers which can each be produced through additive manufacturing, adding to the adaptability and sustainability of the wheel through modularity and complying with REQ-SSYS-5.2.1.1.1. Michelin's Moon Wheel, having a diameter of 71 cm will certainly be able to cross obstacles with a height of 15 cm thereby satisyfing REQ-SSYS-PM-1.6.1.4.4. Though produced in the 2000s, this wheel fulfills all capabilities required by the LTS, including those presented in the dynamic analysis of Section 5.2, and was, therefore, the chosen wheel design for the LTS.

Figure 5.2: Michelin's First Moon Wheel Design

Figure 5.3: Michelin's New Moon Wheel Design

Michelin recently announced it has developed an improved model, also made out of one of its patented unpublished composites [12]. It is 3D printed, airless and grounded in biomimetics, as portrayed in Figure 5.3. No further details have been published yet, however, it is expected that all capabilities will at least be improved. It is foreseeable that this improved wheel will be commercially available by 2040. Therefore, while Michelin's old wheel design is currently selected as the LTS' wheel design, it is likely to be replaced by the newer model after it has undergone substantial testing and has become available. For now, however, the design process will resume with Michlelin's old wheel design until further notice.

Suspension System

The choice of the suspension system is of very high importance in the design of the LTS. A suspension system is defined by its working mechanism, which was traded-off upon, and its damping properties. This section will expand on the suspension choice, presenting the driving parameters and the preliminary suspension configuration.

According to REQ-SSYS-PM-2.2.1.17.2 and REQ-SSYS-PM-2.2.1.17.1, the LTS' suspension system is required to have a damping ratio between 0.20 and 0.30 and to have a vertical travel range of $\pm$ 15 cm. To meet these requirements, the suspension system shall have certain spring stiffness and damping coefficients. The spring coefficient can be obtained from the range of vertical loads to be experienced by the LTS during its use, obtained in Section 5.2, and the vertical travel requirement, obtaining a required spring stiffness coefficient of 6.5 kN/m. Then, according to Equation 5.1 [13], where c is the damping coefficient, m is the effective mass, k is the spring stiffness coefficient and ζ is the damping ratio, satisfying the first requirement would mean to employ a shock absorber with a damping coefficient in the range of 1 and 1.5 kNs/m; this would ensure the damping ratio requirement is met. These spring stiffness and damping coefficient requirements will ultimately drive the final selection of the shock-absorber and spring systems.

$$\zeta = \frac{c}{2\sqrt{km}} \tag{5.1}$$

There are several major differences in designing a suspension system to be used in the Artemis Exploration Zone and one to be used on proper roads on Earth – reduced gravity, cryogenic temperatures, rough, abrasive and sandy terrain and the absence of air. Firstly, reduced gravity by a factor of 6 implicates that the suspension system would experience a lower load on the Moon when compared to on Earth, meaning that the suspension spring can be softer. On the other hand, reduced gravity also means that the softer suspension system might not be as efficient in maintaining contact with the ground. This is something that is of major issue when encountering terrain irregularities or performing sharp maneuvers and brakes. Further, reduced gravity also means that, once the LTS encounters an irregularity, it might not settle as quickly. Consequently, the damping would need to be adjusted to account for longer oscillation periods. Moreover, cryogenic temperatures heavily complicate the use of hydraulics and very flexible elastomers, while the absence of air renders a reliable pneumatic suspension system extremely complex.

Two widely employed suspension systems in the space industry are torsion bar systems and rocker-bogie systems. The rocker-bogie suspension system, however, has the drawback that it is designed for very slow-moving rovers, such as NASA Mars rover Curiosity which has a top speed of 0.12 km/h [1]. A rocker-bogie suspension system design was discarded, as the LTS needs to attain speeds of at least 7.00 km/h; though the motor can provide higher speeds, the LTS will be limited to 10 km/h because of the aforementioned chosen wheel's limit. On the other hand, torsion bars have a fixed spring rate, limiting their ability to adapt to varying terrain conditions – which are abundant in the Artemis Exploration Zone – which effectively reduces shock absorption efficiency. Further, torsion bars allow for limited wheel articulation. Hence, torsion bars were also discarded as a potential suspension design.

Rough terrain would expose a badly suspended LTS to a wide range of vertical oscillation frequencies, which can trigger motion sickness when frequencies of 0.2-1.0Hz are reached, and to a wide range of lateral oscillation frequencies, which can trigger head toss when frequencies of 2-8Hz are reached [14]. In fact, NASA standard STD-3001 requires the the limitation of vibrational loads between 0.5Hz and 80Hz. Passive suspension systems generally have fixed characteristics – fixed damping and spring rates – and hence may not offer great adaptation to varying terrains, leading to more pronounced vehicle motions. Moreover, passive suspension systems, which couple left and right wheels, may even amplify lateral movements in the vehicle. Considering the planned extended trip duration of the LTS and the roughness of the lunar terrain, it is highly questionable that a passive suspension system would successfully prevent the crew from experiencing motion sickness or head toss.

Taking into account the two-day maximum trip duration of the LTS, the irregularity of the lunar terrain in the Artemis Exploration Zone, the increased tendency to lose ground contact and the prolonged oscillation periods, an active suspension system was deemed preferable. Active suspension systems are also most capable of complying with the NASA-STD-3001 standard which limits frequencies in the range of 0.5 to 80 Hz, thereby fulfilling REQ-SSYS-PM-2.2.1.17.3. Research was thus conducted into potentially applicable active suspension systems. A major constraining factor was the aforementioned intention to exclude pneumatic and hydraulic systems. The following non-pneumatic and non-hydraulic active suspension systems have been identified: electro-mechanical, electromagnetic and piezoelectric.

A piezoelectric suspension works as follows: the control unit transmits electric signals to the piezoelectric actuator which then creates an electric field within the piezoelectric material; this material may then deform as commanded by the control unit which maps the terrain beforehand. While piezoelectric technology much resonates with the innovation and inspiration goals of Lunar Industries, piezoelectric material expansion varies a lot with big temperature changes and its functioning in cryogenic temperatures is questionable [15]. Consequentially, piezoelectric suspension systems were discarded, leaving electromagnetic and electromechanical suspension systems.

[1] https://hackaday.com/2023/09/14/rocker-bogie-suspension-the-beloved-solution-to-extra-planetary-rovers/

A trade-off was hence performed between electromechanical and electromagnetic suspensions. Scores from one to ten were attributed to different criteria and justified with a rationale, as presented in Table 5.5. Each criterion received the same weighting, as all the criteria represent essential features for proper functioning and economically, and environmentally, sustainable use of the LTS. As can be seen, the electromagnetic suspension offers the better solution by a margin of 10%, predominantly outscoring its electromechanical alternative.

Table 5.5: Trade-Off Between Electromagnetic (EMA) and Electromechanical (EME) Suspension

Criterion	EMA	EME	Rationale
Innovation	9	7	EMA is highly innovative, advanced technology, while EME is well-established.
Lunar Sand Resistance	8	6	EMA exposes fewer moving parts to sand.
Cryogenic Temperatures Resistance	7	6	Mechanical parts may have thermal expansion issues in EME, while electronics do for both.
Low Complexity	6	8	Magnets are more complex than mechanical parts.
Low Maintenance Difficulty	6	7	More complex for magnets in EMA, but still many moving parts in EME.
Performance (response time & precision)	9	7	Magnetic control in EMA is superior to mechanical inertia and response time.
Power Efficiency	8	7	EMA is very efficient with a potential for energy recovery, while EME has more friction and mechanical efficiency related energy losses.
Potential for Future In-Situ Production	7	5	EMA relies on advanced magnetic and electric components, while EME consists of metals, electronics and polymers and employs more complex manufacturing processes.
Potential for Modularity	8	7	EMA allows for easier integration of modular parts due to magnetic levitation, while the many mechanical connections in EME are more complex.
Adaptability	8	7	EMA is intrinsically very adaptable to varying terrain conditions, while EME would require mechanical adjustments.
Low Weight	7	7	Very similar masses.
Low Power Consumption	7	8	EME generally has lower power consumption due to simpler mechanical systems.
Total	**7.5**	**6.8**	

Nonetheless, electromagnetic suspensions are not an established concept in the automotive industry on Earth. While they seem technologically very appealing due to their very high performance, their drawbacks of high costs and mass [2] rendered them commercially unfeasible in the automotive industry. However, as the harsh lunar environment most likely rules out

[2] https://incompliancemag.com/bose-electromagnetic-car-suspension-system/

durable, and reliable, pneumatic, hydraulic and passive suspension systems, as outlined above, the engineering department of Lunar Industries envisions electromagnetic suspension systems to perhaps establish themselves as the primary viable suspension system for lunar vehicles.

Most notably, Bose, usually specializing in noise-cancelling technologies, aimed at developing the most advanced and highest-performing vibration-cancelling suspension system. While publicly available data comparing different electromagnetic suspension systems is limited, and though dating back to the 2000s, Bose's Project Sound still stands as a significant pioneering effort in this field. Portrayed in Figure 5.4, Bose's suspension system, the engineering department's chosen suspension system, consists of a linearly actuated electromagnetic damper resting on a spring. It thus is made out of three distinct parts – an actuator, a magnetic system and a spring, complying with REQ-SSYS-5.2.1.1 .2. It performs passively, as well as actively. In other words, it can react to road irregularities, but it can also, given signals received from the control unit, anticipate road irregularities and adjust accordingly. The control unit uses the input from several sensors, tracking the terrain, speed, acceleration and braking among others.

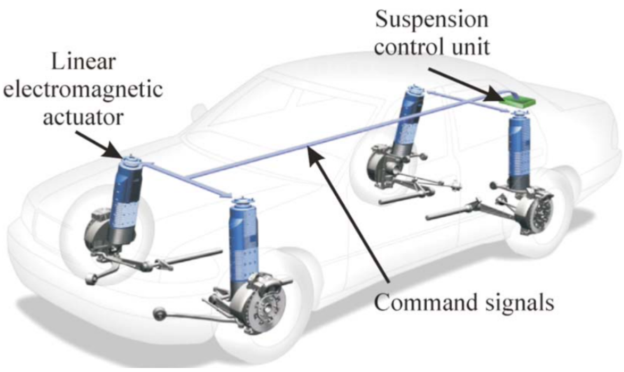


Figure 5.4: Bose Suspension System

Its damping coefficient of 1200 Ns/m fits well within the allowable damping coefficient range, thereby complying with REQ-SSYS-PM-2.2.1.17.2 [14]. Its spring stiffness coefficient lies at 30 kN/m and is thereby heavily over designed. However, this can rather easily be accounted for as the over designed lower spring segment can be redesigned to fit the needs. Off-road tests with potholes and bumps at maximum load forces of 4 kN and nominal load forces of 2 kN, thereby satisfying the maximum and nominal load estimates, have been performed; these demonstrated the high efficacy of the suspension system [14]. Moreover, the suspension system only consumed an average of 16 W per damper throughout the total off-road driving cycle – a value comparable to that of other active suspension systems for normal city driving – though it reached an instantaneous peak damping power of 2 kW. Designing the power system for occasional instantaneous power retrievals of up to 2 kW could hence allow for the use of an overall low-power active suspension system when compared to normal city driving. Moreover, the bound and rebound strokes were tested to be 80 and 58 mm, fulfilling REQ-SSYS-PM-2.2.1.17.1. It has a mass of 40 kg and dimensions: height of 40cm and a diameter of 15cm. The compactness of the design helps mitigate lunar dust related issues, however electronics will need to be kept at operating temperatures. Therefore, Bose will need to integrate a thermal management system into its suspension system. Accounting for this, and for the fact that the lunar terrain features not only many road irregularities as in the aforementioned tests, but also many slopes, a liberal safety factor of 2.5 has been used to fix an upper bound limit for the average power consumption of 40 W per suspension.

In addition to the aforementioned shock-absorber, the suspension system of the LTS will also include some structural components. Using a double wishbone suspension configuration, the wheels will be connected to a central spine of the chassis by four rigid rods. These rods will be connected such as to allow for vertical travel of the wheel, without compromising its angle with respect to the ground, as represented in Figure 5.1.

These rods shall provide rigidity to the powertrain subsystem, and in doing so, they shall resist the loads they are expected to be subjected to during the LTS' use. The dynamic analysis discussed in Section 5.2 proved useful to know the loads the wheel is subjected to, while to determine the internal loads of the structures, they were treated as trussed – thus assumed to only carry axial loads. As shown in Table 5.3, the maximum X, Y, and Z loads that each wheel can be subjected to are 2695, 1522, and 4160 N, respectively. Using these, the worst-case tension and compression loads were calculated: 16 and -16 kN, respectively. For the rods' performance to be satisfactory, they need to not yield in either of the load cases, and to not buckle in compression.

The rods are rigid circular tubes with a thickness of 5 mm. Furthermore, the rods will be composed of titanium, due to this material's good strength-to-weight ratio, and its resilience in harsh conditions [16]. Knowing the internal loads of the rods, a minimum cross-sectional radius R was found to avoid both yielding and buckling, using respectively Equation 5.2 and Equation 5.3; this resulted in a minimum radius of 4.6 mm for yielding, and 11.9 mm for buckling. Incorporating a safety factor of 10 %, the rod radius will be 13 mm.

$$\sigma = \frac{F}{A} = \frac{F}{\pi R^2 - \pi (R-t)^2} \tag{5.2}$$

$$F = \frac{I\pi^2 E}{L^2} = \left(\pi R^4 - \pi (R-t)^4\right) \frac{\pi^2 E}{L^2} \tag{5.3}$$

Driving System

In the dynamic analysis, described in Section 5.2, wheel torques on each wheel and for each use case were computed. As stated earlier, the driving system consists of six independent electric motors. These were all sized to accommodate for the worst-case required driving torque. As highlighted in Table 5.3, the maximum torque to be provided to a pair of wheels is 752.6 Nm, meaning 376.3 Nm per motor. This value can be used in Equation 5.4 to obtain the required driving power, including ω, the wheel's rotational speed, which amounts to 51.5 rpm at nominal speed. In conclusion, it was determined that the maximum amount of power ever required by one of the motors is 2032 W, or 2.72 hp.

$$P = \omega \cdot T \tag{5.4}$$

An electric motor was selected from the ones available on the market. It is the WEG MT040[3] DC motor. This motor is optimal for its use. It has a high reliability, and can operate at temperatures ranging from -250° C to 400° C [4].

[3]https://www.weg.net/catalog/weg/US/en/Electric-Motors/DC-Motor-Products/Fractional-DC-motors/MT-040-3-HP-180-Vcc-4P/p/14461749

[4]https://www.wegmotorsales.com/basics.html

Steering System

One of the LTS' design drivers is its manoeuvrability. An optimal steering system allows the vehicle to turn efficiently, nimbly, and with a minimal turn radius. To achieve optimal manoeuvrability in tight spaces, the LTS will employ four-wheel steering. When making a turn, the front and back wheels will turn opposite directions. It should also be mentioned that, when turning, wheels on different sides of a vehicle turn at different speeds, and therefore, conventional cars employ differential gearboxes, but due to the independence of the six wheels, the LTS's powertrain will not need such a component.

REQ-SSYS-PM-1.6.1.8.1 and REQ-SSYS-PM-1.6.1.5.1 specify that the minimum turning speed and turning rate are 5 [km/h] and 20 [°/s] respectively. Combining these two requirements, a requirement for a minimum turning radius of 4 [m] was obtained. Figure 5.5 schematically shows the top view of the LTS. It is clear from the figure that the required steering angle is given by Equation 5.5. In conclusion, from the steering requirements, a steering range of $\pm$ 30° was identified.

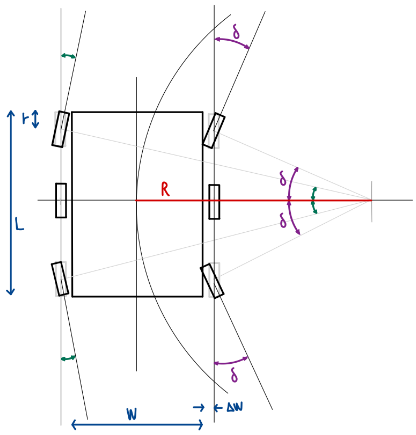


Figure 5.5: Steer angle sketch

$$\delta = arctan\left(\frac{L/2 - r}{R - W/2 - \Delta W}\right) \tag{5.5}$$

Steering will be enabled by four steering motors, one per steerable wheel. These were selected based on the power they are required to provide. To estimate the power, a contact analysis was performed. The Hertzian Contact Surface Theory was used [17]. This theory describes deformations due to the contact of two cylinders based on their material properties, geometries, and the applied force. It assumes a quadratic pressure distribution within the contact patch of the wheel and the Moon's surface, with the peak being along its center line.

Applying this theory, the results depicted in Table 5.6 and Figure 5.6 were obtained - a is half the length of the contact surface, d is the depth that the wheel sinks into the surface, P_{Max} is the maximum pressure, while P_{Avg} is the average pressure on the ground. As for the driving system, the power required is given by Equation 5.4. The steering speed was assumed to be 5 °/s, this was deemed an appropriate figure for the speed range of the LTS. The steering torque required is the moment created by the friction forces, which was assumed to act along the two lines of average pressure, showed in Figure 5.6. The friction force was calculated based on the dynamic friction coefficient, estimated to be 0.63 [10], and the wort-case vertical force, obtained in Section 5.2. Following this procedure, a required steering power of 23.6 W was found.

Table 5.6: Results of Hertzian contact analysis

Parameter	Value	Unit
a	7.14	cm
d	1.4	cm
P_{Max}	486776	Pa
P_{Avg}	47095	Pa

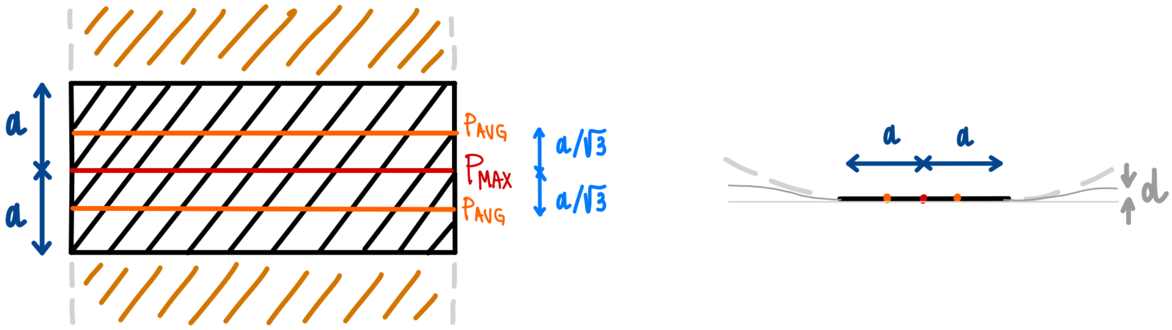


Figure 5.6: Sketch of Contact analysis Parameters

As a steering motor, the WEG ML1601[5] was selected. It can provide up to 0.05 hp, or 37 W, and is therefore more than sufficient to satisfy the LTS' steering needs. As for the driving motor, the WEG ML1601 motor is reliable and can operate at a wide temperature range, manifesting itself as the most suitable choice for the LTS.

In addition to driving the choice of the steering motor, the Hertzian contact theory is also useful to estimate the surface pressure of the wheel. According to NASA standards, to maintain sufficient traction without sliding, a wheel on the Moon shall not exert more than 70 kPa [10]. Also, REQ-SSYS-PM-2.3.1.1.1 states that the ground pressure should not be more than 70 kPa. As shown in Table 5.6, even though the maximum pressure is more than required, the average pressure is well below it, verifying the LTS's capability to meet this requirement.

Braking System

For the LTS to be agile, an efficient braking system is essential. To brake, the vehicle will exploit the ability of its DC electric motors to be used as generators [18]. This way, the LTS can harvest some of its kinetic energy, at the same time as braking and recharging its batteries.

The energy the LTS is able to recover from braking was determined by its kinetic energy and the motor's regenerative efficiency, which was estimated to be 70% [6]. This means that decelerating from its operating speed to a halt allows the LTS to recover about 6 kJ.

Though regenerative braking manifests itself as a sustainable option during nominal use of the LTS, an additional braking system is required to keep the vehicle in position when parked – whether when in hibernation or when in parking mode during explorations. To account for this, it was decided to include a "hand brake" which can be employed autonomously and via a stop button. The mechanism behind this secondary braking system would be disk brakes and will be elaborated past the 10-week mark.

[5] https://www.weg.net/catalog/weg/US/en/Electric-Motors/DC-Motor-Products/Sub-Fractional-DC-motors/ML1601-1-20-HP-12-Vcc-4P/p/14475882

[6] https://www.jdpower.com/cars/shopping-guides/what-is-regenerative-braking

5.4. Mass and Power Budgets

In this section, the mass and power budgets of the LTS' powertrain and mobility subsystem are presented. These are shown in Table 5.7, including a power measure in nominal mode and one in peak mode. Only the suspension system is expected to experience a deviation in power consumption in rare peak modes. In fact, the suspension system can experience a peak power consumption of up to 2 kW. Liberally assuming that three suspension systems could experience this peak power at a time, the total power consumption of all suspension system could rise up to 6000 kW. These instances of peak power, nonetheless, represent rare extreme impacts and were designed for in Chapter 14. It is also noteworthy that all components of the powertrain and mobility subsystem will not consume any power in the hibernation mode.

Table 5.7: Mass and power budget of the powertrain and mobility subsystem.

Component	Quantity	Mass [kg]	Power in nominal mode [W]	Power in peak mode [W]
Wheels	6	15	0	0
Shock-absorber	6	25	40	2000
Suspension rods	24	1.5	0	0
Driving Motors	6	33	330	330
Steering Motors	4	5	37	37
Structural Parts	6	8	0	0
	Total:	**542 kg**	**2368 W**	**8128 W**

5.5. Failure Modes

In this section, as part of a reliability analysis, the identified failure modes of the Powertrain & Mobility subsystem are analysed. Table 5.8 presents the failure modes per each component, along with their respective effects, prevention/response possibilities, and a criticality score, ranging from 1 - insignificant failure, to 4 - critical failure.

Table 5.8: Failure modes of the powertrain and mobility subsystem.

Component	Failure Mode	Effect	Prevention/ Response	Criticality
Wheels	Single Failure	Loss of motion of one wheel	Use remaining wheels to return to base and repair	2
	Complete Failure	Complete loss of motion	Request rescue mission	4
Shock absorber	Single Failure	Decrease in suspension performance	Return to base for repair	2
	Complete Failure	Loss of entire suspension capabilities	Return back to base for repair	4
Suspension rod	Single Failure	Insignificant	Repair at end of mission	1
	Complete Failure	Probable loss of motion	Request rescue mission	4
Driving Motors	Single Failure	Decrease in available power	Use remaining motors to return to base and repair	3
	Complete Failure	Complete loss of motion	Request rescue mission	4
Steering Motors	Single Failure	Partial loss of manoeuvrability	Return to base and repair	2
	Complete Failure	Complete loss of steering	Request rescue/repair mission	4

5.6. Design Analysis

The engineering department of Lunar industries aspired to find an innovative, adaptable, and sustainable solution in designing its powertrain and mobility system. Modularity was selected to play a big role.

The selected wheel design showcases promising prospect for all these goals. Though produced around twenty years ago, the wheel features a unique patented composite, solely developed for this application. Further, as can be seen in Figure 5.2, the wheel consists of two main components: a titanium core surrounded by a 3D-printed layered composite. A core design with replaceable spokes could help recycle them. Moreover, the three distinct 3D-printed composite layers could be produced to be individually replaced, further enabling recycling. Additive manufacturing is a process that could certainly be used on the Moon prospectively, and increased knowledge on wheel performance could motivate a slight alteration in the 3D printed pattern of the three outer layers, showcasing an overall very adaptable design. Degraded parts could be melted and used again for 3D-printing. Overall, the chosen wheel design offers a lot of modularity: an outer 3D-printed tread layer, a 3D-printed layer of hollow cylinder, a 3D printed layer of boxes, titanium spokes and a titanium core.

Though also produced around twenty years ago, the selected suspension system design manifests itself as a big innovation too, as electromagnetism is not a concept used in vehicles on Earth. Further, consisting of three distinct parts – a spring, an actuator and a magnetic system – the system is quite modular and allows for partial replacement of failing parts. The chosen system is very durable compared to the aforementioned alternatives, contributing to a very sustainable design. The use of an electromagnetic suspension system unfortunately requires the use of electronics; however, this would be the case for any other active suspension system, which was established as a necessity. Moreover, the spring will eventually degrade due to fatigue, however, it can be melted and reproduced provided the facilities will be available on the Moon. Likewise, magnets can be demagnetized, processed, purified and eventually used to create new magnets through processes like sintering. The adaptability of the suspension system is a topic that has been addressed by its working mechanism; an electromagnetic suspension system is in itself adaptive. Still, given new advancements in magnet technology, the magnetic mechanism could be rearranged to allow for even more precise alignment.

The design of the structural rods was also decided upon with sustainability in mind. The LTS includes 24 of them, four per wheel. The dynamic analysis yielded that each of them needs to support a certain load, different from the rest. To have an optimal design, each of them would then have a different design, dependent on the expected loads, however, for ease of manufacturability and sustainability, it was decided upon rendering them the same, all employing the same material and geometry.

The motors employed in the LTS also allow the system to be innovative, adaptable and sustainable. Having six independent motors makes the LTS adaptable to a wide range of terrain configurations. Also, the independence of the motors allows them to be easily repairable and/or replaceable, effectively making this subsystem more modular, and therefore more sustainable. Overall, the driving system of the LTS is not only innovative, but modular and sustainable as well.

5.7. Compliance Matrix

The engineering department of Lunar Industries has used its subsystem requirements as constraining conditions in designing the powertrain and mobility subsystem. Subsystem requirements pertaining to specific components – wheels, driving system, braking system, suspension system and steering system – have been verified in the above sections. Reflecting on each powertrain and mobility subsystem requirement, the following can be said.

Table 5.9: Compliance matrix of the powertrain & mobility subsystem.

Requirement ID [REQ-SSYS-PM-#.#.#.#.#]	Description	Method	Compliance	Justification
5.2.1.1 .1	The wheel shall be made of at least three parts	Inspection	TBD	The wheel consist of an inner titanium core, replaceable titanium spokes, a box-shaped composite layer, a cylinder-shaped composite layer and a tread composite layer, see Figure 5.2.
1.6.1.1.1	The traction system shall be able to overcome a 20 degree longitudinal incline while accelerating at 0,5 m/s2	Analysis/Demonstration	Yes/TBD	Load pertaining to this scenario was calculated in Section 5.2, and the maximum wheel load of 4500 N is sufficient. Moreover, the tread pattern was specifically designed for the lunar terrain.
1.6.1.2.1	The traction system shall allow the LTS to traverse lateral tilts of up to 20 degrees without tipping over	Analysis/Demonstration	Yes/TBD	Same rationale as for REQ-SSYS-PM-1.6.1.1.1.
1.6.1.5.1	The traction system shall enable the LTS to achieve a minimum turning rate of 20 deg/s at operating speed	Analysis/Demonstration	Yes/TBD	Same rationale as for REQ-SSYS-PM-1.6.1.1.1.
1.6.1.8.1	The traction system shall enable the LTS to achieve a minimum turning speed of 5 km/h	Analysis/Demonstration	Yes/TBD	Same rationale as for REQ-SSYS-PM-1.6.1.1.1.
1.6.1.4.3	The wheels shall be able to sustain the weight of the LTS under all manoeuvres	Analysis/Demonstration	Yes/TBD	Same rationale as for REQ-SSYS-PM-1.6.1.1.1. The prototype will demonstrate this
1.6.1.4.4	The traction system shall be able to cross step-obstacles of height of at least 15 cm	Analysis/Test	Yes/TBD	A flexible wheel with a diameter of 71cm is able to cross over obstacles with a height of 15cm, see Figure 5.2. Also the prototype will conduct the test.
2.3.1.1.1	The wheels shall not exert a pressure on the ground greater than 70 kPa	Analysis/Test	Yes/TBD	The computed average pressure is well below 70kPa./The test will be done with the prototype
5.2.1.1 .2	The suspension system shall be made of at least three parts	Inspection	TBD	Inspection will be done with the final vehicle
2.2.1.17.1	The suspension system shall allow for no more than $\pm$ 15 cm of displacement per wheel	Analysis/Inspection	Yes	A comparison was performed; the suspension system was tested to have bound and rebound strokes of 80 and 58 mm, respectively./Inspection done for prototype and before launch
2.2.1.17.2	The suspension system shall have a damping ratio between 0.20 and 0.30	Analysis	Yes	Given this range, a range of allowable damping ratios was computed in Section 5.2. The damping coefficient of 1200 Ns/m fits within the calculated range.
2.2.1.17.3	The suspension system of the LTS shall limit frequencies between 0.5 and 80 Hz	Analysis	TBD	It was detailed that active suspension systems could best limit frequencies in this range with electromagnetic ones performing best.
1.6.1.7.1	The braking system shall bring the LTS to a complete stop within a maximum distance of 5 meters	Analysis/Test	Yes/TBD	Same rationale as for REQ-SSYS-PM-1.6.1.1.1.

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Table 5.9 – *Continued from previous page*

Requirement ID [REQ-SSYS-PM-#.#.#.#.#]	Description	Method	Compli-ance	Justification
1.6.1.4.1	The powertrain & mobility subsystem shall provide the LTS with a nominal operating speed of at least 7 km/h	Analysis/Inspection	Yes/TBD	On one hand, the dynamic analysis of Section 5.2 set constraints considering this minimum speed which translated to all components designs. On the other hand, the power calculated based on this speed helped select the correct driving and steering motors.
1.6.1.4.2	The powertrain & mobility subsystem shall enable the LTS to reach its operating speed in 5 seconds in level conditions	Analysi./Tests	Yes/TBD	In the dynamic analysis of Section 5.2, this scenario resulted in lower load and motor power requirements than others, hence by designing for the limiting ones, this is accounted for. Also the test will be done with the prototype
5.7.1.1.1	Development cost of the Powertrain & mobility shall not exceed [TBD] euros	Analysis	TBD	Stakeholder meetings concerning monetary budget constraints are planned to take place after the 10-week mark
5.7.1.2.1	Manufacturing cost of the Powertrain & mobility shall not exceed [TBD] euros	Analysis	TBD	Same rationale as for REQ-SSYS-PM-5.7.1.1.1.
5.7.1.3.1	Maintenance cost of the Powertrain & mobility shall not exceed [TBD] euros	Analysis	TBD	Same rationale as for REQ-SSYS-PM-5.7.1.1.1.
5.7.1.4.1	Operational cost of the Powertrain & mobility shall not exceed [TBD] euros	Analysis	TBD	Same rationale as for REQ-SSYS-PM-5.7.1.1.1.
1.6.3.1	The LTS shall not topple over	Test	TBD	This will be verified when the LTS is produced and tested in facilities simulating the lunar environment.
2.3.1.1.2	No single point failure within the powertrain & mobility subsystem shall abort the mission	Test	TBD	Same rationale as for REQ-SSYS-PM-1.6.3.1
2.3.1.1.3	No second point failure within the powertrain & mobility subsystem shall endanger the crew	Test	TBD	Same rationale as for REQ-SSYS-PM-1.6.3.1
2.3.1.1.4	The powertrain & mobility components shall be designed to operate reliably within the expected temperature range on the lunar surface of 40K to 200K	Analysis	Yes/Testin	All components have been designed such that these temperatures are viable. All components shall be tested whether they function in this range
2.3.1.1.6	The powertrain & mobility subsystem design shall minimize the generation of lunar dust during operation	Analysis/Demonstration	Yes/TBD	All components are designed with reliability in mind, including in the presence of lunar dust in the environment. Demonstration will be done during the mission
2.3.1.1.7	The powertrain & mobility subsystem shall be designed to operate in the lunar vacuum	Test	TBD	Same rationale as for REQ-SSYS-PM-1.6.3.1
3.5.1.1.1	The powertrain & mobility subsystem shall be designed with redundancy or fault tolerance features to minimise the loss of motion probability to less than 0.001	Analysis	TBD	As described in this chapter, redundancy was well taken care of during the design of this subsystem. Specific loss of motion probability calculations are still to be done

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Table 5.9 – *Continued from previous page*

Requirement ID [REQ-SSYS-PM-#.#.#.#.#]	Description	Method	Compli-ance	Justification
1.6.1.6.1	The powertrain shall manage the energy discharge rate to stay within TBD Watts	Analysis	TBD	Detail interface of the Powertrain & Mobility subsystem and the power distribution has not been developed at this point

6 | Structures and Cabin

Authors: Jan, Aleksei, Thyme

In this chapter the designs of the structural components of the LTS are described. In Section 6.2 the design process of the cabin is described and the final result is presented. Part of the cabin is the airlock, in Section 6.3 its design process and result is presented. In Section 6.4 the design process of the frame of the LTS is presented. Finally, Section 6.5 shows the design analysis of the whole structures subsystem.

6.1. Overview

The structures chapter contains the design of the cabin, airlock and frame. The first goal of the structure is to house the astronauts in a pressurized environment large enough to fit all required amenities. secondly, it must allow access in and out of the LTS for the astronauts while ensuring their safety, maintain pressurisation and minimize dust contamination. Last but not least, the structure needs to connect all subsystems together and provide rigidity to the LTS.

Launch Loads

During launch, the loads the LTS and, therefore, the structure is subjected to are much higher than the operational loads. The launcher used is SpaceX's starship. The maximum axial load during launch is 6 g, and the lateral load is 2.5 g. The maximum operational loads are only about 2% of that. To not completely over design the structure of the LTS, it was decided to launch the LTS with temporarily additional supporting structures capable of resisting the increased loads. This means that the design described in this chapter only takes the operational load cases into account.

6.2. Cabin

In this section, the design process of the cabin will be discussed. Firstly, the cabin will be defined, including its functions. Next, the requirements and load cases the cabin is subjected to will be examined. After that, various design options and their advantages and disadvantages will be discussed. Finally, the sizing of the cabin will be detailed, and the final design will be presented.

Pressure Cabin Definition

The LTS needs to support the crew for 2 days (48 hours) as defined in stakeholder requirements REQ-STK-1.7 and REQ-STK-2.1. To support the astronauts for 48 hours a pressurized cabin is needed, since currently designed space-suits can only support a person for up to 7 hours [19].

A pressure cabin is defined as an enclosed, airtight compartment designed to maintain a stable internal pressure. Its functions are to resist the operational loads and pressure loads, maintain the internal pressure, protect the crew from the lunar environment, provide comfortable space for the crew and ensure enough space and structural integrity for other subsystems to be attached.

Cabin Load Cases

The pressure cabin has different and similar load cases as the other components in the structures subsystem. The pressure cabin should still be able to resist the launch loads in stowed configuration. Next to that, the pressure cabin is subjected to a pressure difference of 68 kPa. Finally, the pressure cabin is also subjected to the operational loads from subsystem requirement REQ-SSYS-ST-1.6.1.7.1 until REQ-SSYS-ST-1.6.1.7.3. It was found that the pressure load was leading, and therefore the pressure cabin was sized for the pressure difference.

Cabin Shape

To fit all required subsystems inside the cabin, it was found that the cabin needs to have dimensions of at least 3.9 x 2.5 x 2.1 cubic meters (l x w x h). To attach the cabin to the other structural components, the bottom part of the cabin needed to be flat and 2.5 meters long. The other sides were made circular as this is better at resisting the pressure. All corners should be made round to limit stress concentrations. The final shape of the cabin can be seen in Figure 6.1.

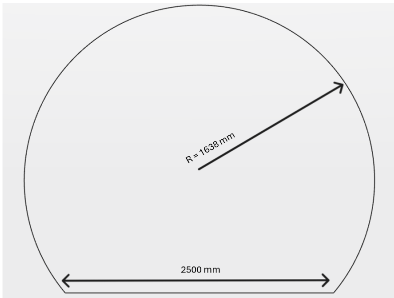


Figure 6.1: Cross section view of the Pressure cabin

Cabin Sizing

Since the dimensions are set and the shape of the cabin has been defined, the only thing left to size the pressure cabin for is the thickness. To size for the thickness it was assumed the cabin is a cylinder. From pressure loads, the hoop stress for a cylinder is the highest and therefore the cabin is sized for that. The equation for the hoop stress is given in Equation 6.1.

$$\sigma_{hoop} = \frac{\Delta P \cdot R}{t} \tag{6.1}$$

The equation to size for the thickness is then the following:

$$t = \frac{\Delta P \cdot R}{\frac{\sigma_{yield}}{SF \cdot SCF \cdot UF}} \tag{6.2}$$

Here, SF stands for Safety Factor, SCF stands for Stress Concentration Factor and the UF stands for Uncertainty Factor. The safety factor that was used is 2 and comes from NASA standards NASA-STD-5001B [20]. The SCF used will be discussed later. Finally, an Uncertainty Factor of 2 was added, since at the back of the LTS, a hole was made for a window on the airlock. When depressurised the back of the pressure cabin is no longer in the shape of a cylinder, and calculating the loads caused by the pressure proved challenging. The fact that the cabin cross section is not completely cylindrical also adds uncertainties. Therefore, an Uncertainty Factor of 2 was added, but it should be researched and designed more thoroughly in the future.

To decide on a thickness, a material needed to be chosen since the yield stress is needed. A trade-off was done for the material choice. The following criteria were considered: mass, cost and (in-situ) manufacturability. For the latter criterion, materials were chosen which have the potential to be made on the moon. These were found to be: Aluminium alloy (AlSi10Mg), Titanium alloy (Ti-6Al-4V) and stainless steel (AISI 316L). The mass criterion has been given a weight of 1, since there is a limiting mass but it is very high (1̃00 tons) as set by system requirement 5.7.1.5 flowing from the launcher choice. The cost criterion has been given a higher importance (weight 3) than mass, since it is linked to sustainability as well. (In-situ) Manufacturability has been given the most importance (weight 5). The result of the trade-off was that AlSi10Mg is the best material from the materials considered, therefore this will be used for the pressure cabin. AlSi10Mg has a yield stress of 250 MPa [21], resulting in a thickness of 3 mm. Finally, with this thickness, the pressure cabin's mass was calculated to be 445 kg.

Coming back to the Stress Concentration Factor, at the back of the pressure cabin, there is a door to the airlock. Stress concentrations occur around the cutout for the door, and therefore the material around the cutout should be reinforced. One way is to make the material thicker. This extra thickness can be calculated by using the stress concentration factor, which can be as high as 8. If only the material is made thicker and no other reinforcing methods are applied, the thickness around the door will increase to roughly 2 cm, significantly increasing the weight at the back. It requires further research in a later design stage to design this reinforcement.

6.3. Airlock Subsystem

In this section, the design process of the airlock subsystem will be discussed. First, the airlock shall be defined, then the possible airlock concepts are presented, and a trade-off is done. With the chosen airlock concept, the sizing process will be explained, and finally, the complete airlock subsystem assembly will be presented.

Airlock Subsystem Definition

The airlock subsystem is defined as a room which separates two areas which have different pressures. Similarly to the pressure cabin, the need for an airlock subsystem in the LTS flows down from REQ-STK-1.7 and REQ-STK-2.1 and the mission need statement. Due to the lack of an atmosphere on the Moon, a very low pressure [22] and the need for human exploration, a pressure cabin was required. From this, also follows that an airlock is required, otherwise entry and exit from the pressure cabin cannot be done safely.The functions which are essential to an airlock system are:

- Ensure access to fully suited passengers;
- Ensure (de)pressurisation of the airlock volume;
- Ensure for storage of 2 astronaut suits;
- Ensure for a thermally insulated door, connecting the airlock to the lunar environment;
- Ensure for radiation shielded door, connecting the airlock to the lunar environment;
- Ensure for a small window in the door, according to NASA standards [23], connecting the airlock to the lunar environment.

Airlock Concept Trade-off

Multiple airlock concepts for space have been designed from 1965 onwards. Most contemporary airlock concepts in use are rigid pressure vessels. However, this is not the only type of airlock concept feasible to use. In Table 6.1, the possible airlock types are presented:

Table 6.1: Airlock design concepts

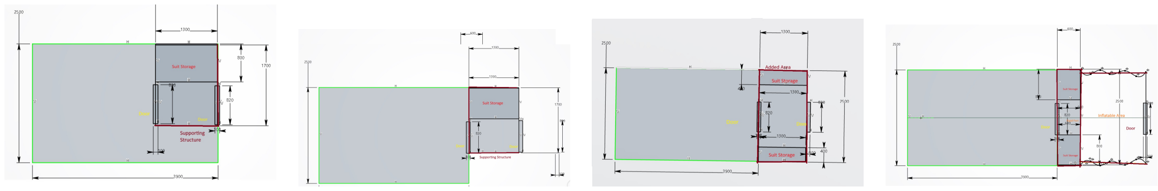
Type/ Chamber	Definition	Advantages	Disadvantages
Rigid	Stiff, non-flexible structure	Low complexity, possible to integrate into cabin, applicable for LTS use, low maintainability, high TRL	Volumetric constraint
Inflatable	Retractable, morphing, flexible structure	Low operational volume, low launch volume, applicable for LTS use, Volumetric not constrained	High Complexity, high maintainability, low/medium TRL
Integral	Combined pressure cabin and airlock	No additional airlock volume, no additional airlock mass	No cleaning room; cabin becomes contaminated, not applicable for LTS use, long (de)pressurisation time
(Rigid) Dual	Stiff, non-flexible structure with separated suit changing room	Low (de)pressurization time, low complexity, applicable for LTS use, high TRL	Needs more volume than rigid chamber, needs more mass than rigid chamber, volumetric constraint

For the LTS airlock system required, the main driving factor in choosing the airlock is to have it occupy the smallest possible volume. This is because when the volume is limited, the mass of both, the airlock system as all other structural supports, will also be minimised.

The integral chamber will not be considered in the following trade-off because it conflicts with REQ-1.7; it is essential to limit the amount of dust going into the pressure cabin, which has an extremely low likelihood with an integral chamber type. Therefore, it could be concluded that this chamber type is not suitable for low gravity exploration vehicles. Furthermore, the rigid dual chamber type will also not be taken into further consideration because the expected total volume will always be larger than for a single rigid airlock. This is because a separate airlock compartment is required in which all the same operations are to be performed.

Figure 6.2 presents the considered layouts for the airlock subsystem. In the following list, each layout will be briefly discussed:

- Figure 6.2a presents the first rigid layout concept, which is fully integrated into the pressure cabin. While this is ideal for volume optimisation and limiting the additional length of the LTS, internally, a more complicated load case does occur. To counter this additional support, would need to be added locally, to prevent stress concentrations from occurring.
- Figure 6.2b presents the second rigid layout concept, which is completely connected outside of the pressure cabin. While this concept is ideal for creating a modular airlock system, it will require a significant increase in total mass of the LTS due to the essential supporting structure.
- Figure 6.2c presents the third rigid layout concept, which is completely connected to the outside of the pressure cabin. While this layout is not volume optimised like the first and second configurations, it does allow for a limited additional length. Also this layout would allow for the potential of a modular airlock design.



(a) volume optimised rigid chamber integrated into pressure cabin

(b) volume optimised rigid chamber, externally connected to pressure cabin

(c) symmetric rigid chamber, externally connected to pressure cabin

(d) Inflatable chamber, externally connected to pressure cabin

Figure 6.2: Airlock type configurations

- Figure 6.2d presents the only considered inflatable airlock concept. While there are several inflatable airlock concepts being developed, this concept proved to be one of the few to be applicable for the lunar environment [24]. Similar to the DCIS, this concept is made out of flexible fabric structure, which upon pressurisation, enhances the airlock volume [24]. While this concept proves to be ideal for limiting volume needed for the the airlock in stowed configuration, it still needs a significant volume for the EVA suits to be stored. Next to that, the TRL level of flexible inflatable structures, capable of withstanding radiation and dust wear, is not very high.

Due to the fact that the airlock concept is mainly driven by volume optimisation, and that the TRL of inflatable airlocks for permanent use in highly corrosive environments is rare, concept 1 was chosen.

Airlock Sizing

While the aim is to minimise the airlock volume, is the critical driving factor, the sizing of the three dimensional airlock is based upon multiple other requirements which need to be adhered to. Combining all of these requirements resulted in the selected airlock concept, presented in the next section, in which the assembly of the airlock will be shown.

Ergometric Requirements

Requirements REQ-SSYS-ST-2.4.1.1.1 to REQ-SSYS-ST-2.4.1.1.6 present the ergometric requirements which drive the airlock design according to NASA standards [23]. These ergometric requirements are requirements based on the allowable movements of the crew inside the airlock. The main dimensional constraints derived from these requirements are the following:

- The minimum airlock door dimensions (820mm x 1900mm);
- The minimum airlock height (2000mm);
- The minimum unconstrained area necessary for one crew member to put on the EVA suit (1000mmx1000mm);
- The cabin-airlock door shall be able to turn a minimum of 90 deg inwards into the airlock.

Suit Storage

It is essential for two complete astronauts to fit inside of the airlock subsystem, while they are wearing their suits. The astronaut suit used for sizing the airlock subsystem is the Artemis spacesuit prototype, the AxEMU [1], presented in Figure 6.3. Due to the lack of information on the dimensions of this suit, a combination of known suit dimensions and extrapolation was

[1]https://www.nasa.gov/humans-in-space/spacesuit-for-nasas-artemis-iii-moon-surface-mission-debuts/

used to estimate the dimensions. The following list presents the assumptions made on the suit dimensions:

Figure 6.3: the Artemis III spacesuit prototype, the AxEMU

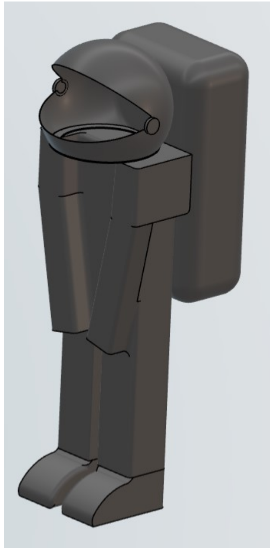


Figure 6.4: Extrapolated stowed suit assembly

- The stowed suit configuration is 25-30 % smaller in width only because it is assumed modern space-suits are semi-flexible;
- It is assumed that the suits are comfortable for people of a maximum height of 190 cm;
- It is assumed that the shoes of the suit are detachable;
- It is assumed that the backpack is fully rigid and detachable;
- It is assumed that the helmet is fully rigid and detachable;

Using the above mentioned assumptions, the extrapolated AxEMU suit used to size the airlocks is shown in Figure 6.4. The final volumetric constraints from adding two astronaut suits into the cabin are presented in the following list:

- 2x Stowed suit with maximum dimensions of 600 mm x300 mm x1384 mm;
- 2x Set of boots with maximum dimensions of 450 mm x 388 mm x 162.2 mm ;
- 2x Backpacks with maximum dimension of 600 mm x 350 mm x 950 mm;
- 2x Helmets with maximum dimensions of 574 mm x 450 mm x 450 mm.

Airlock Subsystem Assembly

Figure 6.5 presents the airlock assembly used for the LTS. The following list discusses some main features of the airlock system:

- Both doors have an aluminium frame ensuring an airtight connection. The left door frame (connecting the LTS to the lunar environment) has a higher thickness than the right door frame because it also contains thermal insulation ;
- The curvature of the airlock cabin matches the curvature of the thermal insulation layer to which it is attached;
- The area of the airlock system is sized to be 1300 mm x 1700 mm such that all previously mentioned constrains are adhered to;
- The tilted suit is attached to the wall and the helmets and upper backpack are attached onto a shelf.

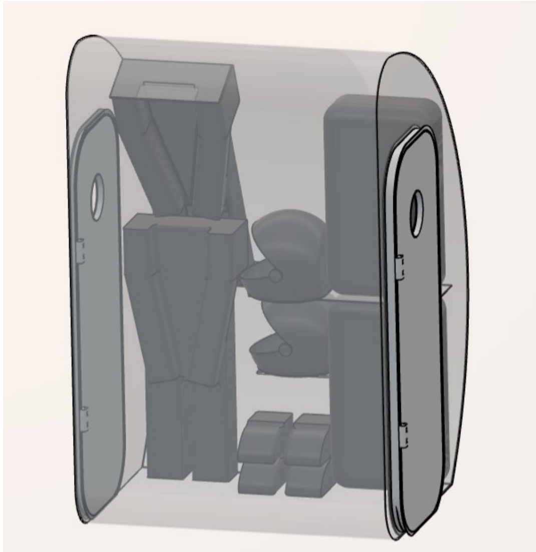


Figure 6.5: Airlock Assembly

6.4. Frame

The function of the frame is to connect the cabin to the suspension below it and to provide rigidity to the whole assembly.

Relevant Loads

As the LTS performs various manoeuvres, it subjects itself to a lot of different load cases, all of which the frame needs to be able to deal with. To start designing the frame the main loads need to be identified:

- Weight of the cabin shell: This weight is introduced into the frame of the LTS along its outside perimeter where the shell rests on the frame and the iso-grid is connected to. This results in a large distributed load.
- Weight of the cabin floor: The weight of the cabin floor and interior forms a distributed load over the area within the perimeter.
- Wishbone suspension loads: The wishbones are responsible for transferring the horizontal loads imposed on the wheels into the frame. These loads are mainly induced by the friction forces acting on the wheels during braking or turning manoeuvres.
- Shock/spring vertical loads: The vertical loads imposed on the wheels during driving are mainly transferred into the frame at the 6 attachment points for the shocks/springs directly above each wheel.

For each manoeuvre the LTS is expected to perform, the imposed loads on each wheel are calculated in the longitudinal and lateral direction. These loads form the basis for the requirements of the frame.

Design Direction

The various manoeuvres during driving induce a lot of complicated loads onto the frame which all need to be accounted for.

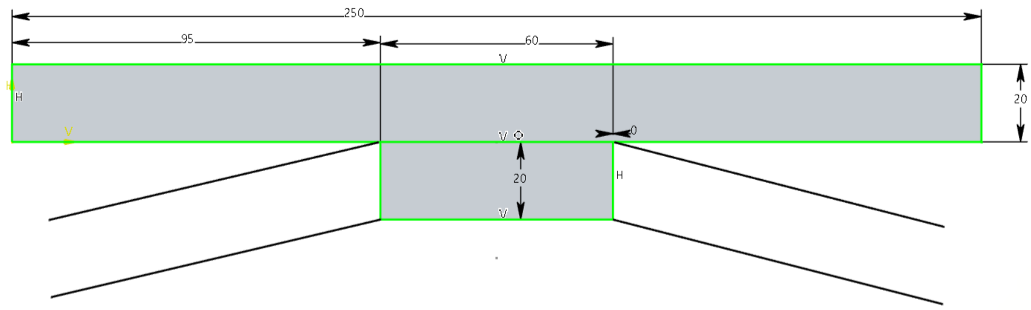


Figure 6.6: Envisioned frame design space in cm

Figure 6.6 shows a sketch of the envisioned design space of the LTS frame. The gray area shows the main dimensions of the frame in cm, additionally the wishbones are represented in black. The upper gray box is represents the allowed design area for the main load carrying structure, the lower gray box represents the design space for where the suspension wishbones can connect too.
The gray area is entirely available for structural component. The area besides the gray box where the suspension rods are is only available as long as it does not interfere with the wishbones, as these move up and down when the suspensions are compressed.

Generative Design
For a complicated load case, such as this one, many structural solutions can be conceived. The first attempt to solve this ambiguous problem was to use generative design to conceive a concept for the frame geometry. Generative design in 3D experience uses the external loads applied on points or surfaces to create a structure within the set allowed design space which would be optimised for the imposed loads.

Unfortunately, due to some limitations of the software available, the conceived frame concept proved unusable. However, some of the structural elements that the software created were carried over to the next design

Conventional Design
As the topology method proved unfeasible, a more conventional design process was used. The first design contained the following features:

- The outer perimeter was envisioned as a rectangle of I-beams as these are ideal for transferring the vertical distributed loads of the cabin shell to the shock absorbers connected to the wheels.
- The smaller box from Figure 6.6 is added with diagonals to provide rigidity. This box is placed six times in total, one at each of the connection points of the suspensions.
- Vertical webs are added to connect the boxes to the main part of the frame.

Structural Iteration
The first iteration of the frame was based on a few general design decisions but was not optimised for the actual loads imposed on the structure. In order to optimize the structure, 3D experience tools were used to calculate the stresses and deformations present in the structure under the loads. These values were in turn used to alter the conceived structure.

Two types of changes were made to address the results of the simulation.

- Bases on the stress value, the local thickness of the element was either made thicker or slimmer.
- Elements for which the deformations were too big, either stiffeners or connections to neighbouring elements were added.

The conceived structure was iterated by repeating this procedure for over 10 cycles while making small changes each time.

It should be noted that for this method of iteration, the frame does not make fundamental changes; each cycle improves the design slightly, but it does not alter the general geometrical idea behind the design, thereby limiting the final design to something relatively similar to the initial design. For a more optimized end result the iteration process should be performed for several starting concepts/philosophies. Eventually the best of these options could then be selected. However, due to time constraints only one initial concept was considered.

Finalized Frame

The finalized frame is presented in Figure 6.7; the frame is shown from the bottom to show the relevant structural components.

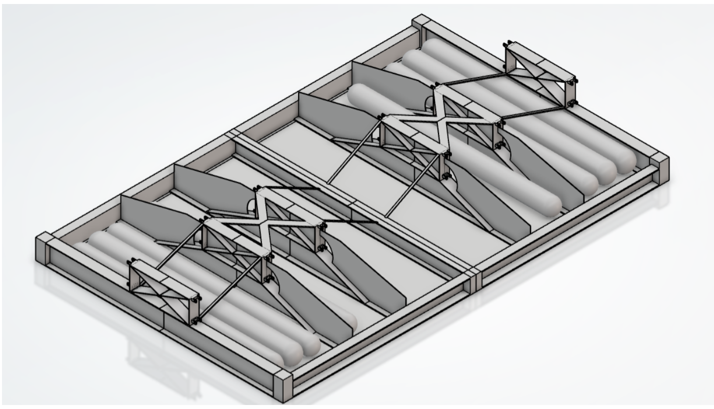


Figure 6.7: Bottom view of LTS frame

The frame consists of an outer perimeter of I beams with local reinforcements to which the shock absorbers are attached. Lateral webs are added which resist the sideways loads coming from the wishbones. The webs are cut short on the top and have cut outs to allow space for the suspension rods to attach to. The H_2 and O_2 tanks are placed between the lateral webs and will be covered in insulation. The remaining space between the webs houses the fuel cells and could in the future be used to increase the reactant storage to increase range and duration.

Design Verification

The same tool as used for the structural iteration also doubles as the verification tool for the conceived structure. The structure is tested by inputting all the different load cases and the finalized structure into the static solver and verifying that the stresses on the structure are within limits everywhere.

6.5. Design Analysis

The following three sections show some of the design characteristics.

Sustainability
One way in which the LTS's structure is sustainable is that it does not use materials from Earth for its radiation shielding but uses regolith, this significantly reduces the weight that needs to be transported to the moon.

Modularity
The LTS design is modular in the sense that the frame, cabin, fuel tanks and wheel assemblies are wholly separate and can be separated relatively easily. This means that as the lunar market develops all of these components could relatively easily be redesigned and the revision implemented without having to replace the whole LTS.

Recyclability
The LTS structure is designed with mostly recyclable parts, the frame and pressure cabin are made out of aluminium which can easily be repurposed at EOL. Additionally, the frame and cabin are fully detachable, so if either part has become obsolete or damaged beyond repair the other can easily be repurposed for other projects.

6.6. Mass budgets
In Table 6.2, the mass budget of the main structure is shown.

Table 6.2: Mass budget of the main structure

Component	Mass [kg]
Pressure cabin	445
Supporting structure	220
Airlock	350
Frame	300
Total	**1315**

6.7. Failure Modes
In Table 6.3, the failure modes relevant to the structures part are presented with their effects, prevention/response and criticality score.

Table 6.3: Failure Modes Analysis of Structures Subsystems

Component	Failure mode	Effect	Prevention/response	Criticality
Pressure cabin	Complete failure	Depressurization of cabin	Use patch kit and close the hole, alternatively put on the suits	4
Surrounding structure	Single failure	Part of the radiation shielding is lost	Return to base and repair	1
	Complete failure	All of the radiation shielding is lost	Put on space suit and return to base	2
Frame	Single failure	Loss of functionality of one (set of) wheel	Return to base for repairs	1
	Complete failure	Total loss of structural rigidity and mobility	Call for help and limit power consumption	4
Airlock when inside	Complete failure	Can not exit the LTS	Return to base for repairs	2
Airlock when outside	Complete failure	Can not enter the LTS	Call for help	4

6.8. Compliance Matrix

Table 6.4: Compliance matrix of the Structures Subsystem

Requirement ID [REQ-SSYS-ST-#.#.#.#.#]	Description	Method	Compli-ance	Justification
1.2.1.1.1	The MS shall be able to support a cargo payload of up to 50 kg	Testing	TBD	Parts are designed for this requirement but not yet tested
1.2.1.1.2	The loaded and unloaded cargo subsystem shall not have a natural frequency which resonates with any frequencies present in the LTS	Analysis	TBD	All possible loaded cargo configurations have not yet been analysed
1.6.1.7.1	The structure shall resist a max positive x-axis acceleration of magnitude 0.93 m/s^2	Analysis	Yes	3D experience stress calculations indicate compliance
1.6.1.7.2	The structure shall resist a max negative x-axis acceleration of magnitude 1.05 m/s^2	Analysis	Yes	3D experience stress calculations indicate compliance
1.6.1.7.3	The structure shall resist a max y-axis acceleration in either direction is 1.035 m/s^2	analysis	Yes	3D experience stress calculations indicate compliance
2.2.1.1.1	The airlock door shall be able to have a cut-out for the window of [0.025] m^2	Inspection	Yes	The airlock is designed for this requirement
2.2.1.1.3	The Pressure Cabin shall be able to resist a pressure load of 68 kPa	Testing	TBD	Calculations indicate compliance but test have not been perfromed yet
2.2.1.17.1	The maximum vibrational acceleration shall be within 10^{-2} to 10^{-1} m/s^2 RMS	Testing/Inspection	TBD/TBD	Testing/Inspection will be done by prototype

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Table 6.4 – *Continued from previous page*

Requirement ID [REQ-SSYS-ST-#.#.#.#.#]	**Description**	**Method**	**Compliance**	**Justification**
2.2.1.17.2	The natural frequency of the MS shall be higher than 80 Hz	Analysis	TBD	3D experience software indicates compliance for all tested part, however some parts do not have set stiffness yet
2.3.1.1.1	The structure shall be designed safe life until EOL	Analysis	TBD	Fatigue test have not yet been performed
2.4.1.1.1	Hatches and doors shall be operable on either side by a single crew member without the use of tools in expected gravity conditions, orientations, suit configurations, and operational configurations, according to the NASA-STD-3001 standards	Inspection	TBD	Airlock is designed for this but test have not yet been performed
2.4.1.1.2	Hatches shall require two distinct and sequential operations to unlatch, according to the NASA-STD-3001 standards	Inspection	Yes & TBD	Airlock has been designed according to this requirement, This will also be inspected to
2.4.1.1.4	The forces required to operate each crew interface for the hatches and doors shall be smaller than 54 N for one arm pushing movements, according to the NASA-STD-3001 standards	Testing	TBD	Design indicates compliance but has not yet been tested
2.4.1.1.5	The forces required to operate each crew interface for the hatches and doors shall be smaller than 49 N for one arm pulling movements, according to the NASA-STD-3001 standards	Testing	TBD	Design indicates compliance but has not yet been tested
4.1.1.1.1	In stowed configuration, the structure shall be able to withstand axial launch loads of 6 g	Testing	TBD	Tests have not yet been performed
4.1.1.1.2	In stowed configuration, the structure shall be able to withstand lateral launch loads of 2.5 g	Testing	TBD	Tests have not yet been performed
4.2.1.1.1	The structure shall be able to be assembled on the Moon within a time of 14 days	Demonstration	TBD	Demonstration has not yet been performed
5.1.1.3.1	The structure shall be able to be O2 corrosion resistant till EOL	Analysis	Yes	Chosen materials indicate compliance
5.1.1.3.4	The structure subsystems shall have an operational life of at least 10 years	Analysis	TBD	Fatigue test have not yet been performed
5.2.1.1.1	The MS shall allow external mounting of cargo payload of up to 50 kg	Inspection/Demonstration	Yes/TBD	Supporting structure is designed in compliance with this requirement. This will also be demonstrated by the prototype
5.2.1.1.2	The MS shall allow for the external mounting of a cargo payload with the volume of 135 liters	Inspection	Yes	Cargo hold is designed in compliance with this requirement
5.2.1.1.3	The pressure cabin shall be able to be mounted to the main structure	Inspection	Yes	Design of both parts indicate compliance
5.2.1.1.4	The radiation protection shall be mounted to the supporting structure	Inspection	Yes	Supporting structure is designed in compliance with this requirement
5.2.1.1.6	The locomotion subsystem shall be mounted to the main structure	Inspection	Yes	Both parts have been designed for compatibility
5.2.1.3.1	The main structure shall be fully inspectable	Inspection	Yes	Main structure has been designed in compliance with this requirement

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Table 6.4 – *Continued from previous page*

Requirement ID [REQ-SSYS-ST-#.#.#.#.#]	Description	Method	Compli-ance	Justification
5.2.1.3.2	The PC shall be fully inspectable	Inspection	TBD	Inner and outer panes are designed to be removable for inspection. This will be shown in the prototype
5.5.1.1.1	At EOL, the main structure shall be able to be recyclable to standards in Section 21.4	Analysis	Yes & TBD	See Section 21.4
5.5.1.1.2	At EOL, the pressure cabin shall be able to be recyclable to standards in Section 21.4	Analysis	Yes & TBD	See Section 21.4
5.5.1.1.3	At EOL, the airlock system shall be able to be recyclable to standards in Section 21.4	Analysis	Yes & TBD	See Section 21.4
5.7.1.1.1	Development cost of the structure shall not exceed 1,000,000,000 euros	Inspection	TBD	Individual component cost have not been estimated yet
5.7.1.2.1	Manufacturing cost of the structure shall not exceed 100,000,000 euros	Inspection	TBD	Individual component cost have not been estimated yet
5.7.1.3.1	Maintenance cost of the structure shall not exceed 10,000,000 euros	Inspection	TBD	Individual component cost have not been estimated yet
5.7.1.4.1	Operational cost of the structure shall not exceed 1,000,000 euros	Inspection	TBD	Individual component cost have not been estimated yet

7 | Life Support

Author: Thomas

In this chapter the life support system of the LTS will be described. In Section 7.1 a general overview will be given of the life support system. After that, in Section 7.2 - Section 7.6, the subsystems of the Life Support system will be described. Subsequently, in Section 7.7, the mass and power budgets of the life support system will be given. Then, in Section 7.8, the life support components failure modes will be described. Next, in Section 7.9, the sustainable aspects of the life support system will be described. Finally, in Section 7.10, the requirement compliance matrix will be given.

7.1. Overview

The life support system of the LTS, depicted in Figure 7.1, has the function of keeping the crew alive when it is using the LTS, and of making sure that the atmosphere is comfortable to work in. In order to do this, the life support system consists of various subsystems: the pressure control system (Section 7.2), which manages the pressure inside the cabin; the air revitalization system (Section 7.3), which manages the humidity, the level of CO_2 and the trace contaminants in the system; the waste management system (Section 7.4), which manages the waste produced by the crew, including urine and feces; the food and water management system (Section 7.5), which manages the water used by the crew, and the fire detection and suppression System (Section 7.6), which detects fires and enables the crew to suppress them.

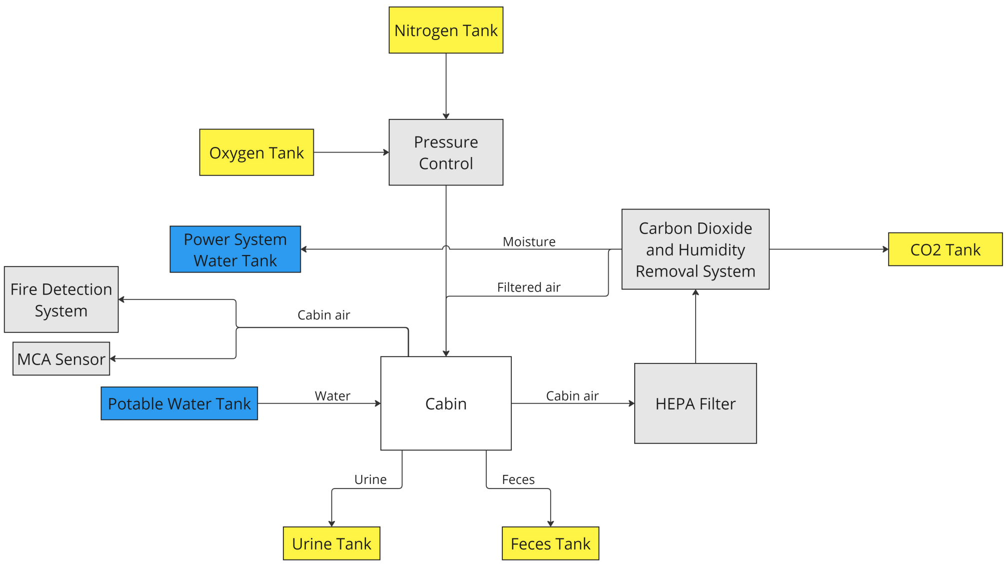

Figure 7.1: General overview of the life support system component and functions

7.2. Pressure Control System

The main functions of the pressure control system (PCS) are to monitor the total pressure, and the O_2 and N_2 partial pressures, and to maintain these pressures to ranges which make the cabin habitable to the crew. Before the crew enters the LTS, it will be pressurized to 68 kPa, consisting of 32% oxygen and 68% nitrogen. These atmospheric parameters were chosen in order to limit the pre-breathe time required to be able to perform an EVA – the pre-breathe time is the time the astronauts have to breathe pure oxygen to prevent decompression sickness. With these atmospheric parameters, the pre-breathe time is decreased to only one hour, while maintaining the pressure levels defined in the requirements [25].

After this initial pressurization, the PCS has to monitor and maintain the pressure levels in the cabin. The PCS will use a pressure sensor to monitor the total pressure in the atmosphere. In order to monitor the oxygen and nitrogen levels in the atmosphere, it will use the Spacecraft Atmosphere Monitor (S.A.M.) which can measure the nitrogen and oxygen partial pressures [26]. To be able to maintain the pressure levels in the cabin, the PCS uses the nitrogen and oxygen stored onboard of the LTS. The oxygen is stored in the same tanks that will be used for the power generation. The nitrogen is stored in a separate tank. The pressure at which the nitrogen will be stored is 200 bar. The tanks will contain enough oxygen for breathing and enough oxygen and nitrogen to be able to re-pressurize the cabin once. So, the amounts of oxygen and nitrogen that need to be stored are 15.3 and 17.4 kg, respectively. At 200 bar, this means that a nitrogen tank volume of about 76 L is required. The nitrogen tank was sized using the method which will be explained in Section 14.3, and the material it is made of is T1000.

7.3. Air Revitalization System

The Air Revitalization System (ARS) will perform several important functions. Firstly, it will monitor and control the level of carbon dioxide in the cabin atmosphere. Furthermore, it will remove the humidity which is produced by the crew and the equipment inside the LTS. It will also monitor and remove trace contaminants from the cabin atmosphere. The ARS monitors the CO_2 level and the level of certain trace contaminants with the same sensor that will be used to monitor the oxygen and nitrogen levels, namely the Spacecraft Atmosphere Monitor (S.A.M.) [26]. In order to monitor the humidity level, the ARS will use a humidity sensor.

There air will be ventilated around the cabin by the ventilator which is part of the thermal control subsystem (see Chapter 13). The ventilation rate can be controlled by changing the rotating speed of the ventilator. To remove the airborne particulate matter and airborne microorganisms, a HEPA filter will be used. More specifically the advanced media HEPA filter from Paragon Space Development was selected to be employed. This filter has been shown to remove up to 99.97% of lunar dust particles starting from a size of 0.1 µm [27]. Next to this, the filter also removes other airborne particulate matter and microorganisms at or above an efficiency of 99.97% at 0.3 µm, as is typical for a HEPA filter.

After the air has passed over the HEPA filter, the excess carbon dioxide and water vapor will be removed. In order to do this, the Carbon Dioxide Removal by Ionic Liquid System will be used [28]. As the name indicates, this system will use ionic liquid to remove carbon dioxide and water vapor from the cabin air. The selected ionic liquid is liquid at room temperature, nonflammable, and non-toxic. The advantage of this system over solid adsorbent-based approaches is the fact that these approaches require a complicated valve system to switch the beds between adsorption and desorption modes, however, the circulating liquid in CDRILS allows the scrubber and stripper to have fixed roles. Furthermore, since the liquid can be rapidly moved between adsorption and desorption, less adsorbent is required, reducing the required weight and volume [28]. The carbon dioxide will be stored in a tank in order to be able to convert it into water and methane back at Basecamp. The amount of CO_2 which will be produced by the crew members during the use of the LTS is 4.16 kg [29]. The CO_2 will be stored at 200 bar, this means that about 11.5 L of tank volume is needed. This tank was sized using the same method as the nitrogen tank (see Section 14.3) and will be made of the same material. For the water vapor, no separate tank will be provided in the LTS; this water vapor will be stored in the power generation water tanks. The total amount of water vapor which needs to be stored is 5.72 kg [29].

In order to choose a system that can remove trace contaminants, such as ammonia, methane and ethanol, the generation of these trace contaminants was modeled [30]. The final concentration values at the end of the two days were compared to the Spacecraft Maximum Allowable Concentrations (SMAC) as defined by NASA [31]. The result of this analysis can be seen in Table 7.1

Table 7.1: Trace contaminant amounts compared to maximum allowable amount [30] [31]

Trace Contaminant	Maximum allowed amount (mg)	Maximum amount after 2 days (mg)
Acetaldehyde	132.4	2.43
Acetone	1721.2	77.08
Ammonia	66.2	200.02
Benzene	49.7	8.8075
n-Butanol	2648	3.4
Carbon monoxide	2085.3	72.6
Ethanol	66200	19.5
Formaldehyde	3.97	1.6
Furan	2.32	1.20
Hexamethylcyclotrisiloxane	2979	0.05
Hydrogen	11254	168
Methanol	861	3.99
Methane	125780	1316.19
Methylene chloride	1622	1.02
Toluene	497	3
Trimethylsilanol	132	0.05
Xylenes	2416	1.91

As can be seen from Table 7.1, only the ammonia amount in the cabin exceeds the maximum allowed amount. This could lead to eye irritation and headaches for the crewmembers. However, the CDRILS, which will be used for removing carbon dioxide and humidity, also has some capability to remove trace contaminants. In the case of ammonia, the CDRILS can remove 150 mg/day, which is enough to reduce the amount of ammonia in the cabin below the maximum allowed amount [32]. Based on this analysis, it was chosen to not have a separate trace contaminant removal system inside the LTS. The cabin air does have to be pumped through the trace contaminant removal system which will be present at Basecamp after each use of the LTS. That way, the trace contaminants will not exceed the SMACs.

7.4. Waste Management System

The function of the waste management system is to store and manage all waste produced by the crew while it uses the LTS. The body waste management system that will be used on the LTS is the Universal Waste Management System (UWMS). The UWMS is currently used on the ISS and is planned to also be used on the Orion spacecraft from the Artemis program. The urine will be transported through the hose of the UWMS into the urine tank. This tank will be able to store up to 28 L of urine. Additionally, the LTS will be able to store up to 16.3 L of feces and menses in a separate tank.

7.5. Food and Water Management System

The Food and Water Management System (WMS) will manage all the food and water the crew will use during the time they spend in the LTS. This system is crucial in enabling the crew members to maintain optimal working performance. In Table 7.2 you can see the water budget per crew member per day. This table has been based upon subsystem requirement REQ-SSYS-LS-2.2.1.9.1-REQ-SSYS-LS-2.2.1.9.5.

Table 7.2: Water budget of a crew member, per day

Use category	Amount of water (L)
Hydration	2.5
Personal hygiene	0.4
Eye irrigation	0.5
Medical use	5 (per event)
EVA	0.24 (per hour)

Assuming one medical event and six hours of EVA per crew member for the two days, this results in a total water requirement of 26.48 L. This water will be stored in a tank inside the cabin. The tank will contain a heating and cooling element, as well as sensors to monitor the quality of the water. In order for the crew to access the water, the tank will be connected to a tap in the cabin. As defined in REQ-SSYS-LS-2.2.1.14.1 and REQ-SSYS-LS-2.2.1.14.2, the life support system shall provide the crew with an average of 3035 kcal of food per day, in addition to 200 kcal per EVA hour. So, the life support system shall provide the crew with a total of 14540 kcal over the two days the LTS will be used. This food will be stored in a cupboard in the cabin.

7.6. Fire Detection and Suppression System

If there is a fire in the cabin, this poses a life threatening risk to the crew. In order to minimize this risk, the LTS will have a fire detection and suppression System (FDSS). To be able to quickly detect fires when they occur, the FDSS will use two types of sensors. The first type of sensor that will be used is a smoke detector. Next to a smoke detector, the FDSS will use a tunable laser absorption spectrometer (TLAS) to detect whether the combustion products CO, HCN, HCl and HF are in the air. The sensor that has been chosen has the required detection ranges [33]. When a fire has been detected, the crew should have access to the required equipment to suppress the fire. The equipment that will be used is a fire extinguisher which is filled with pure CO_2. For redundancy two fire extinguishers will be onboard at all times. Furthermore, the LTS will have first aid medical supplies onboard in case of injury.

7.7. Budgets

In this section, the mass, power and cost budgets of the life support system of the LTS will be given. In Table 7.3, you can find the mass budget. For CDRILS, the only available weight information is for a system designed for 4 crew members [34]. So, in order to scale it down to a system for 2 crew members, the weight was divided by two and a 20% margin was added. This results in the weight which can be found in the table. In Table 7.4, you can find the power budget for the Life Support system. To determine the power consumption of the CDRILS, the same approach as the one used for the weight has been followed.

Table 7.3: Mass budget life support system

Component	Mass (kg)
Water tank	5
Water	26.5
Urine tank	5
Feces tank	7.5
UWMS	52
Nitrogen tank	4.6
CO_2 tank	0.6
CDRILS	140
Spacecraft Atmosphere Monitor [26]	9.6
TLAS sensor [33]	5
Fire extinguisher [35]	6.8
Total	**262.6**

Table 7.4: Power budget life support system

Component	Power (W)
UWMS	235
CDRILS	791
Spacecraft Atmosphere Monitor [26]	20
TLAS sensor [33]	42
Total	**1088**

7.8. Failure Modes

In this section the failure modes of the life support subsystem will be described, as well as their effect, response and criticality level.

Table 7.5: Failure modes of the life support subsystem.

Component	Failure Mode	Effect	Prevention/Response	Criticality
Water tank	Single Failure	Only pure water without any minerals available	Use the water tank of the power subsystem	1
	Complete Failure	No more water for the crewmembers	Return back to base for replacing of tank	3
Urine tank	Complete Failure	Leakage of urine into cabin, no more urination possible	Return back to base for replacing of tank	3
Feces tank	Complete Failure	Leakage of feces into cabin, no more defecation possible	Return back to base for replacing of tank	3
UWMS	Complete Failure	No more urination and defecation possible	Return back to base for repair	3
Nitrogen tank	Complete Failure	No nitrogen supply for pressure control	Return back to base for repair	4
Oxygen tank	Single Failure	Less oxygen on board for breathing	Use one of the backup tanks	1
	Complete Failure	No oxygen supply for breathing or repressurization	Return back to base for repair	4
CO_2 tank	Complete Failure	Storage of CO_2 no longer possible	Vent removed CO_2 overboard	3
CDRILS	Complete Failure	No CO_2 and humidity removal from the cabin	Return back to base for repair	4
Spacecraft Atmosphere Monitor	Complete Failure	No data on major and trace atmospheric consituents	Return back to base for repair	3
TLAS sensor	Complete Failure	No data on toxic combustion products	Return back to base for repair	3
Fire extinguisher	Single Failure	Insignificant	Use the backup fire extinguisher	1
	Complete Failure	Loose fire extinguishing capabilities	Return back to base for replacement of fire extinguishers	4

7.9. Sustainability

The life support system contributes to the sustainability of the design in multiple ways. As was said in Section 7.3, the fact that the CO_2 the crew produces is captured means that it can be used in a reactor back at Basecamp in combination with hydrogen to produce water. This water can then in turn be electrolized to produce hydrogen and oxygen, which can either be used in the fuel cells for producing power, or the oxygen can be used for breathing. This whole process makes the design more sustainable by reducing the amount of oxygen and hydrogen that needs to be transported to the Moon.

Another way in which the life support system contributes to the sustainability is the fact that the urine which the crew produces is also stored. When the LTS returns to Basecamp, this urine tank can be emptied and the urine can be processed to produce potable water. Similarly to the water which the CO_2 produces, this water can either be electrolized or used for drinking. This recycling of urine decreases the total water that is lost during operations and thus increases the sustainability of the LTS. The same can be done with the water vapor which is produced by the crew members during the use of the LTS.

7.10. Compliance Matrix

Table 7.6: Compliance matrix of the life support subsystem.

Requirement ID [REQ-SSYS-LS-#.#.#.#.#]	Description	Method	Compliance	Justification
2.2.1.1.1	The LTS' crew shall be exposed to pressures in the range between 34.5 kPa and 103 kPa	Analysis/Inspection	Yes/TBD	See Section 7.2
2.2.1.1.2	The area of the LTS the crew is exposed to shall have an oxygen partial pressure of 145-155 mmHg	Analysis/Inspection	Yes	See Section 7.2
2.2.1.1.3	The area of the LTS the crew is exposed to shall have an average one-hour CO2 partial pressure of no more than 3 mmHg	Analysis/Inspection	Yes/TBD	See Section 7.3
2.2.1.1.4	The LSS shall ensure local and remote control of atmospheric pressure, humidity, temperature, ventilation and ppO2	Analysis/Inspection	Yes/TBD	See Section 7.3
2.2.1.1.5	The LSS shall automatically record pressure, humidity, temperature, ppO2, and ppCO2 data continuously	Analysis/Demonstration	Yes	See Section 7.2 and Section 7.3. Also the prototype will demonstrate this
2.2.1.1.6	The LSS shall display real-time values for pressure, humidity, temperature, ppO2 and ppCO2 data to the crew locally and remotely	Analysis/Inspection	Yes/TBD	See Section 7.2 and Section 7.3
2.2.1.1.7	The LSS shall alert the crew locally and remotely when atmospheric parameters, including atmospheric pressure, humidity, temperature, ppO2, and ppCO2 are outside safe limits	Analysis	Yes	See Section 7.2 and Section 7.3
2.2.1.1.8	The LSS shall monitor in real-time the CO concentration in the internal area in the range of 5-1000 ppm with an accuracy of +-10% and resolution of 1 ppm	Analysis/Inspection	Yes/TBD	See Section 7.6
2.2.1.1.9	The LSS shall monitor in real-time the HCN concentration in the internal area in the range of 2-50 ppm with an accuracy of +-25% and resolution of 1 ppm	Analysis/Inspection	Yes/TBD	See Section 7.6

Continued on next page

Table 7.6 – *Continued from previous page*

Requirement ID [REQ-SSYS-LS-#.#.#.#.#]	Description	Method	Compliance	Justification
2.2.1.1.10	The LSS shall monitor in real-time the HCl concentration in the internal area in the range of 2-50 ppm with an accuracy of +-25% and resolution of 1 ppm	Analysis/Inspection	Yes/TBD	See Section 7.6
2.2.1.1.11	The LSS shall monitor in real-time the HF concentration in the internal area in the range of 2-50 ppm with an accuracy of +-25% and resolution of 1 ppm	Analysis/Inspection	Yes/TBD	See Section 7.6
2.2.1.1.12	The LSS shall alert the crew locally and remotely when the toxic atmospheric components including CO, HCN, HCl and HF are in the internal area	Analysis/Inspection	Yes/TBD	See Section 7.6
2.2.1.1.13	The LSS shall limit gaseous pollutant accumulation in the habitable atmosphere below individual chemical concentration limits specified in JSC-20584	Analysis/Inspection	Yes/TBD	See Section 7.3
2.2.1.1.14	The LSS shall limit the levels of lunar dust particles to less than 10 µm in size in the habitable atmosphere below a time-weighted average of 0.3 mg/m^3	Analysis/Inspection	Yes/TBD	See Section 7.3
2.2.1.1.15	The LSS shall limit the habitable atmosphere particulate matter concentration for total dust to <3 mg/m^3 with a crew generation rate of 1.33 mg/person-minute, and the respirable fraction of the total dust <2.5 µm (micrometer) in aerodynamic diameter to <1 mg/m^3 with a crew generation rate of 0.006 mg/person-minute.	Analysis/Inspection	Yes/TBD	See Section 7.3
2.2.1.1.16	The LSS shall provide air in the habitable atmosphere that is microbiologically safe for human health and performance	Analysis/Inspection	Yes/TBD	See Section 7.3
2.2.1.3.1	The area of the LTS the crew is exposed to shall contain diluent gas of at least 30%	Analysis/Inspection	Yes/TBD	See Section 7.2
2.2.1.4.1	For pressure differences greater than 1.0 psi, The LTS' crew shall not be exposed to pressure rates greater than 13.5 psi/min	Analysis/Inspection	Yes/TBD	See Section 7.3
2.2.1.4.2	During a commanded pressure change, the LSS shall be able to pause within 1 psi of the pause command being issued by the suited or unsuited crew member, with the ability to increase or decrease pressure as needed after the pause	Testing	TBD	
2.2.1.6.1	The LTS' crew shall be exposed to a relative humidity in the range 30-60%	Analysis/Inspection	Yes/TBD	See Section 7.3
2.2.1.7.1	The LSS shall ensure a ventilation rate in the range between 4.57-36.58 m/min	Analysis/Inspection	Yes/TBD	See Section 7.3
2.2.1.7.2	The LTS' crew shall be able to turn off the ventilation in the cabin	Analysis/Inspection	Yes/TBD	See Section 7.3
2.2.1.8.1	The LSS shall provide aesthetically acceptable potable water that is chemically and micro-biologically safe for human use, including drinking, food hydration, personal hygiene, and medical needs	Analysis/Inspection	Yes/TBD	See Section 7.5

Continued on next page

Table 7.6 – *Continued from previous page*

Requirement ID [REQ-SSYS-LS-#.#.#.#.#]	Description	Method	Compli-ance	Justification
2.2.1.8.2	The LSS shall prevent potable and hygienic water supply contamination from microbial, atmospheric, chemical, and non-potable water sources to ensure that potable and hygiene water are provided	Analysis/Inspection	Yes/TBD	See Section 7.5
2.2.1.8.3	The LSS shall provide the capability to monitor water quality and notify the crew locally and remotely when parameters are approaching defined limits	Analysis/Inspection	Yes/TBD	See Section 7.5
2.2.1.9.1	The LSS shall provide the crew with a minimum of 2.5L of water per crew member per day for hydration	Analysis/Inspection	Yes/TBD	See Section 7.5
2.2.1.9.2	The LSS shall provide the crew with a minimum of 400 mL of water per crew member per day for personal hygiene with a temperature of between 29 and 46 degrees Celsius	Analysis/Inspection	Yes/TBD	See Section 7.5
2.2.1.9.3	The LSS shall provide the crew with a minimum of 500 mL of water per crew member for eye irrigation with a temperature of between 16 and 38 degrees Celsius	Analysis/Inspection	Yes/TBD	See Section 7.5
2.2.1.9.4	The LSS shall provide the crew with a minimum of 5L per event for medical use and medical contingency with a temperature of between 18 and 27 degrees Celsius	Analysis/Inspection	Yes/TBD	See Section 7.5
2.2.1.9.5	The LSS shall provide the crew with a minimum of 240 mL of water per crew member per EVA hour	Analysis/Inspection	Yes/TBD	See Section 7.5
2.2.1.9.6	The LSS shall dispense the water at a rate that is compatible with the food system	Analysis/Inspection	Yes/TBD	See Section 7.5
2.2.1.9.7	The water shall be dispensable in specified increments that are compatible with the food preparation instructions	Analysis/Inspection	Yes/TBD	See Section 7.5
2.2.1.10.1	The LSS shall provide the crew with 600 mL of hot water per meal per crew member with a temperature of between 68 and 79 degrees Celsius	Analysis/Inspection	Yes/TBD	See Section 7.5
2.2.1.10.2	The bulk supply of water for hydration shall be accessible as nominal (18-27 degrees Celsius) or cold (maximum of 16 degrees Celsius) temperature	Analysis/Inspection	Yes/TBD	See Section 7.5
2.2.1.14.1	The LSS shall provide each crew member with an average of 3035 kcal of food per day	Analysis/Inspection	Yes/TBD	See Section 7.5
2.2.1.14.2	The LSS shall provide an additional 200 kcal per EVA hour above nominal caloric intake per crew member	Analysis/Inspection	Yes/TBD	See Section 7.5
2.2.1.14.3	The LSS shall provide the capability for preparation, consumption and stowage of food	Analysis/Inspection	Yes/TBD	See Section 7.5
2.2.1.15.1	The LSS shall provide a means to remove or isolate released chemical and biological contaminants and to return the environment to a safe condition	Analysis/Inspection	Yes/TBD	See Section 7.4
2.2.1.15.2	The LSS shall provide the capability for collection, containment, and disposal of body waste for both males and females	Analysis/Inspection	Yes/TBD	See Section 7.4

Continued on next page

Table 7.6 – *Continued from previous page*

Requirement ID [REQ-SSYS-LS-#.#.#.#.#]	Description	Method	Compli-ance	Justification
2.2.1.15.3	The LSS shall prevent the release of body waste from the body waste management system	Analysis/Inspection	Yes/TBD	See Section 7.4
2.2.1.15.4	The human body waste management system of the LSS shall be able to collect and contain 500 mL of feces per event for two events per day per crew member	Analysis/Inspection	Yes/TBD	See Section 7.4
2.2.1.15.5	The human body waste management system of the LSS shall be able to collect and contain 500 mL of diarrhea per event for eight events per day for up to 2 days per crew member	Analysis/Inspection	Yes/TBD	See Section 7.4
2.2.1.15.6	The human body waste management system of the LSS shall be able to collect and contain 1000 mL of urine per event for seven events per day per crew member	Analysis/Inspection	Yes/TBD	See Section 7.4
2.2.1.15.7	The human body waste management system of the LSS shall be able to collect and contain 114 mL of menses per crew member	Analysis/Inspection	Yes/TBD	See Section 7.4
2.2.1.15.8	The human body waste management system shall be isolated from the food preparation and consumption areas	Analysis/Inspection	Yes/TBD	See Section 7.4
2.2.1.15.9	The LSS shall provide privacy during the use of the human body waste management system	Analysis/Inspection	Yes/TBD	See Section 7.4
2.2.1.15.10	Body waste management supplies shall be provided for each crew member and be located within reach of crew members using the human body waste management system	Analysis/Inspection	Yes/TBD	See Section 7.4
2.2.1.15.11	Body waste management trash collection shall be accessible to and within reach of crew members using the human body waste management system	Analysis/Inspection	Yes/TBD	See Section 7.4
2.2.1.15.12	The LSS shall provide odor control for the body waste management system	Analysis/Inspection	Yes/TBD	See Section 7.4
2.2.1.16.1	The LSS shall have emergency medical supplies	Analysis/Inspection	Yes/TBD	See Section 7.6
2.2.1.18.1	The LTS shall use only chemicals that are Toxic Hazard Level Three or below, as defined in JSC-26895, in the habitable volume of the spacecraft	Analysis/Inspection	Yes/TBD	Only non-toxic chemicals are used in the cabin
2.2.1.18.2	The LTS shall prevent chemicals that are Toxic Hazard Level Four or below, as defined in JSC-26895, from entering the habitable volume of the spacecraft	Analysis/Inspection	Yes/TBD	If Toxic Hazard Level Four chemicals are used in the design these are carefully isolated from the cabin
2.2.1.19.1	The LSS shall have a fire protection system composed of detecting, warning, and extinguishing devices which do not create a hazardous environment	Analysis/Inspection	Yes/TBD	See Section 7.6
2.2.1.19.2	The fire protection system shall be capable of being manually activated and deactivated	Analysis/Inspection	Yes/TBD	See Section 7.6
2.3.1.1.1	The LSS design shall be such that no single point failure shall abort the mission and no second failure should endanger the crew	Analysis/Inspection	Yes/TBD	See Table 7.5
5.2.1.1.1	The LSS shall employ a modular design	Analysis	Yes	

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Table 7.6 – *Continued from previous page*

Requirement ID [REQ-SSYS-LS-#.#.#.#]	Description	Method	Compliance	Justification
5.4.1.1.1	The LSS shall not discharge any waste during operation	Analysis	Yes	
5.5.1.1.1	At EOL, the LSS should be recyclable to [TBD standards of extent]	Analysis	TBD	When a more detailed design will be made this can be analysed
5.7.1.1.1	The development costs of the LSS shall not exceed [TBD] Euros	Analysis	TBD	When a more detailed design will be made this can be analysed
5.7.1.2.1	Manufacturing costs of the LSS shall not exceed [TBD] Euros	Analysis	TBD	When a more detailed design will be made this can be analysed
5.7.1.3.1	Maintenance costs of the LSS shall not exceed [TBD] Euros	Analysis	TBD	When a more detailed design will be made this can be analysed
5.7.1.4.1	Operational costs of the LSS shall not exceed [TBD] euros	Analysis	TBD	When a more detailed design will be made this can be analysed

8 | Radiation and Micrometeorite Shielding

Author: Thomas

In this chapter, the radiation shielding sizing process will be described. First, in Section 8.1 the radiation environment on the Moon will be identified. After this, in Section 8.2, the radiation analysis and sizing of the shielding will be described. Subsequently, the micrometeorite shielding capability will be analysed in Section 8.3. Then, in Section 8.4, the radiation shielding of the airlock window will be sized. Afterwards, the sustainability aspects of the radiation shielding will be described in Section 8.6. Finally, the requirement compliance matrix will be given in Section 8.7.

8.1. Radiation Environment

One of the biggest dangers astronauts are exposed to while being on the Moon is radiation. On Earth, humans are shielded from high energy radiation and solar flares by the atmosphere and magnetosphere. However, on the Moon, there is little to no atmosphere and magnetosphere. This means that the radiation dose humans are exposed to is much higher. The two types of radiation they are exposed to are Galactic Cosmic Rays (GCR) and solar flares/storms which are grouped under the term Solar Particle Events (SPE). These SPEs have a very high flux density and energy, but do not occur that often. GCR are a very different type of radiation, they are present constantly, coming from intergalactic space. Compared to SPEs they have a low flux density, however they do have much higher energies and can produce secondary particles, making them hard to shield against.

8.2. Radiation Shielding

There are two main types of radiation shielding: passive and active shielding. Passive shielding does not require any power to function. There are two ways to passively shield from radiation; you can either move the radiation source further away, or you can put bulk shielding between the person and the radiation source. In this case, moving the radiation source further away from the crew is not possible. Active shielding does require power to shield the crew from the radiation. Two main types of active shielding have been proposed; these are magnetic shielding, where a strong magnetic field is used to deflect the particles away, and electrostatic shielding, where high voltages are used to slow the radiation particles down. A lot of research has been done into these two types of active radiation, however they are considered unfeasible with existing technology [36]. Therefore, the LTS will use bulk shielding.

The tool that was used to size the shielding is the On-Line Tool for the Assessment of Radiation in Space[1]. OLTARIS is user-friendly radiation analysis tool designed and managed by NASA. With this tool you can analyse the radiation dose experienced on the lunar surface for different types of shielding. There are two ways to model your vehicle/habitat in OLTARIS, either as a sphere or by uploading a thickness distribution. A thickness distribution would be a more accurate model of the LTS, however currently the only commercially available way to get a thickness distribution straight from a CAD model is through FASTRAD. The engineering department of Lunar Industries unfortunately does not have access to this tool. However, if this tool will be accessible in the future, the radiation analysis will be repeated using this tool. For now, the LTS was modeled as a sphere made of one layer with a so-called phantom (human target) at the center of this sphere.
Three different types of shielding materials were compared during the radiation analysis: water, polyethylene, which is the current state-of-the-art material for radiation shielding in space, and regolith. Each of these three materials was already defined in OLTARIS. Based on the requirements REQ-SSYS-LS-2.2.1.13.1 and REQ-SSYS-LS-2.2.1.13.2, two different radiation scenarios were chosen to compare the different shielding materials. For the Solar Particle Event, the sum of the October 1989 events was chosen, as defined in NASA-STD-3001 [37]. Additionally, for the Galactic Cosmic Rays, the 2010 solar minimum was chosen, as this would be the worst case scenario against which the crew would have to be protected. The three materials were compared for shielding thicknesses ranging from one to five centimeters. The results of the simulations can be seen in Figure 8.1 and Figure 8.2.

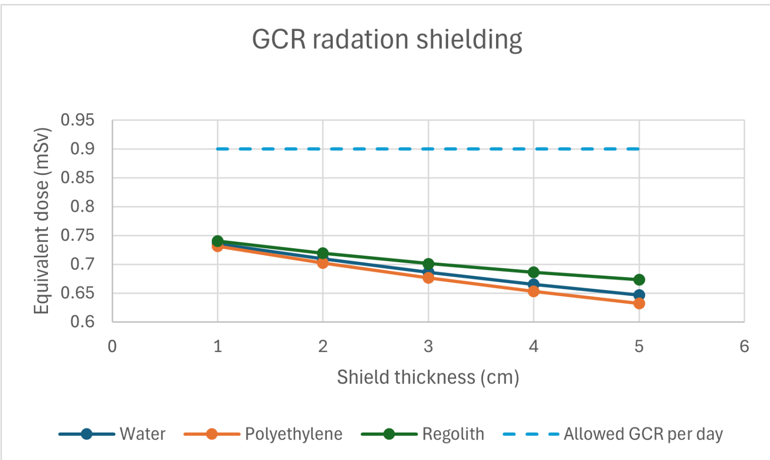


Figure 8.1: GCR radiation dose comparison for the 2010 solar minimum

[1] https://oltaris.larc.nasa.gov/

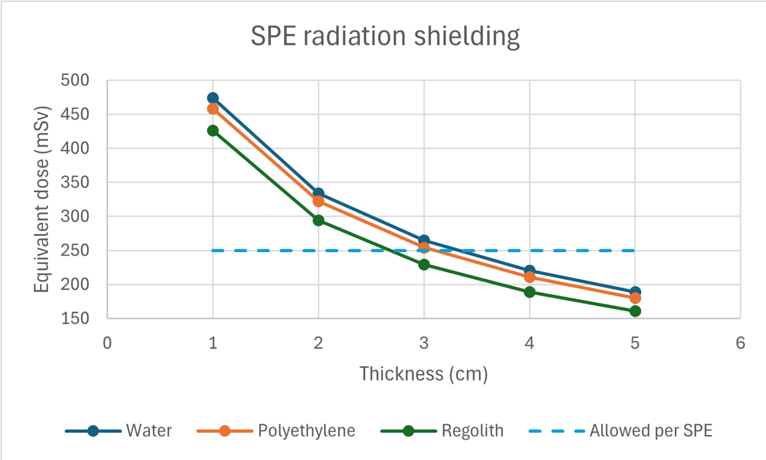


Figure 8.2: SPE radiation dose comparison for the sum of the October 1989 events

As can be seen in Figure 8.1, for the GCR radiation the shield thickness does not matter. For every shield thickness and every material, the equivalent dose is below the maximum allowed dose per day. This is because the lunar surface shields the LTS from a large part of the GCR radiation. There is some lunar albedo, however this only accounts for about 20% of the total GCR radiation dose. On the other hand, for the SPE radiation, the shielding thickness does clearly matter. With the regolith material shielding the equivalent dose is below the maximum allowed dose at a shield thickness of 3 cm. For polyethylene and water this happens at a shield thickness of about 4 centimeters. However, regolith has a density of 1.6 g/cm^3 compared to 1.0 g/cm^3 for both water and regolith. So, even though the shield thickness required is lower, the total weight of the shielding would be higher for the regolith.

The final radiation shielding material which was chosen is regolith. Even though this will make the radiation shielding significantly heavier, this mass will not have to be sent to the Moon from Earth, because regolith is abundantly available on the Moon. On the other hand, using regolith for shielding would require some infrastructure to produce the shielding panels, however this infrastructure could be reused to manufacture different structures out of regolith. To conclude, the final radiation shielding that was chosen will be made of regolith with a thickness of three centimeters. It will have a mass of around 1500 kg. This shielding will not completely surround the whole LTS; the floor will be left unprotected since the hydrogen, oxygen and water tanks will be placed there. These tanks provide enough shielding against the lunar albedo.

8.3. Micrometeorite Shielding

Next to shielding the crew from radiation, the shield should also protect the crew from micrometeorites. To analyse the protection the three centimeter thickness regolith shielding gives from micrometeorites, the Fisher-Summers equation was used [38]. This equation is defined as follows:

$$t = K_1 m^{0.352} V^{0.875} \rho^{\frac{1}{6}}$$

Where t is the target thickness (cm), K_1 is material constant for the target, m is the projectile mass (kg), ρ is the projectile density (g/cm^3) and V is the projectile velocity (km/s). The material constant in this equation has only been defined for aluminum, however we can convert the thickness for aluminum to the thickness for regolith using:

$$t_{reg} = \frac{t_{AL} \cdot \rho_{AL}}{\rho_{reg}}$$

Where t_{reg} and t_{AL} are the regolith and aluminum thickness, and ρ_{reg} and ρ_{AL} are the regolith and aluminum density. These equations were used, with a projectile density of 1 g/cm^3 [39] and projectile velocity of 70 km/s [40]. Assuming the micrometeorites to be spherical, the three centimeter of regolith can protect against meteorites with a size of up to 1.1 mm. This is large enough since most micrometeorites are 30-150 µm in size [40].

8.4. Airlock Window

The LTS will not have a large window in the cabin. However, it will have a small window in the outside airlock door. It is required by NASA to have a window in any hatch to the outside, in order to detect potential risks outside of the LTS [23]. This window will also need to have radiation shielding to prevent large amounts of radiation entering the LTS. A material which is suitable for this purpose is PMMA (acrylic glass), since it offers similar radiation protection performance to polyethylene [41]. This means that a thickness of four centimeter should be enough to meet the requirements. But, PMMA is known to darken with high-energy radiation [42]. A way to solve this problem is by making the load bearing part of the window out of silica glass, which does not suffer from radiation darkening, and using a replaceable PMMA cover to provide radiation shielding.

8.5. Failure Modes

Table 8.1: Failure modes of the powertain and mobility subsystem.

Component	Failure Mode	Effect	Prevention / Response	Critic -ality
Shield panel	Single Failure	Small increase in radiation dose, decreased protection against micrometeorites	Replace panel	2
	Complete Failure	Large increase radiation dose, no protection against micrometeorites	Return back to base for repair	3

8.6. Sustainability

In order to make the radiation shielding modular, the shielding will be made of multiple panels.

For the first LTS, the panels will be made of an outer aluminum shell of one millimeter thickness which will be filled with regolith. There will be enough space in the shell for three centimeters of regolith surrounding the whole structure except for the floor of the LTS. An example of a panel which will be placed on the cylindrical part of the LTS can be seen in Figure 8.3. At the top of the panel, a hole can be seen where the regolith can be put into the panel. When another panel with the same shape is put on top of this panel, the hole at the top will be closed, ensuring that the regolith will stay inside the panel.

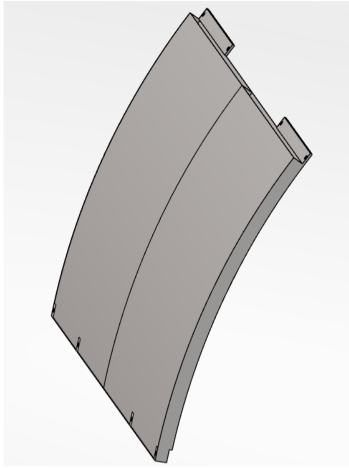


Figure 8.3: Example of a radiation panel of the LTS

In future versions of the LTS, the panels could be wholly made of regolith. An example of a process through which this could be done is regolith sintering [43]. This would require significant manufacturing infrastructure, however as said before, this infrastructure could be reused to make different structures making it scalable. There are two main advantages to making the regolith shielding out of separate panels. Firstly, if the radiation levels on the Moon increase due to the Solar cycle, the radiation shielding can be easily removed and replaced by thicker panels. Secondly, if the radiation shielding gets damaged, the panels which have been damaged can just be removed and replaced by undamaged panels. Both of these advantages increase the sustainability of the LTS.

8.7. Compliance Matrix

Table 8.2: Compliance matrix of the life support subsystem.

Requirement ID [REQ-SSYS-LS-#.#.#.#.#]	Description	Method	Compliance	Justification
2.2.1.13.1	The LTS shall protect the crew from exposure to the galactic cosmic ray environment to less than an effective dose of 0.9 mSv per day	Analysis/Inspection	Yes/TBD	Analysed using OLTARIS
2.2.1.13.2	The LTS shall protect the crew from exposure to the design reference Solar Particle Event (SPE) environment proton energy spectrum (defined in NASA-STD-3001) to less than an effective dose of 250 mSv	Analysis/Inspection	Yes/TBD	Analysed using OLTARIS

9 | Telecommunication

Author: Dani

Telecommunication is essential for any space mission as a communication channel with Earth is mandatory. As it was previously proposed, the main communication link will be through LunaNet [2][44][45]. This architecture is going to be further developed in this chapter. First, the required data rates are discussed, then the link budget is analysed. After that, the LunarSAR and system architecture is shown. Finally, a design and a failure mode analysis is present, after which the compliance matrix is included too for completeness sake.

9.1. Overview

The Telecommunication subsystem utilises the proposed LunaNet satellite system which allows positioning, and navigation services, and also a low-power communication channel with Earth and other agents. Thanks to it, the antennas are lightweight and consume almost negligible power. Additionally, there will be a separate emergency antenna integrated into both, the LTS and the crew's tablets, which allows distress broadcast even in the most dire situations, as they work from their own small battery. Lastly, the LTS will also allow WiFi connection to the crew during EVAs. This will allow them to be able to communicate with the LTS efficiently in a range of 15-45 m depending on the frequency.

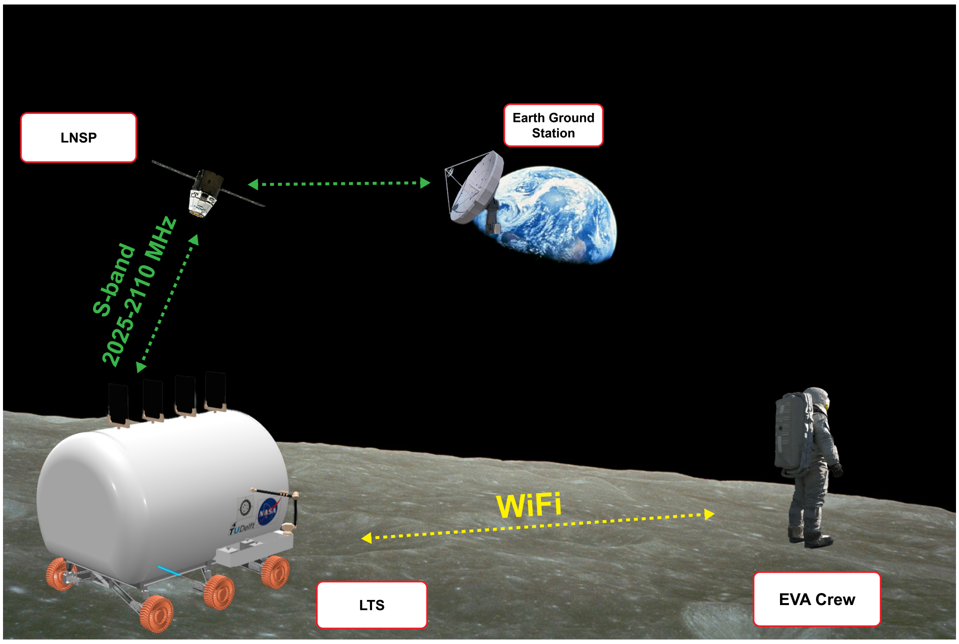


Figure 9.1: Telecommunication channels

9.2. Data Rate Required

First, before any link budget can be established, the required data rate has to be analysed. Most of the telemetry data (system and crew health, position) that the LTS has to send roughly amounts to the order of kilobytes per second, as these are mere compressed text messages. However, the main data rate is generated from the crew's text and video messages which amount to roughly 1.2 Mbps[1]. Additionally, training data for the autonomous system is also sent back to Earth, which should amount to roughly 0.3 Mbps. Thus, adding these up and allowing some margin (in case of potential peak data sending), the total uplink data rate needed can be estimated to be around 2 Mbps. Regarding downlink, much more data is needed. On one hand, this is due to all the data needed from LunaNet satellites, like space weather, global positioning and commands, which amounts to 3 Mbps. On the other hand, the needed data that is due to the crew, to provide them with video messaging and access to the internet, is roughly 10 Mbps. Thus, in total the required downlink is 13 Mbps. In the case of emergency, the LTS will only send the most important data to save power which amounts to some kilobytes,

[1] https://support.zoom.com/hc/en/article?id=zm_kb&sysparm_article=KB0060748

so the system is over designed and should allow for easy emergency broadcast.

9.3. Link Budget

As it was previously stated, the LTS will communicate with Basecamp and Earth through LunaNet, because in the Artemis Exploration Zone there is no permanent line of sight to Earth. Additionally, Moon orbiting satellites help drive down the required antenna size for proper communication (less free space loss). Now the LunaNet Service Provider (LNSP) satellites are still in development, so exact parameters of them are unknown. However, a good estimation for them is any Global Navigation Satellite Systems, like the European Galileo[2]. As per the LunaNet Specifications, the communication channel that the LTS will use is going to be the S-band. The link budgets for uplink and downlink can be seen in Figure 9.2 and Figure 9.3 [2].

Uplink S-band (LS-to-LO)

Input	Value	Unit
LTS Signal Power	15	W
Signal Frequency	2,0675	GHz
Signal Bandwidth	0,085	GHz
Data Rate	2000000	bits/s
LTS Antenna Diameter	0	m
LTS Efficiency Factor	0	-
LTS Antenna Gain	6,5	dB
LNSP Antenna Diameter	1,4	m
LNSP Efficiency Factor	0,55	-
LNSP Orbital Height	2000	km
LNSP Pointing Offset	0,73	deg
LNSP Half-power Angle	7,26	deg
System Noise Temperature	614	K
Antenna Noise Temperature	0	K
Amplifier Noise Figure	0	K
Reference Temperature	290	K
Cable Loss Factor	0,7	-
Transmitting Loss Factor	0,8	-
Receiving Loss Factor	0,8	-
Transmitting Path Loss Factor	2,5	dB
Boltzmann Constant	1,38E-23	J/K
Speed of Light	300000000	m/s
Gravitational Constant	6,67E-08	-

Output	Value	Unit
Signal Power	11,76	dBW
Transmitting Loss Factor	-0,97	dB
Gain Factor Transmitting	6,50	dB
Transmitting Path Loss Factor	-2,50	dB
Free Space Loss	-164,77	dB
Gain Factor Receiving	27,04	dB
Pointing Loss Factor	-0,24	dB
Receiving Loss Factor	-0,97	dB
Data Rate	-63,01	(bits/s)^-1
Boltzmann Constant	228,60	(J/K)^-1
System Noise Temperature	-27,88	(K)^-1
Signal to Noise Recieved	13,56	-
Signal to Noise Required	0,02	-
Margin	13,54	-

Figure 9.2: Uplink S-band (LS-to-LO)

[2]https://www.eoportal.org/satellite-missions/galileo-foc#genesis-project

Downlink S-band AFS (LO-to-LS)

Input	Value	Unit
LNSP Signal Power	10	W
Signal Frequency	2,492028	GHz
Signal Bandwidth	0,0165	GHz
Data Rate	13000000	bits/s
LTS Antenna Diameter	0	m
LTS Efficiency Factor	0	-
LTS Antenna Gain	6,5	dB
LNSP Antenna Diameter	1,4	m
LNSP Efficiency Factor	0,55	-
LNSP Orbital Height	2000	km
Pointing Offset	0,60	deg
Half-power angle	6,02	deg
System Noise Temperature	135	K
Antenna Noise Temperature	0	K
Amplifier Noise Figure	0	K
Reference Temperature	290	K
Cable Loss Factor	0,7	-
Transmitting Loss Factor	0,8	-
Receiving Loss Factor	0,8	-
Transmitting Path Loss Factor	2,5	dB
Boltzmann Constant	1,38E-23	J/K
Speed of Light	300000000	m/s
Gravitational Constant	6,67E-08	-

Output	Value	Unit
Signal Power	10,00	dBW
Transmitting Loss Factor	-0,97	dB
Gain Factor Transmitting	33	dB
Transmitting Path Loss Factor	-2,50	dB
Free Space Loss	-166,39	dB
Gain Factor Receiving	7	dB
Pointing Loss Factor	-0,24	dB
Receiving Loss Factor	-0,97	dB
Data Rate	-71,14	(bits/s)^-1
Boltzmann Constant	228,60	(J/K)^-1
System Noise Temperature	-21,30	(K)^-1
Signal to Noise Recieved	14,59	-
Signal to Noise Required	0,73	-
Margin	13,86	-

Figure 9.3: Downlink S-band Augmented Forward Signal (LO-to-LS)

After analysing the link budget it was decided that a small patch antenna is the best design option for this application. Weighing in sustainability, weight and power consumption, the preliminary choice for an antenna is the S-Band TT&C Antenna[3] of a French company, ANYWAVES. Of course the antennas can be developed in house as well, however, this provides a good first estimate of the design.

9.4. LunarSAR

For the emergency broadcast, a separate system is in place in case the whole telecommunication system fails. The LTS will employ a small emergency antenna which does not need considerable power; it can function off of its own independent battery. The crew will also have a little emergency antenna[4] integrated into its tablets to ensure continuous distress signals even if the LTS emergency antenna is damaged. The communication channel for this will be a 406 Mhz channel which is the global distress channel [2]. This allows to expand the system already used on Earth, the Search and Rescue (SAR) system, to the Moon as well[5].

9.5. System Architecture

The communication channels can be seen in Figure 9.1, with the addition of a WiFi channel to allow the crew to communicate with the LTS during EVAs. This solution was chosen because a link from the crew to the LunaNet satellites, which they relay back to the LTS, was deemed too costly and time-consuming. It will allow the crew to receive information from the LTS and even operate it (control airlock for example) from the outside. The range of this on Earth is between 15-45 m (depending on the frequency) which is overall a safe first estimate for the Moon as well, however tests need to be carried out to assess vacuum WiFi performance.

[3] https://satcatalog.s3.amazonaws.com/components/1300/SatCatalog_-_Anywaves_-_S-Band_TTC_Antenna_-_Datasheet.pdf?lastmod=20220216050117

[4] https://www.sarsat.noaa.gov/wp-content/uploads/2021/08/SAR_2018_NASA_Mazzuca_Mar8-1.pdf

[5] https://www.nasa.gov/missions/artemis/nasa-develops-second-generation-search-and-rescue-beacon-technology/

Additionally, the mass and power budget can be seen in Table 9.1. In the quantity column the numbers in italic and in parenthesis mean that they are the quantity of the components including the redundant ones that nominally will not be used. Similarly, in the power in active mode column the parenthesis mean that the emergency beacon would require at max 1 W, however nominally it is not used of course.

Table 9.1: Mass and power budget of the telecommunication subsystem. Numbers in italic and in parenthesis mean that they are the quantity of the components including the redundant ones. In the power in active mode column the parenthesis mean that the emergency beacon would require at max 1 W non-nominally.

Component	Quantity	Mass [kg]	Power in active mode [W]	Power in hibernation [W]
Antennas	4 *(6)*	0.8	60	30
Emergency Beacons	1 *(2)*	0.1	0 (1)	0
WiFi	1 *(2)*	0.5	5	0
Amplifiers	4 *(6)*	1.2	4	2
Electronics	1	0.5	3	3
	Total:	**2.2**	**69**	**32**

9.6. Failure Modes

The reliability and safety of the subsystem is very important as if the crew gets stranded somewhere on the Moon, communication with Basecamp or with Earth is at utmost importance. One of the most common ways to analyse component reliability is the FMEA method which ensures that all failures can be tracked, isolated or prevented. The Telecommunication subsystem's failure mode analysis is presented in Table 9.2. This is not as extensive as it will be once the detailed design is done, however, for a preliminary design, it is a good starting point which later on can be further developed.

Table 9.2: Failure modes of the telecommunication subsystem.

Component	Failure Mode	Effect	Prevention / Response	Criticality
Antennas	Single Failure	Insignificant	Use backup	4
	Complete Failure	Loose communication capabilities	Return back to base for repair	1
Emergency Beacons	Single Failure	Insignificant	Use backup	4
	Complete Failure	Loose emergency capabilities	Return back to base for repair	2
Electronics	Connection Failure	Loose communication capabilities	Highly reliable and redundant wiring	1

9.7. Design Analysis

Employing a combination of multiple antennas is common practice in the automotive industry for high redundancy, as they are lightweight and do not consume significant power. This allows for easy repair and maintenance, even if one is removed it can still function properly. Moreover, the low power requirement ensures that the antennas can be switched out for other models if needed, making the system highly modular.

9.8. Compliance Matrix

Table 9.3: Compliance matrix of the telecommunication subsystem.

Requirement ID [REQ-SSYS-TC-#.#.#.#.#]	Description	Method	Compliance	Justification
1.5.1.1.1	The telecommunication subsystem shall comply with the interfaces specified in the LunaNet Interoperability Specification LNIS V.5.	Analysis/Inspection	Yes/TBD	See Section 9.3.
1.5.1.1.2	The telecommunication subsystem shall provide a lunar communications relay capable of real-time data relay services between LNSPs and the LTS.	Analysis/Inspection	Yes/TBD	See Section 9.3.
1.5.1.1.3	The telecommunication subsystem shall provide a lunar communications relay capable of store-and-forward data relay services between LNSPs and the LTS.	Analysis/Inspection	Yes/TBD	See Section 9.3.
1.5.1.1.6	The telecommunication subsystem shall provide dedicated S-band return data services to LNSPs.	Analysis/Inspection	Yes/TBD	See Section 9.3.
1.5.1.1.7	The telecommunication subsystem shall provide dedicated S-band forward data services to LNSPs.	Analysis/Inspection	TBD	See Section 9.3.
3.6.1.1.1	The telecommunication subsystem shall provide nominal operational service availability of 99.99%.	Analysis	TBD	See Section 9.3.
3.6.1.1.2	The telecommunication subsystem shall provide critical operational service availability of 99.99%.	Analysis	TBD	See Section 9.4.
1.4.1.3.1	The telecommunication subsystem shall send telemetry data every 20 min.	Analysis/Inspection	TBD	See Section 9.3.
1.4.1.2.1	The telecommunication subsystem shall support messaging services between different lunar users, enabling exchange of navigation, schedule and space situational awareness information.	Test	TBD	See Section 9.3.
1.4.1.2.2	The telecommunication subsystem shall maintain user provided encryption sent to and from the lunar relays and Earth ground stations.	Test	TBD	See Section 9.3.
1.4.1.2.3	The telecommunication subsystem shall provide real-time status and Quality of Service (QOS) metrics to the user.	Test	TBD	See Section 9.3.
1.4.1.2.6	The telecommunication subsystem shall provide real-time S-band data rate return link of 13 Mbps.	Analysis/Inspection	TBD	See Section 9.3.
1.4.1.2.7	The telecommunication subsystem shall provide real-time S-band data rate forward link of 2 Mbps.	Analysis/Inspection	Yes/TBD	See Section 9.3.
1.4.1.2.8	The telecommunication system shall provide real-time WLAN for crew during EVA.	Test	TBD	Needs to be tested of how the lunar environment might affect it.
5.2.1.1.1	The telecommunication subsystem shall comply with electrical interfaces in the LTS.	Analysis	Yes	See Section 9.3.

Continued on next page

Table 9.3 – *Continued from previous page*

Requirement ID [REQ-SSYS-TC-#.#.#.#.#]	Description	Method	Compliance	Justification
5.2.1.1.2	The telecommunication subsystem shall employ a modular design.	Inspection	Yes	See Section 9.3.
2.3.1.1.1	The telecommunication subsystem shall provide a forward emergency channel of 406 MHz to the LTS.	Analysis	Yes	See Section 9.4.
2.3.1.1.2	The telecommunication subsystem shall provide a return emergency channel of 406 MHz to the LTS.	Analysis/Inspection	Yes/TBD	See Section 9.4.
2.3.1.1.3	The telecommunication subsystem shall provide at least 2 back-up antennas.	Inspection	Yes & TBD	See Section 9.3.
5.7.1.1.1	Development cost of the Telecommunication shall not exceed [TBD] euros.	Analysis	TBD	Stakeholder meetings concerning monetary budget constraints are planned to take place after the 10-week mark.
5.7.1.2.1	Manufacturing cost of the Telecommunication shall not exceed [TBD] euros.	Analysis	TBD	See 5.7.1.1.
5.7.1.3.1	Maintenance cost of the Telecommunication shall not exceed [TBD] euros.	Analysis	TBD	See 5.7.1.1.
5.7.1.4.1	Operational cost of the Telecommunication shall not exceed [TBD] euros.	Analysis	TBD	See 5.7.1.1.

10 | Guidance, Navigation, and Control

Author: Lee

The Guidance, Navigation, and Control (GNC) system is crucial for precise maneuver, hazard avoidance, and autonomous operation of the LTS. Before designing the GNC system, subsystem requirements were first formulated from the system requirements to guide the design and align it with the stakeholder needs and goals. Considering these requirements, various sensors, algorithms, and modules would have to be integrated to enable reliable and autonomous exploration in the challenging lunar environment. Firstly, the sensors were selected in Section 10.2, considering comprehensive aspects such as redundancy, accuracy, and Field Of View(FOV). The GNC modules encompass a range of functionalities crucial to autonomous operation of teh LTS, which will be explained in Section 10.3 in detail. Based on the selected sensor and stated modules, the mechanism for the GNC system will be further explained by presenting the GNC architecture in Section 10.4. Also, the possible failure modes with their criticality for the cargo handling system will be assessed in Section 10.5. Lastly, a compliance matrix will be presented in Section 10.6 to show if the designed GNC meet the requirements.

10.1. Overview

Figure 10.1 provides an overview of the designed GNC system, illustrating its selected sensors. For this GNC system, LIDARs were selected for the relative localisation by measuring acceleration and orientation of the buggy, while Hazcam and LIDARs were chosen for hazard detection and mapping creation for path planning. This configuration of sensors ensures redundancy and integrity of the overall GNC system.

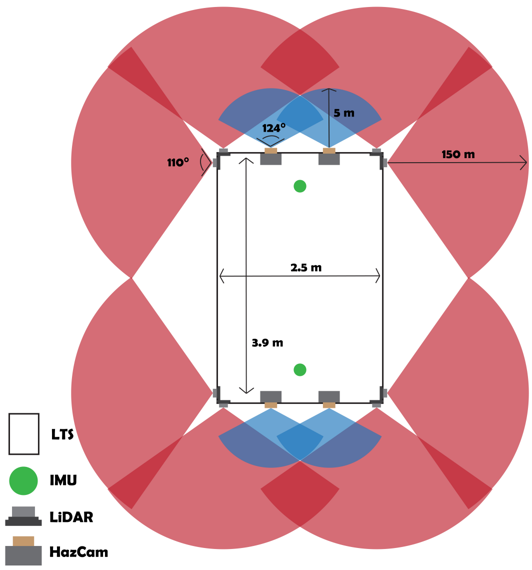


Figure 10.1: GNC illustration

10.2. GNC Sensors Selection

The fusion of internal sensors and external sensors significantly enhances the overall accuracy and redundancy of GNC system. The methods using internal sensors are wheel odometry and inertial odometry.These methods are mainly used for relative localisation by tracking the movement and orientation of the buggy based on wheel rotations or inertial measurements. These methods have a high frequency which is crucial for real-time localisation, but also have a low accuracy[46] compared to external sensor methods[47]. Also, these methods are relatively energy efficient compared to the methods employed for external sensors[47]. The methods using external sensors are LIDAR odometry and visual odometry. These methods capture the image or detect the terrain condition, which enables precise path planning procedure. The accuracy using these methods is significantly high but frequency is low[46]. Therefore, combining these methods integrates their respective functions, compensating for individual weaknesses and applying redundancy to the GNC system. For the methods using internal sensors, a trade-off was performed in Table 10.1 to choose either wheel odometry or inertial odometry.

Table 10.1: Trade-off Table: Wheel Odometry vs. Inertial Odometry

Criteria	Weight	Wheel Odometry	Inertial Odometry
Accuracy	0.4	2	3
Frequency	0.3	5	4
Environmental Robustness	0.3	2	4
Overall Score		3.35	3.75

The accuracy was weighted highest, because it is directly linked to the effectiveness of localisation. For the accuracy, wheel odometry scored lower (2) compared to inertial odometry (3), as errors from wheel slippage in wheel odometry are more difficult to mitigate than drift errors in inertial odometry. Then, the frequency was chosen as the second most important criteria, because it is crucial for real-time localisation by frequent update. Wheel odometry has a slightly higher frequency than Inertial odometry[47]. The last criteria was environmental robustness. The wheel odometry is more susceptible to varying terrain conditions and obstacles, because it relies on the physical contact and movement of the wheels unlike inertial odometry[47]. Considering these aspects, inertial odometry was determined to be used, following the result of the performed trade-off inTable 10.1.

For the method using external sensors, both LIDAR and visual odometry were selected. LIDAR odometry is more accurate and robust in low-light environments while visual odometry better facilitates real-time mapping by providing more frequent updates [46]. Therefore, LIDAR odometry was chosen to be used for the long-range detection of 3D mapping and path planning, while visual odometry was selected to be mainly used for short-range detection, such as for hazard detection. Based on the selected odometry, sensors were selected as portrayed in Table 10.2, with their properties, such as FOV and range, detailed in Table 10.3.

Table 10.2: GNC system budget

Component	Position	Quantity	Mass (kg)	Power (W)	Cost(€)
LIDARs: LD-MRS UAV[48]	2 LIDARs at each corner	8	0.77	8	8326
Hazcam: MSL Hazcam[49]	2 middle at front and back	4	0.245	2.2	2330
IMU: HG5700[50]	1 at front and back	2	1.36	7	8087
TOTAL			9.86	86.8	92102

Table 10.3: Component properties

Component	Property
LIDARs: LD-MRS UAV[48]	FOV:110° x 3.2° Range:150 m, Frequency: 50 Hz, Data Interface: ETH/CAN
Hazcam: MSL Hazcam[49]	FOV:124° x 124°, Range: 5 m
IMU: HG5700IMU[50]	Drift rate: 0.048 µrads/s, Data rate: 100 Hz (Guidance) and 600 Hz (Control)

The selection of these sensors was performed in order to satisfy subsystem requirements. Therefore, the IMU, with the drift rate and data rate specified in Table 10.3, was used. The drift rate of the chosen IMU satisfies REQ-SSYS-GNC-3.1.1.2.1, given the range and speed of the LTS. Also, mass and power requirements were considered in the selection of the sensors, while still ensuring full image coverage. To satisfy REQ-SSYS-GNC-2.3.1.1.1, redundancy was taken into account to determine the necessary quantity of each sensor. For redundancy, two LIDARs were placed at each corner of the buggy, summing up to 8 LIDARs to be used. This placement of LIDARs also ensure comprehensive coverage from every angle in the LTS, which enables precise and safe mapping and path planning. The Hazcam was positioned to detect any hazards within a 5-meter range both in front of and behind the buggy. To ensure redundancy and fulfill this purpose, a total of four Hazcams were used: two positioned at the middle front and two at the middle rear of the buggy. Lastly, two IMUs were installed within the buggy for redundancy. These sensor selection and positioning measures ensure comprehensive coverage, redundancy, and reliability for the GNC system. An overview of the placement of the selected sensors can be seen in Figure 10.1.

10.3. GNC Modules

For autonomous GNC operations, multiple GNC modules are required to be processed. The perception module processes stereo images to create disparity maps. Then, these maps are transformed into Digital Elevation Models (DEMs) and into 3D terrain modelling by the navigation module. The path planning module computes optimal paths, while trajectory control executes these paths. Then, Absolute and Relative Localization ensure precise positioning of the buggy in global coordinates and relative to its surrounding. Lastly, Simultaneous Localisation and Mapping (SLAM) allows for autonomous GNC operations. In this section, the functional flow of GNC subsystem will be addressed by explaining each of the GNC modules.

Perception

The perception system analyzes stereo images captured by camera to produce disparity maps. This disparity map will be used in later navigation models to generate a 3D model of the terrain, which is essential for navigation. To produce an accurate and timely disparity map, the system uses a multi-resolution approach to maximize terrain coverage and mitigate adverse processing time[51]. Laplacian of Gaussian filter is used to produce gradient images, and then correlation algorithms scan these stereo images horizontally to detect the disparity value[51]. Then, a final filtered multi resolution disparity map is produced by using disparity map filtering which removes any erroneous values from raw a disparity map[51].

Navigation

The navigation module creates maps for the path planning and traverse monitoring modules. The disparity maps generated by the perception module are transformed into Digital Elevation Models (DEMs) and then into terrain models with estimated mean, minimum, and maximum elevations[51]. Then, a 3D point data cloud from LIDARs enhances the precision of this terrain model by providing detailed terrain features[52]. This terrain model results in terrain feature maps after accounting for the buggy's traversal capabilities such as maximum step height and rover tilt angle[51]. This terrain feature map then becomes a location navigation map, which classifies areas as traversable, non-traversable, or unknown, with cost values assigned to traversable areas based on difficulty and risk[51].

Path Planning

At each navigation stop, the path planning module plans the next path sequence for trajectory control. The planned path sequence consist of smooth curves and point turns where the rover is to rotate without moving forward[51]. The module selects an escape point on the boundary of the known map to plan a route by avoiding non-traversable and unknown areas while minimizing costs[51]. This path planning will be executed using an A* search algorithm on a hybrid search graph as shown in Figure 10.2.

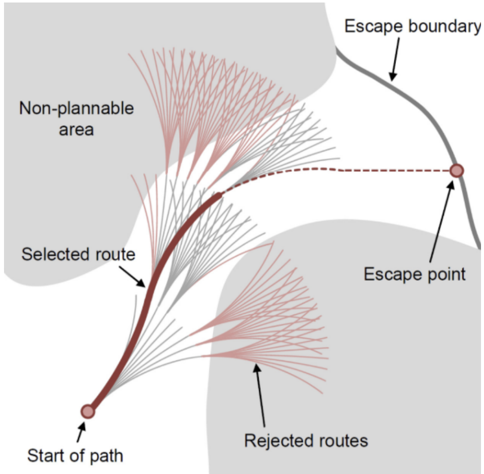


Figure 10.2: Illustration of path planning algorithm[51]

Localisation

The localisation module can be separated into two categories: absolute localisation and relative localisation. Absolute localization refers to the detection of the position of the buggy in a global coordinate system[51]. This will be achieved by LunaNet which accurately tracks the buggy's location in the global coordinates of the lunar surface. On the other hand, relative localization refer to the determination of the buggy's position and orientation relative to its immediate surroundings[51]. This relative localisation is achieved by IMUs, LIDARs, and stereo cameras, which continuously monitor the buggy's movements and surroundings in the lunar environment. The IMU provides real-time measurements of acceleration and angular rates while LIDARs and stereo cameras create detailed 3D maps of the terrain and identify surrounding objects. These sensors facilitate the accurate relative localisation, enabling the buggy to stay on its planned trajectory during the operation.

Trajectory Control

The estimated rover position and attitude provided by localisation modules are used for the trajectory control module. This trajectory control module calculates the locomotion maneuver commands required to control the rover to drive along the planned path sequence, using closed-loop control[51]. This module is to maintain lateral and heading errors within the limits for path segments to comply with REQ-SSYS-GNC-3.1.1.2.2. Based on the calculated locomotion commands, the powetrain and mobility system will be controlled to keep the rover along the planned path.

Simultaneous Localisation and Mapping (SLAM)

Simultaneous Localization and Mapping (SLAM) is an algorithm used to create a map of the environment and localize the buggy simultaneously. This capability is crucial for achieving autonomy of the buggy to satisfy the autonomy capability requirements stated in Section 10.6. The SLAM integrates data from LIDARs and stereo cameras to update the map and position of the buggy simultaneously, enhancing the accuracy and reliability of the GNC system.

10.4. GNC Architecture

The Based on the GNC modules described in Section 10.3 and selected sensors from Section 10.2, the GNC architecture was created in Figure 10.3. This GNC architecture provides visual illustration for interaction between selected sensors, actuators, and LunaNet, integrating the GNC modules.

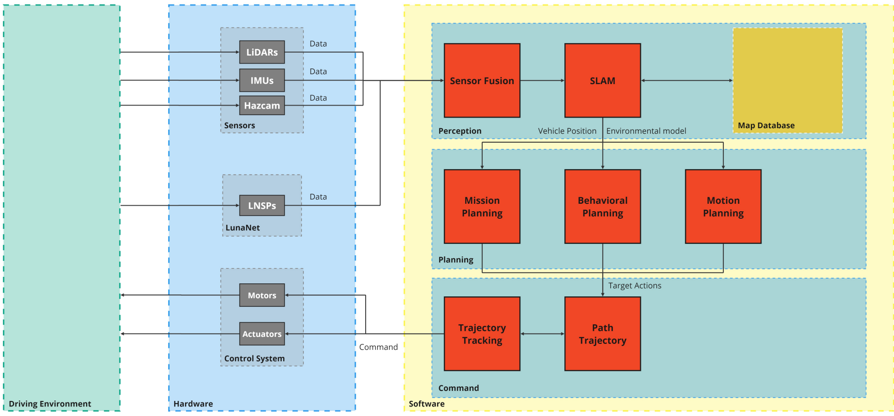


Figure 10.3: GNC architecture

10.5. Failure Modes

Table 10.4: Failure modes of the GNC subsystem

Component	Failure Mode	Effect	Prevention / Response	Criticality
LIDAR	Single Failure	Insignificant	Use backup	4
	Complete Failure	Inability for terrain detection	Return back to base for repair	1
IMU	Single Failure	Insignificant	Use backup	4
	Complete Failure	Inability for measurement of orientation of the LTS	Return back to base for repair	2
Hazcam	Single Failure	Insignificant	Use backup	4
	Complete Failure	Inability for hazard detection	Return back to base for repair	1

10.6. Compliance Matrix

Table 10.5: Compliance matrix of GNC subsystem

Requirement ID [REQ-SSYS-TC-#.#.#.#.#]	Description	Method	Compli-ance	Justification
2.3.1.1.1	The GNC shall have redundant sensors	Analysis/Demonstration	Yes/TBD	see Section 10.2
2.3.1.1.2	The GNC shall have redundant actuators	Demonstration	TBD	Redundant actuators will be used
3.1.1.1.1	The GNC shall have autonomous trajectory control	Test	TBD	see SLAM module

Continued on next page

Table 10.5 – *Continued from previous page*

Requirement ID [REQ-SSYS-TC-#.#.#.#.#]	Description	Method	Compli-ance	Justification
3.1.1.1.2	The GNC shall have autonomous path planning	Test	TBD	see SLAM module
3.1.1.1.3	The GNC shall have autonomous mapping and localization	Test	TBD	see SLAM module
3.1.1.1.4	The GNC shall utilize an autonomous navigation software	Test	TBD	see SLAM module
3.1.1.2.1	The GNC shall have a cross range error of less than 10 percent in mission range	Test	TBD	see Section 10.2
3.1.1.2.2	The GNC shall have a heading reference error of less than 5 degrees	Test	TBD	Testing to be performed
3.1.1.2.3	The GNC shall have a drift rate of less than 0.47 $\mu rads/s$	Test	TBD	see Section 10.2
3.1.1.2.4	The GNC shall have a pointing accuracy smaller than 1 degree	Test	TBD	Testing to be performed
3.1.1.2.5	The GNC shall have a relative localization frequency of at least 10 Hz	Test	TBD	see Section 10.2
3.1.1.2.6	The GNC shall take less than 20 seconds to operate the perception system	Test	TBD	Testing to be performed
3.1.1.2.7	The GNC shall continuously update the map	Test	TBD	Testing to be performed
5.7.1.1.1	Development costs of the GNC shall not exceed 100,000 Euros	Demonstration	TBD	see Section 10.2

11 | User Interface

Author: Dani

In this chapter, the user interface of the LTS will be discussed. This subsystem will be the one the crew interacts with first and through which they will communicate with the LTS and other agents. The chapter presents an argumentation for the subsystem's functions, the main system elements and their analysis. Moreover, the each component's failure modes are also assessed, to provide a basis for reliability and safety. Lastly, a compliance matrix is included to check if all the requirements are met by the design.

11.1. Overview

The subsystem consists of four main components, the chairs that also function as beds, the dashboard that allows the crew to control and monitor the LTS, the lights which ensures the crew's healthy and stimulated natural rhythm and finally, a window mimicking system. The window mimicking system will satisfy the crew as it will act as an immersive display showing the outside of the LTS just like a window would, but without the disadvantages (structural weakness, lack of radiation protection).

11.2. System Elements

It was decided, based on functionalities, that the user interface subsystem should consist of most the system elements that the crew of the LTS will interact with. This includes the chairs, the lights, the dashboard and its accompanying components, a window mimicking screen and finally a tablet used during EVA. All of these are further being discussed in detail below.

Chairs

As the LTS is designed in such a way to allow trips of up to 2 days, the crew will need a place to sleep in addition to one to just sit. Due to limited space availability, it was decided that the chairs should employ a two-in-one design, providing both sitting and sleeping capabilities. Several options were analysed, like a foldable bed or a chair where the backrest can be reclined. However, in the end, a zero-gravity body posture chair was chosen for its added benefits.

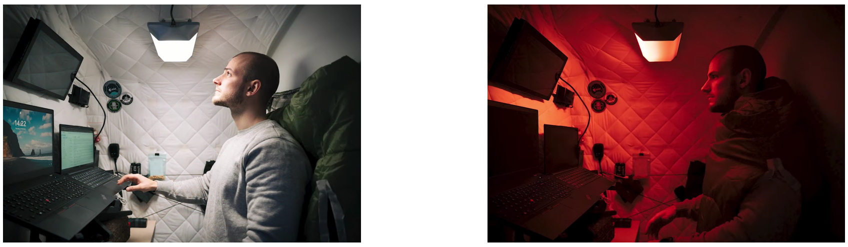
Numerous studies show that the zero-gravity body posture (the shape the human body naturally takes up in zero gravity environments) has lots of health benefits, like relieving pressure on the heart, neck, back, hips, and knees [53]. This is something that NASA noticed as well and based on their studies, some companies used the newly gained knowledge to create a new generation of massage chairs[1]. Now as far as designing a chair for long trips goes, one of the most important aspects is comfortability. To achieve this, the closest thing that can comes to mind is a layout similar to a massage chair that was designed for maximum health benefits and comfortability[2]. As the example shows, this chair is easily movable into a sleeping position without any special mechanics. Moreover, the zero-body posture, shaped inside, allows easy access to the chair (even for people with mobility impairments) and perfect conditions for watching the screens, reading, relaxing.

Of course, the chairs will not be a one-to-one copy of a massage chair as lots of the functionalities of a massage chair are costly to implement on the Moon. However, regarding the reclining functions, and their size, it is a good first estimate for the chairs' characteristics. In a sitting position, one chair has dimensions of 165 x 83 x 121 cm (L x W x H), while in horizontal position dimensions of 208 x 83 x 87 cm (L x W x H). Lastly, some extra functions can be added to the chairs, like USB connection points for easy charging, or a tablet holder (the function of it will be discussed more in detail later on).

Lights

A light source inside the cabin is essential for the crew; not just for comfortability reasons, but also because it helps maintain good well-being by getting stimulated and keeping a healthy circadian rhythm [23]. To achieve this, a literature study concluded that the most promising lighting technology for space applications is a lamp developed by a Danish company, SAGA. The Circadian Light Panel works on the basis of emitting different wavelengths from its three sides to stimulate its users while making sure that it syncs up to their natural 24-hour cycle. It provides lighting conditions similar to what someone would experience on Earth (Figure 11.1[3]). This allows the crew throughout longer trips to easily fall asleep and keep a healthy rhythm. Additionally, the lamp is currently being tested on the ISS[4] to validate and improve the concept, but the technology already shows great success and promise.

[1] https://spinoff.nasa.gov/Spinoff2020/cg_5.html
[2] https://restlords.com/nl/massagestoelen/irest-brillactiq-a665/
[3] https://www.saga.dk/projects/circadian-light
[4] https://commercialisation.esa.int/2023/07/sagas-circadian-light-revolutionising-astronauts-sleep-in-space/



(a) Day lighting (b) Sunset lighting

Figure 11.1: Different lighting conditions depending on the crew's schedule

Dashboard

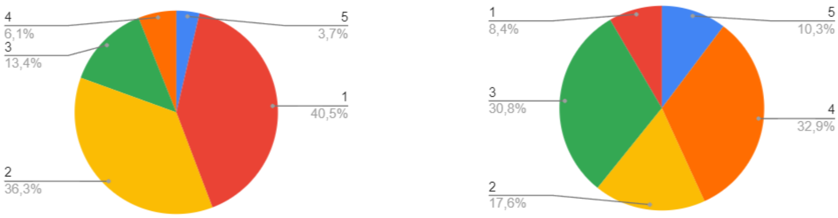
The dashboard is consisting of several elements: two display screens, some mechanical buttons and a sound system. All of these are housed in a very lightweight aluminium/plastic casing in front of the chairs. For the displays, several options were considered such as a HUD or VR headset, however these were discarded in favor of a normal screen because of complexity and ease of use. Now regarding each component, the displays will convey all the necessary information to the crew that it needs. These include but are not limited to the cabin temperature and humidity, vibration levels, system health, space weather information, a real-time map and the planned route of the LTS. Moreover, the displays should allow for control of the LTS through inputting destination coordinates. The system will also have the option for general communication (like texting) between the crew and Basecamp or with Earth.

The sound system will ensure a redundant way to communicate with the LTS in case both of the displays fail. Additionally, it allows a voice messaging service and even music entertainment for the crew. Then, the mechanical buttons are included to provide a high level of redundancy and reliability. These buttons will also be the ones to be used to stop the LTS in case the crew wants to go for an originally unscheduled EVA. Lastly, the EVA tablets can also be used for indoor control, to allow versatile accessibility of the command center for the crew anywhere in the LTS.

Windows

Having windows in a pressure vessel is hard to design for. It induces huge stress concentrations, sometimes leading to stress concentration factors of 4 [54], leading to unreasonable skin thicknesses of around 2 cm. Furthermore, having windows also leads to complications when designing the radiation and micrometeorite shielding. A lot of effort would have to go into designing these large windows to have as much protection from radiation and micrometeorites as the rest of the structure. Due to the transparent materials which can be used for the windows, having worse protection properties, the windows would have to be very thick and heavy. So having no windows and a screen with live video feed makes it easier to design the pressure vessel and to protect against radiation and micrometeorites. Next to that it is more modular, since a camera is easier to replace than a whole window.

Moreover, a survey collecting answers from a cross-sectional array of over 200 people – cross-generational and across professions – was conducted to explore the comfortability concerns of not including windows. The poll yielded that only a third of the people being asked would feel comfortable without a window at all (Figure 11.2a). However a two thirds of the surveyed people would be fine and comfortable with a window mimicking system (Figure 11.2b). The main concern is motion sickness, due to latency and acceleration issues, however there are already studies and researches developing new technologies to counter them. Thus it was decided that a window mimicking system would be a better option, and to satisfy NASA requirements, a small window on the airlock was added.



(a) Having no windows or mimicking system **(b)** Using a mimicking system

Figure 11.2: Window Mimicking System survey results, where 1 means very uncomfortable and 5 means very comfortable.

EVA Tablets

During EVA the crew has to communicate with the LTS somehow in order to, for example, open the airlock or ask for status updates. One option is to have a communication device with them that sends data to an LNSP satellite and that satellite relays it back to the LTS. However, this option is very inefficient and time-consuming, so another option was chosen. The crew will have their own personalized tablets, that they bring with themselves to the EVAs and use to communicate with the LTS through a WiFi connection. This allows for a versatile and quick channel, where the components are easily switchable in case the need arises. This is an already existing idea as well within NASA (Figure 11.3); NASA also has similar plans for EVAs [1][55]. Moreover, these tablets would have a holder integrated into the chairs inside the LTS to allow easy access and usefulness "indoors" as well.

Figure 11.3: EVA tablet concept by NASA [1]

11.3. Internal Layout

The user interface architecture can be seen in Figure 11.4, where all the system components are placed in their respective locations ensuring easy accessibility and maintenance. Additionally, the broken down mass and power budgets for the subsystem are presented in Table 11.1.

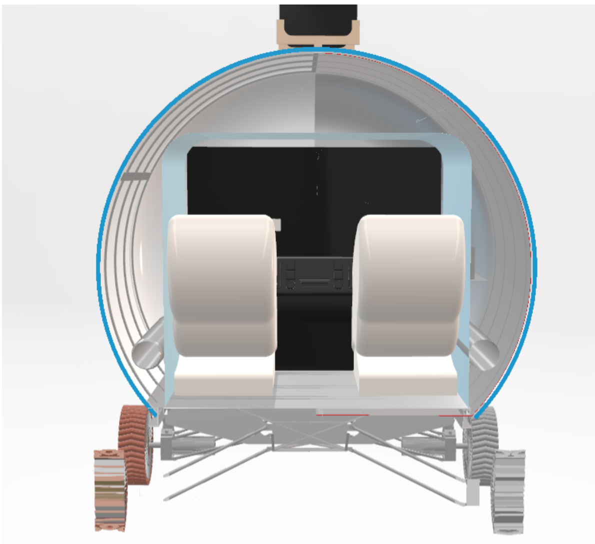


Figure 11.4: Internal layout of the user interfaces

Table 11.1: Mass and power budget of the user interface subsystem.

Component	Quantity	Mass [kg]	Power in active mode [W]	Power in hibernation [W]
Chairs	2	160	40	0
Lights	2	2	20	0
Dashboard Displays	2	2	20	0
Sound System	1	2	10	0
Mechanical Buttons	2	$\sim$0	$\sim$0	0
Window Mimicking System	1	43.5	176	0
EVA Tablets	3	1.2	$\sim$0	0
	Total:	210.7	266	0

11.4. Failure Modes

The reliability and safety of the subsystem is very important this is the crew's way to communicate with the LTS and with other agents. One of the most common ways to analyse component reliability is the FMEA method which ensures that all failures can be tracked, isolated or prevented. The User Interface subsystem's failure mode analysis is presented in Table 11.2. This is not as extensive as it will be once the detailed design is done, however for a preliminary design it is good starting point which later on can be further developed.

Table 11.2: Failure modes of the user interface subsystem.

Component	Failure Mode	Effect	Prevention / Response	Criticality
Chairs	Electrical Failure	Electrical connections fail	Charge the devices with the dashboard, and repair at Basecamp	4
Lights	Single Failure	Lower visibility	Use back up torch	3
	Complete Failure	No visibility	Use back up torch and/or go back to Basecamp to repair	1
Dashboard Displays	Single Failure	Reduced control availability	Use the other display	3
	Complete Failure	No digital control	Use voice control and EVA tablet	2
Sound System	Single Failure	Reduced control availability	Repair whenever the LTS is back at Basecamp	3
Mechanical Buttons	Single Failure	Reduced control availability	Use the other button	2
	Complete Failure	No mechanical control	Go back to Basecamp to repair	1
Window Mimicking System	Total Failure	No visibility to outside	Go back to Basecamp to repair	2
EVA Tablets	Single Failure	Reduced control availability	Use the back up tablet	3
	Complete Failure	No EVA is possible	Go back to Basecamp to repair	2

11.5. Design Analysis

The user interface as a whole offers lots of opportunities for modularity. The dashboard displays will have a standardized power and data connection, thus they can be switched to a new one at any time, or to a completely different model (up until they meet the interface requirements). The same is true for the EVA tablets; they just need to be compatible with the wireless network, but the model of them is of no concern. Additionally, the employed lights are very easy to remove and replace with a new one, or to use a different light source in its place. However, the SAGA Light Panel is the preferred option as the company has a high sustainability profile, and one of the project's objectives is to have a sustainable supply chain for the LTS.

11.6. Compliance Matrix

Table 11.3: Compliance matrix of the user interface subsystem.

Requirement ID [REQ-SSYS-UI-#.#.#.#.#]	Description	Method	Compliance	Justification
1.4.1.1.1	The user interface shall display real-time data such as temperature, vibration, and humidity	Inspection	TBD	See Section 11.2.
1.4.1.1.2	The user interface shall provide a control panel for navigation, temperature, humidity, and energy management	Inspection	TBD	See Section 11.2.
1.4.1.1.3	The user interface shall display real-time mapping	Inspection	TBD	See Section 11.2.
2.3.1.1.1	The user interface shall have a back up control panel	Inspection	TBD	See Section 11.2.
1.5.1.2.1	The user interface shall be able to communicate the need for maintenance to the crew	Inspection	TBD	See Section 11.2.
1.5.1.2.2	The user interface shall alert the user for any malfunction of system	Inspection	TBD	See Section 11.2.
5.7.1.1.1	Development costs of the user interface will not exceed [TBD] euros	Analysis	TBD	Stakeholder meetings concerning monetary budget constraints are planned to take place after the 10-week mark.
5.7.1.2.1	Manufacturing costs of the user interface will not exceed [TBD] euros	Analysis	TBD	See 5.7.1.1.
5.7.1.3.1	Maintenance costs of the user interface will not exceed [TBD] euros	Analysis	TBD	See 5.7.1.1.
5.7.1.4.1	Operational cost of the user interface will not exceed [TBD] euros	Analysis	TBD	See 5.7.1.1.

12 | Cargo Handling

Author: Lee

The cargo handling system is crucial to efficiently manage and transport cargo. To design a robust cargo handling system, it is important to set up comprehensive subsystem requirements first and to then design the system based on these requirements. In Section 12.1, an overview of the designed cargo handling system will be presented. Considering subsystem requirements, the design process of the cargo handling system including the cargo container, cargo compartment, and robot arm will be described in Section 12.2, Section 12.3, and Section 12.4, respectively. Based on the designed cargo handling system, the operational range and measures of modularity will be addressed in Section 12.5 and Section 12.6. Then, the cargo handling mechanism will be explained by presenting a cargo handling architecture in Section 12.7. Also, the assessment of possible failure modes with their criticality for every component of the cargo handling system will be detailed in Section 12.8. Lastly, a compliance matrix will be presented in Section 12.9 to show if the designed cargo handling system meets the requirements.

12.1. Overview

Based on the cargo handling subsystem requirements, an autonomous cargo handling system is to be implemented to carry cargo with a mass carrying capacity up to 50 kg and a volume of up to 135 liters. For an autonomous cargo handling, the cargo is expected to have location data with a RFID tag and to be placed at a designated position on the grid of a cargo compartment by scanning RFID tag with scanner. Then, the mass and volume requirements impact the design of the cargo container, cargo compartment, and robot arm. Also, the cargo handling system shall be designed to withstand the lunar environment. Considering these aspects, the cargo handling system was designed. Figure 12.1 and Figure 12.2 illustrate the detailed 3D designs of the robot arm and entire cargo handling system. The external cargo compartment in Figure 12.2 was determined to be attached to the cabin compartment. The robot arm was made out of carbon-fiber composite tube (XN-70 and XN-80) for robot arm tube and Bulk Metallic Glass(BMG) for its actuator, considering light weight design and operability at extreme lunar temperatures. This designed robot arm has a length of 1.8 m and is equipped with five-claw grippers. For the cargo container and cargo compartment, titanium Alloy (Ti-6Al-4V) was used, and cargo containers with two different sizes were designed for modularity.

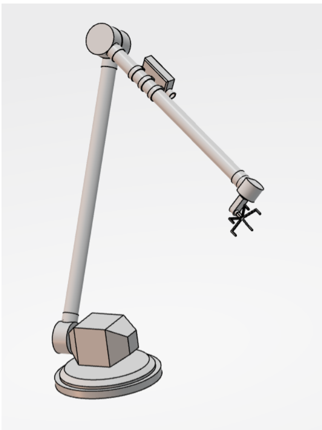


Figure 12.1: The 3D design of robot arm

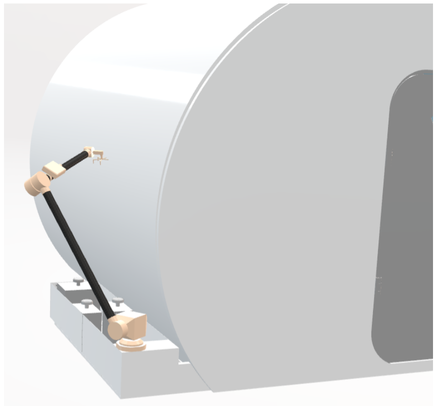


Figure 12.2: The 3D design of cargo handling system

12.2. Cargo definition

For autonomous and efficient cargo handling, it was ideal to design the cargo container to store the cargo in it. To design and determine the size of the cargo container, it was important to define the cargo and consider its size. Also, the cargo container shall have an identifier, such as a RFID tag , with its location data to perform autonomous cargo handling to designated positions. In this section, the cargo sizing and positioning methodology will be explained.

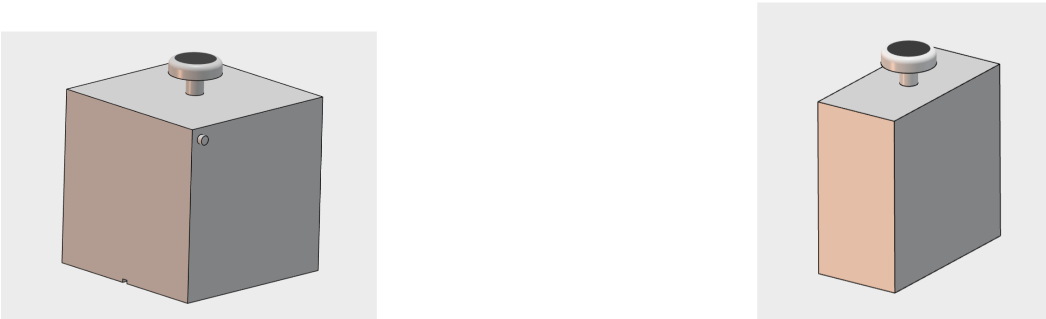
Cargo size and material

The cargo was determined to include components such as repairing kits, sensors and small components of the buggy. Based on the size of these components, it was determined to store these into 5 square containers with a size of (0.3x 0.3x 0.3m). By REQ-SYS-1.2.1.1, the cargo handling system is to carry a payload of up to 50 kg. Therefore, these 5 cargo containers were designed to carry up to 10kg for each. However, this may require the cargo to be stored into more than 5 containers. Therefore, the cargo containers will also be designed to have a maximum of 10 rectangular containers with a size of (0.15x0.3x0.3m) and with a mass carrying capacity up to 5kg for each containers. Table 12.1 shows the type of containers and its dimensions. The determined sizes of the cargo containers comply with REQ-SSYS-CH-1.3.1.1.1. Depending on the type of cargo container, its positioning was determined and this will be explained in Section 12.2

Table 12.1: The container type dimension and weight

Type	Length[m]	Width[m]	Height[m]	Weight[kg]
B	0.3	0.3	0.3	10
A	0.3	0.15	0.3	5

Titanium Alloy (Ti-6Al-4V) was chosen as the material to be used for the cargo container. This material provides a high strength-to-weight ratio, making the cargo container light and strong [56]. Also, this alloy is significantly resistant to oxidation and radiation [56]. Moreover, this alloy can withstand temperatures of up to 350 ℃[56]. Additionally, this alloy provides high fatigue resistance, which is an especially important aspect for the lunar buggy which experiences substantial vibration. Therefore, this material was selected for the cargo container, ensuring protection of the cargo against the lunar environment. Figure 12.3 shows the 3D modelling of the designed cargo container.



(a) The Cargo container (Type-B)

(b) The Cargo container (Type-A)

Figure 12.3: The Cargo containers

Cargo positioning

Each of the cargo containers contains a metallic knob on top of the containers; it is centrally placed. This knob will be handled by the gripper of the robot arm. Sample cargo containers with their metallic knob are shown in the Figure 12.3a and Figure 12.3b . This additional component was designed considering the size of the robot arm gripper. Without this component, the size of the robot arm gripper would have been large and a substantial load would have been carried at the end of the robot arm, requiring a heavier subsystem design and causing reliability and energy efficiency problems.

Then, the the RFID tag was selected to be attached to the surface of this metallic knob. The main reason for choosing RFID tag over QR code is because RFID tag can be read from several meters away[57]. Also, RFID can hold more data and can be read more quickly and simultaneously[57]. This RFID tag contains information such as size, mass, and location. The location will be decided based on each type of cargo container stated in Table 12.1. For example, For example, type-B containers will be placed on B-row while type-A containers will be placed on either A or C row on the grid of cargo compartment shown in Figure 12.4. Because of the size of the cargo compartment, each column on the grid from 1 to 5 can be filled with only 1 B-type container or 2 type-A containers. Considering this, the cargo placement will be processed in a way that none of A-type containers will be picked up in the presence of a B-type container with same column number. This RFID tag will be protected by encapsulation it with a transparent and durable material, and two RFID tags will be attached to each of cargo container for redundancy.

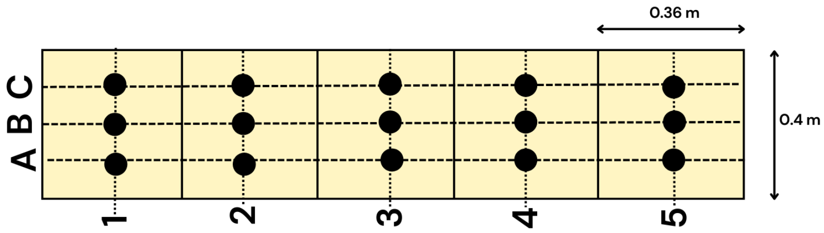


Figure 12.4: The cargo container location

12.3. Cargo compartment

Because of limited space in the cabin compartment, the external cargo compartment was determined to be attached to the cabin compartment. The cargo compartment was designed to place the cargo container at specified position. From Section 12.2, the size of the cargo container was determined. Besides the cargo container, the robot arm, lidar, and stereo camera were decided to be attached to the cargo compartment. The robot arm will pick up and manipulate the cargo container while the stereo camera will continuously check if the cargo container is placed correctly in the cargo compartment. Lastly, the lidar will detect obstacles at a distance.

Considering the cargo container size and the placement of the lidar, robot arm, and camera, the size for the cargo compartment was determined to be 2 x0.4x0.4 m with a thickness of 1.5mm for the side part and 2.5mm for the bottom part. This was to ensure that there is some marginal spacing between each of the cargo's containers. The cargo partition was implemented as shown in Figure 12.5. This partition is to ensure that the cargo container does not slide to other sections and that it stays at its specified position throughout the mission. This partition was designed to be adaptable by upward sliding, taking into account modularity. A length of 2 meters was divided into two section: a 1.8 meter length for the cargo placement and a 0.2 meter part for the placement of the robot arm, lidar, and camera.

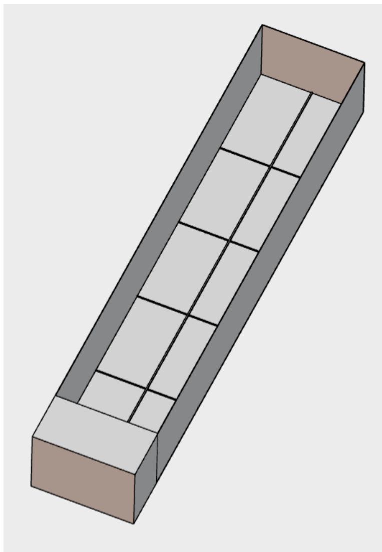


Figure 12.5: The cargo container compartment

12.4. Robot Arm Design

To be able to pick up and control the object, designing the robot arm is essential. In designing the robot arm, the degree of freedom (DOF) was determined considering multiple factors, such as object shapes and the terrain. Based on the chosen DOF, the size of the robot arm and multiple components, such as the actuator and bearing, were designed, considering the lunar environment and the size of the cargo compartment and cargo containers. The DOF selection and robot arm design choices will be explained in this section.

Degrees of Freedom

The first design consideration for the robot arm was choosing the degree of freedom of the robot arm. With a higher degree of freedom, the robot arm can perform more flexible maneuver. However, the more degrees of freedom are included, the more actuators will be needed and the heavier the design will be. Thus, the trade-off for the DOF was evaluated based on several criteria with their respective weights as shown in Table 12.2

Table 12.2: Trade-off Analysis for Robotic Arm Degrees of Freedom (DOF)

Criteria	Weight	3DOF	4DOF	5DOF	6DOF
Reliability	0.35	4	4	3	2
Flexibility	0.35	2	3	4	5
Weight	0.15	5	4	3	2
Energy efficiency	0.15	5	4	3	2
Overall Score		3.6	3.65	3.35	3.05

The *reliability* and *flexibility* criteria were given the highest weight of 0.35 while *weight* and *energy efficiency* were given a relatively low weight of 0.15. This is because reliability and flexibility are directly linked to functional failures and operational capability unlike the other two criteria. More actuators and moving parts are required with higher degrees of freedom. This means that the robot arm can perform more flexible maneuvers and grip the object more smoothly from various angles. However, having more actuators and moving parts, impacts the overall reliability, adds weight and consumes more power. Considering these pros and cons with higher DOF, the scoring for each criteria was assigned with respective weights, as portrayed in Table 12.2. Ultimately, the 4DOF arm was identified as the most suitable choice, offering the best compromise between reliability, flexibility, weight, and energy efficiency.

Robot arm design choices

The robot arm shall be designed to withstand lunar temperatures and radiation. Therefore, the technology used for the robot arm from other space missions was applied in designing the LTS' robot arm. It was determined that the robot arm would operate with four DOF in Section 12.4. Consequentially, four DOF robot arms used in space missions were considered. The robot arm is to be a minimum pf 1.8 meter to be able to place the cargo in cargo compartments. Considering multifaceted aspects, it was determined to incorporate the materials and components used from the Instrument Deployment Arm (IDA) and Cold Operable Lunar Deployable Arm (COLDArm). The reason for choosing these 2 space robot arms is that IDA is an extremely lightweight robotic arm while COLDArm is the only operable robot arm able to withstand lunar nights in the absence of an electric heater. Also, the robot arm load carrying capability shall be considered. The robot arm shall carry up to 16.2N on the lunar surface because the maximum mass capacity of a cargo container was determined to be 10kg in Section 12.2. The IDA can carry cargo up to 33 N while the COLDArm can carry cargo up to 40 N [58][59]. Therefore, incorporating the components and materials from IDA and COLDArm will definitely satisfy the load carrying capability. The selection of material and components for the designed robot arm will be further explained in this section.

Carbon-fiber composite tube

For a light design of the tube part, the designed robot arm was selected to employ the material used for IDA. The Upper-arm and forearm tubes are made of XN-70 and XN-80 carbon-fiber composites combined with aluminium and titanium. The tube walls have a length of 1 m and 0.8 m with thicknesses of 0.8 mm and 1.1mm, respectively [60]. The XN-70 and XN-80 composites provide significantly low density, making the robot arm light [61]. These composites also provide significantly high stiffness and tensile strength, which enables maintaining the structural integrity and precision of a robot arm under load[61]. Further, these carbon composites have really low thermal expansion and high thermal conductivity, which protects the robot arm from deformation caused by extreme lunar temperatures and overheating through efficient heat dissipation[61]. The detailed overview of material properties for XN-70 and XN-80 composites is provided in Table 12.3.

Material	Young's Modulus (GPa)	Tensile Strength (MPa)	CTE (PPM/°C)	Thermal Conductivity (W/m-K)	Density (g/cm^3)
XN-70	724.975	3654.35	-1.62	311.58	2.159
XN-80	785.43	3654.35	-1.62	406.785	2.159

Table 12.3: XN composite property[61]

BMG Actuators
The actuators are vulnerable to lunar temperatures, which can cause operational failure. Specifically, most of the actuators are not operational without an electric heater during lunar nights due to extreme cold temperatures. The Bulk Metallic Glass(BMG) actuator used in COLDArm is the only actuator which can withstand lunar nights [59]. BMG actuators are actuators made from Bulk Metallic Glass alloys; they are solid metallic materials with an amorphous atomic structure[62]. This amorphous structure of BMGs has unique mechanical properties, including high strength, excellent wear resistance, corrosion resistance and low temperature capability. This alloy is stronger and tougher than conventional materials like steel and ceramics and can operate at temperatures down to -173 ℃[62]. Therefore, the BMG actuator significantly reduces the weight, complexity, and power consumption of the robot arm, because it does not require additional heating systems unlike other actuators. Considering the aforementioned advantages, the BMG actuators were determined to be used for the design of the robot arm.

DACEE Motor controller
The motor controllers are essential for precise control of the robot arm and shall be designed considering the lunar environment. The Dual-Axis Controller for Extreme Environments (DACEE) motor controllers, developed by Motiv under the Small Business Innovative Research (SBIR) program, are capable of operating at lunar nights [59]. DACEE has been specifically designed to operate at temperatures of -190℃, and this has been successfully demonstrated by the ColdArm project [59]. This controller eliminates reliance on Warm Electronic Boxes (WEB) and eliminates the need for heaters, which improves efficiency and longevity by lowering the mass and power consumption[59]. As a back up, in the case of failure of the controller, two DACEE motor controllers were chosen to be used.

Sensors
The robot arm will use the inputs of the force torque sensors, RFID tag reader, IMU and 3D stereo camera. The Cryogenic Capable Six-Axis Force Torque Sensor from IDA will be used for fault protection, precise ground interaction and load control in extreme cold temperatures. Then, the RFID tag reader will be attached near the end effector to read the RFID tag which is attached on the metallic knob of the cargo container. Also, the IMU will be used to measure the acceleration and angular rate of the robot arm while the 3D stereo camera will be used for accurate positioning of objects. The stereo camera and RFID tag readers will be protected by

MultiLayer Insulation (MLI) from extreme lunar temperature. For redundancy, back up sensors will be brought for the mission.

A five-claw gripper

The primary purpose in designing the cargo handling system was to pick up and position the cargo container to designated places. Therefore, it was determined to use a gripper as an end effector. The cargo container has the circular shape of the metallic knob which enables small sizing of the end effector. A five-finger gripper can grapple and securely hold of this metallic knob, ensuring precise handling and placement of the cargo container. Additionally, the five-finger design offers redundancy, as it remains operational even if one finger fails, unlike a two-finger gripper which would critically fail if one finger were to malfunction.

12.5. Operational Range

The designed cargo system has a robot arm with a length of 1.8 m and will be placed on the upper surface of the cargo compartment with a height of 0.3 m. The cargo compartment will be mounted on the cabin compartment positioned 0.5 m above the ground. Considering these dimensions and positions, the operational range for the robot arm was determined as shown in Figure 12.6. Only the cargo container within this range can be picked up by the robot arm. Therefore, the cargo handling system must be integrated with the Guidance, Navigation, and Control (GNC) subsystem to ensure that the buggy approaches the cargo containers within this operational range for cargo handling.

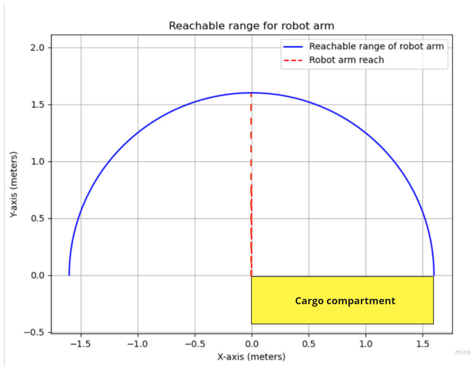


Figure 12.6: The cargo handling operational range

12.6. Modularity

Firstly, the cargo container was classified into two different categories with different sizing and mass capacities in Section 12.2. Classifying cargo containers into two distinct categories enhances modularity by accommodating a wide range of payloads efficiently. The cargo partition was designed for autonomous cargo handling to separate the cargo container to a designated position. The cargo partition shown in Figure 12.5 was designed to be removable by upward sliding considering modularity. The different sizes and shapes of the cargo, rather than the defined cargo containers, can be stored in the cargo compartment by just removing the cargo partition. Also, the various end effectors can be utilized for different tasks. For example, scoops and drills can be used to collect lunar regolith and core samples, respectively.

12.7. Cargo Handling System Architecture

The Figure 12.7 demonstrates the cargo handling architecture and an the overview of the technical flow of the cargo handling system. This architecture shows the interaction between sensors and actuators, highlighting the specific functions of each component. In this section, this architecture will be further explained.

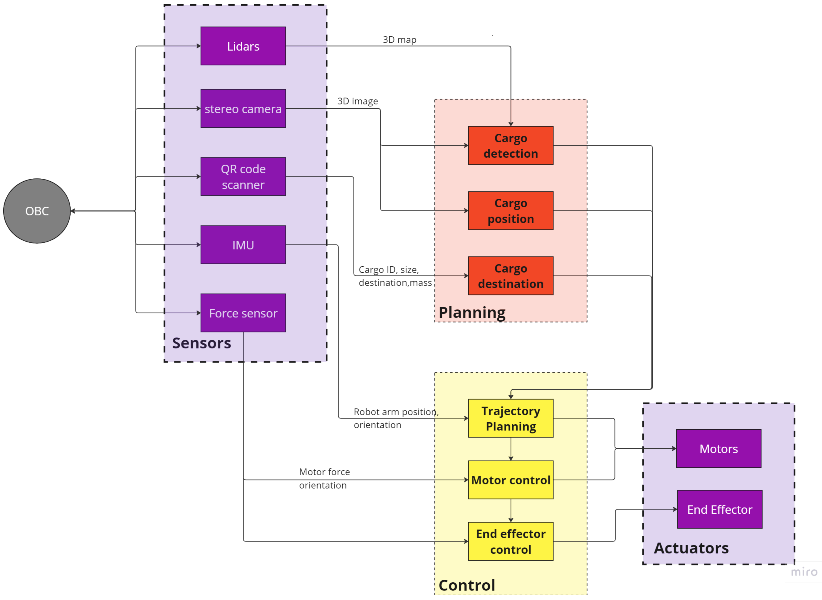


Figure 12.7: The cargo handling architecture

For the sensors, there are lidars, stereo cameras, RFID tag scanners, IMU, and force sensors. The lidar is placed on cargo compartment and it detects the cargo. Then, the lidar will measure the distance of the cargo and the coordinates with the GNC subsystem to ensure that the buggy approaches the cargo within the operational range defined in Section 12.5. There are a total of two stereo cameras: one placed on the cargo compartment and the other one placed on the robot arm. The stereo camera on the cargo compartment will continuously monitor the cargo position by capturing 3D images. This is to ensure that the cargo is positioned at a specified position. The stereo camera on the robot arm will detect the cargo and its size. The RFID tag scanner will encode the RFID tag on the cargo container, providing the cargo id, size, destination, and mass. The size information from the RFID tag will be verified by comparing the size detected from the stereo camera on the robot arm. With these sensors, the cargo position and destination are determined. Additionally, an IMU measures the robot arm's position and orientation. A force sensor measures the interaction forces between the robot arm and the cargo. Combining data from these sensors enables for a precise trajectory planning, motor control and end effector control.

12.8. Failure Modes

Table 12.4: Failure modes of cargo handling system

Component	Failure Mode	Effect	Prevention / Response	Criticality
Cargo container	Structural Failure	Inability to store cargo	Repair	3
Cargo compartment	Structural Failure	Inability to store the cargo container	Return back to base for repair	2
RFID tag	Single Failure	Insignificant	Use backup	3
	Complete Failure	Inability to identify the cargo container	Manual robot arm control	2
Sensors	Single Failure	Insignificant	Use backup	3
	Complete Failure	Inability to detect object	Return back to base for repair	1
Gripper	Single Failure	Use backup	Repair on the spot	3
	Complete Failure	Inability to pick up the cargo	Return back to base for repair	2
Actuator	Mechanical failure	Inability of rotating robot arm	Return back to base for repair	2
Motor controller	Single Failure	Insignificant	Use back up	2
	Complete Failure	Loss of cargo handling control	Return back to base for repair	1

12.9. Compliance Matrix

Table 12.5: Compliance matrix of cargo handling subsystem

Requirement ID [REQ-SSYS-TC-#.#.#.#.#]	Description	Method	Compli-ance	Justification
1.2.1.1.1	The LTS shall be able to lift a cargo payload of up to 50 kg	Test	TBD	see Section 12.4
1.2.1.1.2	The LTS cargo handling subsystem shall have a reach of 1.8 m	Demonstration	TBD	see Section 12.4
1.3.1.1.1	The LTS cargo handling subsystem shall be able to support a cargo payload of up to 135 liters	Demonstration	TBD	see Section 12.2
3.3.1.1.1	The LTS shall be able to load the cargo autonomously	Test	TBD	see Section 12.2
3.3.1.1.2	The LTS shall be able to unload the cargo autonomously	Test	TBD	see Section 12.2
5.2.1.1.1	The LTS cargo handling subsystem shall employ a modular design	Test	Yes	see Section 12.6

13 | Thermal Control Subsystem

Author: Bart

In this chapter, the first in depth design of the thermal subsystem for the LTS will be discussed. First, an overview will be given of the thermal subsystems functions and relevance in Section 13.1. Also, the main layout and characteristics of the thermal subsystem will be presented. The simulation method for the thermal subsystem will be discussed in Section 13.2. Then, based on the simulation results, the passive and active thermal control components will be sized/chosen in Section 13.3 and Section 13.4. When all parts have been defined, the thermal subsystem reliability, modularity, sustainability and performance will be evaluated in Section 13.5 and Section 13.6. Finally, the cost, mass and power budgets for the thermal subsystem will be presented in Section 13.7.

13.1. Thermal Subsystem Overview

The thermal control subsystem is responsible for controlling the temperature of the LTS and all its components. This includes the pressurized cabin, electronics such as the OBC, the fuelcell, the motors and the tanks with fluids like H_2, O_2 and water.

The thermal control subsystem is crucial for the LTS design since the LTS will accommodate a crew. This also means the life support subsystem is very important for the success of this design, more about this can be read in Chapter 7. The life support subsystem demands a very specific temperature to accommodate the crew, see Table 13.1. The constraints originating from the life support subsystem are one of the leading factors in the thermal subsystem design. Without a good thermal control subsystem, the LTS would overheat or freeze when introduced to the extreme temperatures on the lunar surface. Furthermore, the temperature regulation of the electronics and the fuel cell is essential to the performance of the LTS.

The thermal control subsystem consists of active and passive control elements. The active thermal control elements actively force heat flows form or to certain components, these elements require energy to work. Examples of active thermal control elements are fans, heat pumps and radiators. Passive control elements resist or induce convection, induction or radiative heat flows. An example of a passive thermal control element is insulation.

The main elements of the thermal subsystem for the LTS are shown in Figure 13.1. The blue lines represent the earogel used to insulate the lunar rover. The red lines show the red pipes, representing the pipes through which ammonia is pumped from the fuel cell towards the radiators and other parts. The grey lines represent the regolith used mainly for radiation protection and shielding against the lunar environment. The squares with "Hx" act as the heat exchanges in the thermal control system.

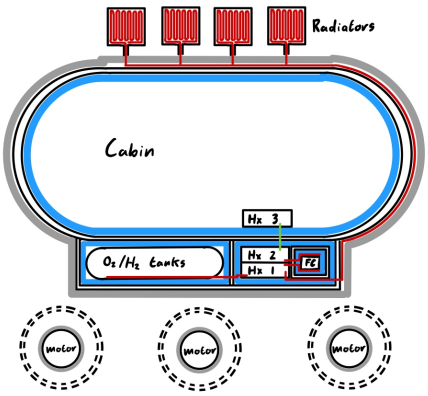


Figure 13.1: Sketch of the thermal subsystem layout with most important parts

Figure 13.2 shows the layout of the thermal subsystem with all relevant heat flows and the most important elements of the subsystem. The red blocks represent elements that create heat and the blue block represents elements that disperse heat. The meaning of the grey blocks will be discussed in Section 13.2. The arrows indicate the heat flows between the elements and the environment.

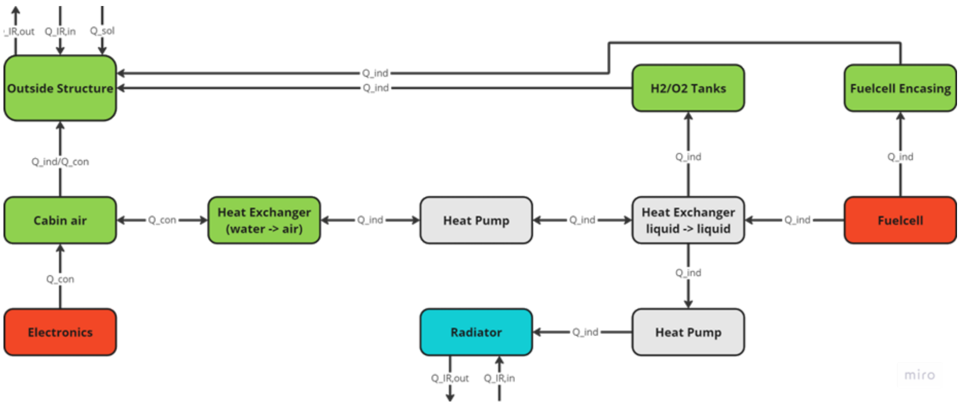


Figure 13.2: Diagram of the thermal subsystem showing relevant components and heat flows

13.2. Thermal Subsystem Simulation

Table 13.1 lists all the thermal constraints for all relevant elements for the thermal control subsystem. To evaluate if the thermal control system can adhere to these constraints, the thermal flows will be simulated.

Table 13.1: Table with thermal constrains of the main elements for the thermal control subsystem

Component	Survivable temperature range [K]	Operational Temperature range [K]
Fuel cell	243-373	303-353
Electronics	218-398	218-398
Motors	23-673	23-673
H2-tank	233-358	233-358
O2-tank	233-358	233-358
Cabin air	283-308	293-298
Antenna	153-393	153-393
Outside structure	20-673	20-673
Elements that use ammonia (radiators)	210-373	210-373

To size the thermal control subsystem, first the heat flows going in and out every main element listed in Table 13.1 need to be simulated. The simulation of temperatures of these parts, and of the heat flows between the parts, is crucial for finding the required performance of the active thermal control element and for sizing the thickness of all the insulation used in the LTS. The simulation considers 8 elements, namely: the outside structure, the cabin air and contents, the heat exchanger in the cabin, the $H_2/0_2$ tanks, the radiator outside, the fuelcell encasing, the fuelcell and the motors. Figure 13.3 shows the relevant heat flows between the elements. The red boxes indicate elements that provide heat and the blue box represents elements that reject/dissipate heat. The elements in grey boxes are part of the active thermal control and help control the heat flow. However, they do not have a simulated temperature. The motors have their separate diagram shown in Figure 13.4. Note that there are six motors, however the motors all have the same condition. Thus, only one motor is simulated.

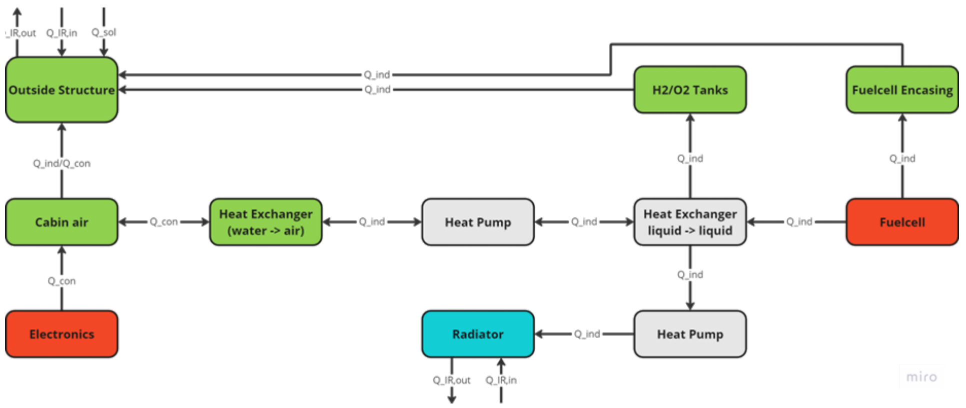


Figure 13.3: Diagram of the thermal subsystem showing relevant components and heat flows

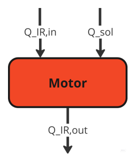


Figure 13.4: Diagram of the thermal subsystem showing relevant heat flows for the motors

Simulation Assumptions

To simulate the heat flows properly within the time constraints of this design process, several simplifications and assumptions have been made:

1. Insulators do not have temperatures, thus only the thermal resistance of the insulators is simulated
2. Heat is spread equally over the elements that have a designated temperature.
3. The power going into the electronics inside the cabin is all converted into heat
4. The electronics have builtin thermal regulation and all heat will be dissipated to the air inside the cabin
5. The H_2/O_2 tanks are simplified with one element
6. The motors are separated from the rest of the vehicle in terms of thermal induction
7. The cabin is simplified to a cylinder with a flat base
8. The cabin and structure are made from alumium without surface treatment

Assumption 2, has a big impact on the accuracy of the simulation. However, this assumption is crucial to making a simple program. The assumption is justified by the great thermal conductivity of the material out of which most of the structure is made: aluminium. Aluminium has a thermal conductivity of 240 W m^{-1} K^{-1} [63]. Also, assumption 4 is easily justified since most electronics used on earth also have their own temperature regulation and make sure that their temperatures stay within a margin of 20-100 ℃. The motors are treated separate from the LTS since they are connected through considerably small slender rods which means the inductive heat flow through these rods will be very low. The infrared radiation from the cabin towards the motor has not been neglected. Finally, assumption 7 is very reasonable since this assumption is conservative. The area of a flat cylinder with the same total length as the area of a cylinder with hemispheres is bigger.

Furthermore, the following heat flows and elements are neglected. The reason is either because there was not enough time to simulate all the elements in detail or because the element was not relevant in this stage of the design. Elements and heat flows that were ignored are:

1. The connection from the underside of the chassis to the wheels is neglected
2. The wheels are neglected
3. The antenna is not simulated because there was no time to integrate this part in the simulation, since it was added later during the design phase
4. The heat pumps do not have heat-loss, neither do the heat exchangers
5. The radiators are pointed perpendicular to the sun and to the LTS which means no heat from either solar radiation or IR radiation will be received by the radiators

Before discussing the formulas used for the simulation, several symbols will be introduced to reference certain elements.

Table 13.2: Reference numbers for simulated elements

Reference Number	Element Name
1	Cabin air and contents
2	Outside structure
3	Inside + fan
4	Radiator outside
5	Fuelcell
6	Fuelcell encasing
7	$H_2/0_2$ tanks

Simulation Formulas

Before making a system of equations, first the formulas that give the heat flows need to be determined. The formula for heat flow through induction between two elements with temperatures T_A and T_B is:

$$\dot{Q}_{ind_{A,B}} = \frac{kA}{t}(T_A - T_B) \tag{13.1}$$

Furthermore, the heat flow through radiation from an element outwards and the heat flow through absorption of radiation can be calculated using the following formulas.

$$\dot{Q}_{IR_A} = \sigma\epsilon_A A_{out} T_A^4 \tag{13.2}$$

$$\dot{Q}_{in_{B,A}} = \sigma\epsilon_B T_B^4 \alpha_A A_{A_{in}} \tag{13.3}$$

with:

$\sigma =$	boltsmann constant	$\epsilon =$	emissivity
$\alpha =$	absorptivity	$A_{out} =$	the emmissive area
$A_{A_{in}} =$	the absorbing area	$T_A =$	the temperature of the object radiating
$T_B =$	the temperature of the object emitting heat towards another element.		

Also, note that if there is a distance between the emitting body and the absorbing body, the quadratic law needs to be used to calculate the reduction in thermal flux:

$$\Phi_A r_A^2 = \Phi_B r_B^2 \tag{13.4}$$

Furthermore, the solar radiation, Q_{sol_A}, is dependent on the solar flux intensity on the moon and can be calculated using:

$$Q_{sol_A} = \Phi \alpha_A S_A \tag{13.5}$$

with:
$S_A =$ the effective area in the sun $\Phi =$ the solar intensity on the moon $\alpha_A =$ the absorptivity of the absorbing body
Finally, heat flow through convection can be calculated using the following equations:

$$\dot{Q}_{con_{A,B}} = hA(T_A - T_B) \tag{13.6}$$

$$h = 10.45 - v_{air} + 10\sqrt{v_{air}} \tag{13.7}$$

with:
$v_{air} =$ the velocity of the air flowing along the heat exchangers surface
$A =$ the surface that is exchanging heat
$T_A =$ the temperature of the air
$T_B =$ the temperature of the surface
$h =$ the convective coefficient

Note that the equation for the convective coefficient is semi-empircal and is only applicable for a standard atmosphere gas mixture. The following system of equations can be found up by summing up all the heat flows between the parts

$$
\begin{aligned}
Q_1 &= -Q_{con_{1,3}} - Q_{ind_{1,2}} + Q_{elec} \\
Q_2 &= Q_{ind_{1,2}} - Q_{ind_{2,6}} - Q_{ind_{2,7}} + Q_{solar_2} + Q_{IR_{moon,2}} + Q_{IR_{space,2}} - Q_{IR_2} \\
Q_3 &= Q_{con_{1,3}} - Q_{hx_{3,5}} - Q_{hx_{3,4}} \\
Q_4 &= -Q_{hx_{4,5}} + Q_{ir_{moon,4}} + Q_{ir_{space,4}} - Q_{IR_4} + Q_{hx_{3,4}} \\
Q_5 &= Q_{hx_{3,5}} + Q_{hx_{4,5}} - Q_{ind_{5,6}} - Q_{hx_{5,7}} + Q_{fuelcell} \\
Q_6 &= Q_{ind_{2,6}} + Q_{ind_{5,6}} \\
Q_7 &= Q_{ind_{2,7}} + Q_{hx_{5,7}} \\
Q_8 &= Q_{motor} + Q_{IR_{moon,8}} + Q_{IR_{2,8}} + Q_{sol_8} - Q_{IR_8}
\end{aligned}
\tag{13.8}
$$

Note that $Q_{hx_{A,B}}$ represents the amount of heat displaced by a heat pump from A to B. This is a fixed amount since heat pumps can control heat flow using power. Also, Q_{elec} is the heat produced by the electronics in the cabin, $Q_{fuelcell}$ is the heat produced by the fuel cell and Q_{motor} is the heat produced by a motor. Unfortunately, this system of equations is not linear. The heat flows through radiation are dependent on T^4 while the induction and convection heat flows scale with T. This is why an iterative solver will be used to calculate the heat flows. The numerical procedure used to solve for the temperatures is Forward Euler. To calculate the temperatures per part the following equation is used:

$$T_{i_{n+1}} = T_{i_n} + \frac{Q_{i_n}}{m_i c_i} dt \tag{13.9}$$

Where i represents the part number and n represents the time step. Using this formula, also the mass, m, and specific heat, c of the different parts is required. However, this simulation focuses on the temperatures the LTS converges to. Thus, the transients will be ignored. This also means the mass and specific heat can be chosen arbitrarily since those terms only influence the transients and their effect will become zero for time going to infinity. For this simulation, the parts are simulated as solid blocks of aluminium with an exception for the cabin air, which will be simulated with properties from ISA.

Simulation Results

The simulation takes two scenarios into account: The LTS in the sun and the LTS in the shadow. In the sun, the solar flux is very high, $\Phi = 1361$ W m^{-2}, which heats up the LTS considerably. Also, the black body temperature of the lunar surface in the sun becomes relatively high, on average $T_{moon} = 200$ K. If the LTS is in the shadow, the solar flux becomes zero and the black body temperature of the moon becomes on average $T_{moon} = 75$ K. For a heat map of the South Pole showing the temperatures in different areas, see Figure 3.5. To be conservative, the temperatures in the shadow have been taken as $T_{moon} = 40$ K. The temperature of the Moon in the Sun has been assumed as just the average temperature since an overestimate of this temperature has more effect on the active thermal system sizing.

Using the equations stated above and the inputs found in Table 13.3 and Table 13.3, the temperatures for each part where analysed. The heat flows of the heat pumps, the size of the radiators outside the cabin and inside the cabin and the airflow, V_{air}, were adjusted to ensure that the temperatures of all parts stayed withing the limits stated in Table 13.1. Furthermore, the insulation thickness was adjusted to get the optimal combination between active and passive thermal control.

Table 13.3: Material characteristics and constants for inputs

Symbol	Value	Unit
ρ_{air} [64]	1.225	kg m^{-3}
ρ_{alu} [63]	2900	kg m^{-3}
$\rho_{aerogel}$ [1]	300	kg m^{-3}
c_{air} [64]	1005	J kg^{-1}
c_{alu} [63]	900	J kg^{-1}
$k_{earogel}$ [2]	0.024	W m^{-1} K^{-1}
$\alpha_{regolith}$ [65]	0.88	-
α_{alu} [63]	0.2	-
α_{carbon} [3]	0.98	-
$\epsilon_{regolith}$ [65]	0.95	-
ϵ_{alu} [63]	0.2	-
ϵ_{carbon} [4]	0.98	-

Symbol	Value	Unit
σ [63]	$5.6704 \cdot 10^{-8}$	W m^{-2} K^{-4}
Φ [5]	1361	W m^{-2}
$T_{moon,shade}$	40	K
$T_{moon,sun}$	200	K
T_{space} [6]	4	K
Q_{elec}	580	W
$Q_{fuelcell}$	2547	W
Q_{motor}	25	W

Note that in addition to these inputs, there are areas and sizes of the LTS defined in the simulation; these areas adhere to the LTS design. Sometimes the areas are conservative, and sometimes they are liberal depending on the role of the area inside the thermal simulation. The full code can be found in the github repository [7]. The output temperatures of the systems are given in Table 13.4, the converged control variables are given in Table 13.5 and the sizes for the passive thermal control are listed in Table 13.6.

Table 13.4: Part temperature outputs

Part	Temperature in shadow [K]	Temperature in sun [K]
1	298.41	292.70
2	235.84	283.81
3	334.56	279.34
4	244.33	337.52
5	353.89	351.38
6	282.97	310.80
7	294.68	289.17
8	205.72	309.69

Table 13.5: Final active thermal control variables from the simulation

Variables	Value in shadow	Value in sun	Unit
$Q_{hx_{3,4}}$	0	300	W
$Q_{hx_{5,7}}$	330	30	W
$Q_{con_{1,3}}$	-1390	300	W
$Q_{hx_{3,5}}$	-1390	0	W
$Q_{hx_{4,5}}$	-823.5	-2515	W
$A_{radiators}$	4	4	m^2

Note that the temperatures of the cabin are in shadow 25.41 $^\circ$ C and in the sun 19.70 $^\circ$ C. This is not within the constraints stated in Table 13.1. This is because using a higher temperature in the shadow is more constraining. The same applies to the lower temperature in the sun. This ensures that the system is capable of keeping the right temperatures. Also, since the thermal control subsystem has active thermal control, the temperature in the cabin can be controlled to be within the specified boundaries.

Table 13.6: Insulation thicknesses and areas

Insulation part	Thickness [m]	Area [m^2]
Cabin/outside structure	0.05	65.6
Fuel cell shell/outside structure	0.03	0.09
Tanks/outside structure	0.03	7
Fuel cell/fuel cell shell	0.05	0.1

[7] https://github.com/TUD-AE/DSE2023-24-Q4-project18

Note that the areas of the insulation has not been changed in the simulation. Also, the area of the fuel cell to fuel cell shell is bigger than the fuel cell shell to outside structure since on one side the fuel cell shell is replaced by a heat exchanger. Also, not all sides of the fuel cell shell are connected to the outside of the structure.

13.3. Passive Thermal Control: Part Design & Selection

The passive thermal control consists of any thermal control elements that do not use power or energy. For the LTS, this is all the insulation present on the lunar rover. Figure 13.5 shows an overview of the main elements in the thermal control subsystem. The blue lines are aerogel insulation, and the grey lines represent regolith.

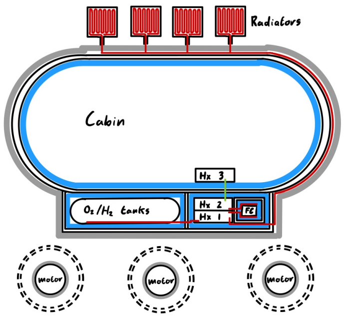


Figure 13.5: Sketch of the thermal subsystem layout with most important parts

The regolith shield, which has as main function to protect against micrometeorite impacts, is packed in an aluminum structure. This means the regolith shield is effectively aluminium in terms of radiative properties. In the simulation, it was also treated as aluminium. The regolith shield is further discussed in Chapter 7. Additionally, the cabin wall under the shield, as well as the chassis, is modeled as only one element in the simulation namely, the outside structure. The sizing of this element is discussed in Chapter 6.

The cabin insulation is aerogel; the values from the simulation found in Table 13.6 are: $A = 65.5$ m^2 and $t = 0.05$ m. The aerogel insulation has been chosen because of the combination of low density and low thermal conductivity. Aerogel manufacturing technologies have become increasingly sophisticated and silica based aerogel itself is getting cheaper. The fuel cell insulation and fuel cell shell insulation are both also made from aerogel. The fuel cell insulation dimensions are: $A = 0.1$ m^2 and $t = 0.05$ m. The fuel cell shell dimensions are $A = 0.09$ m^2 and $t = 0.03$ m. The last part made from aerogel is the insulation around the oxygen and hydrogen tanks, the measurements for this insulation are: $A = 7$ m^2 and $t = 0.03$ m.

Next to the aerogel parts, the fuel cell also has an aluminium shell to confine the first layer of earogel and the fuel cell itself. This fuel cell shell has a thickness of 0.005 m and an area of approximately $A = 0.09$ m^2.

Lastly, to increase the radiative properties of specific parts, a layer of regolith was added. For the engine, this layer of regolith could be very thin. However, to make sure the caps would not brake during handling, the thickness was increased to a thickness of 0.005 m. The regolith will cover the full area of the motor. Regolith has good radiative properties; the emmissivity and absorptivity respectively are: $\epsilon = 0.95$ and $\alpha = 0.88$. Since most of the motor is placed in shadow, the application of the regolith layer has a cooling effect on the part. Also, for the radiators, a special material was used to increase the radiative properties. Carbon powder or graphite powder was chosen. By applying graphite powder on the radiators the radiative properties increase by a factor of five: $\alpha_{alu} = 0.2$ and $\alpha_{carbon} = 0.98$. Further, sizing of the radiators will be discussed in Section 13.4.

13.4. Active Thermal Control: Part Design & Selection

The active thermal control consists of elements that require power or can be adjusted to improve or decrease their thermal conductivity or other characteristics. For the LTS, the design was driven to be as energy efficient as possible. This means a lot of passive thermal control is present, and the active thermal control is kept to a minimum. The main parts for the active thermal control are: heat pumps, heat exchangers, plumbing for cooling liquids, pumps for cooling liquids, radiators, tanks for storing cooling liquids, temperature sensors, flow speed sensors. This section will discuss the selection and/or sizing of all the active thermal components.

Heat Exchangers

To realize the heat flows stated in Table 13.5, heat exchangers and heat pumps are required. The cooling system for the fuel cell will function on ammonia as coolant, since ammonia has a low freezing point ($T_{m,ammonia}$ = -77.73 $^\circ$ C at standard atmosphere) [8]. Also, ammonia stays liquid for temperatures up to 100 $^\circ$ C if a pressure of at least 60 bar is applied to the fluid, see Figure 13.6. Due to its toxic nature, ammonia is only used outside the pressure vessel. For heat pumps and heat transfer inside the pressure vessel, water is used. Water is not used outside the pressure vessel since the outside structure temperatures can drop below zero. Water freezes at zero degrees, freezing of the cooling system would mean the cooling system fails since the water will come to a stand still.

The active thermal subsystem has two water to ammonia heat exchangers. For a first estimation, the heat exchangers from the ISS thermal control system will be used. Also, these heat exchangers are designed for water to ammonia heat exchange. One heat exchanger has a thermal flow performance of about 7000 W [8]. According to Table 13.5, the maximum heat flow between the cabin heat exchanger and the water to ammonia heat exchanger is $|Q_{hx_{3,5}}| =$ 1390 W. So one heat exchanger could easily manage the heatflows required. However, one of the requirements for the thermal control system is about having a backup system. Thus, two heat exchangers are assigned to the thermal control subsystem. One heat exchanger has a mass of 41.28 kg and measures 635 x 530 x 203.2 mm [8]. For the final version of the thermal control subsystem, a custom heat exchanger will be designed to downsize this heat exchanger. However, this will be discussed in the second-order detailed design phase after the 10-week mark.

The second pair of heat exchangers are water to air heat exchangers. These are required to provide a heat flow of at least $|Q_{con_{1,3}}| = 1390$ W to keep the cabin at the right temperature. The convective heat flow was first calculated with formulas, see Section 13.2. The speed of the airflow was set to a maximum of 7 m s^{-1} during the simulation. This shows that the airflow towards a convective heat exchanger does not need to blow unreasonably hard to provide such amounts of power. After finding the required heat flow, parts were selected from ATS

[8] https://www.nasa.gov/wp-content/uploads/2021/02/473486main_iss_atcs_overview.pdf

Thermal Solutions. The ATS-HE21 Series can provide 46 W per ΔT = 1 K [9]. This means the maximum temperature the radiator will operate at is:

$$\frac{1390}{46} + 25 = 30.22 + 25 = 55.22[^{\circ}C]$$

This is a reasonable temperature for a heater or heat exchanger between liquid and air. The weight of one heat exchanger is 6.569 kg, and the size of one heat exchanger with corresponding fans is 266.7 x 147.7 x 10.2 mm [9].

Heat Pumps

Since the temperature differences between the fuel cell and the radiators can be very small: $T_5 - T_4 = 337.52 - 351.38 = 13.86$ K, the active thermal control needs to use heat pumps to force the right amount of heat towards the radiators. For the distance between the radiators and the fuel cell, it is not feasible to enable a heat flow of 2815 through induction. Furthermore, due to the change in scenarios, it is also important to have a way to control heat flow, which is enabled by heat pumps. A heat pump requires a refrigerant fluid with the right properties to evaporate and condensate at precise pressure and temperature combinations. The refrigerant will be discussed in detail in the second-order detailed design phase after the 10-week mark. It can be assumed that a well designed heat pump has a coefficient of performance (COP) of 5 [10]. Using this assumption, the power required to run a heat pump can be calculated. The required heat flow towards the radiators of the LTS based in the sun is the sum of the absolutes of $Q_{hx_{3,4}}$ and $Q_{hx_{4,5}}$.

$$Q_{elec_{req}} = \frac{Q_{heat_{req}}}{COP} = \frac{2815}{5} = 563[W] \tag{13.10}$$

Furthermore, the heat pump actually is a heat exchanger combined with a compressor and a pressure valve. The already chosen heat exchangers are assumed as part of the heat pumps. Thus, the only additional weight and volume will be represented by the compressor that is selected for the LTS. For redundancy, two compressors are added. The weight of one compressor is 11.7 kg, and the volume of the compressor is 485 x 385 x 330 mm [11].

Pipelines & flow control

Next to the heat pumps, the other side of the heat exchangers still need the coolant to flow through the pipes. Also, the radiators need pumps to keep the ammonia flowing inside of them. Therefore, eight small pumps will be added to the active thermal system. They are divided over the radiators outside and the heat exchangers inside: two between the cabin and the fuel cell, two between the O_2/H_2 tanks and the fuel cell and one pump per radiator panel. One pump has a capability to pump 2.271 L s^{-1} and weighs 3.65 kg. The size of one pump is 200 x 200 x 100 mm.

Additionally, the active heat control system needs liquid tanks to store reserves for coolant, one for ammonia, one for water and one for the refrigerant of the heat pump. Also, pipes where the coolants can flow through need to be designed. The pipes need to resist the pressure of the ammonia if the temperature of the ammonia becomes higher. Figure 13.6 [12] shows the phase diagram for ammonia.

[9] https://nl.mouser.com/pdfDocs/ats-heat-exchange-data.pdf
[10] https://www.h2xengineering.com/blogs/heat-pump-cop-and-scop-what-they-mean-and-why-they-matter/
[11] https://airpress.nl/compressor-k-500-1000s-14-bar-7-5-pk-5-5-kw-600-l-min-500-l-36516-n
[12] https://demonstrations.wolfram.com/PressureTemperaturePhaseDiagramForWater/

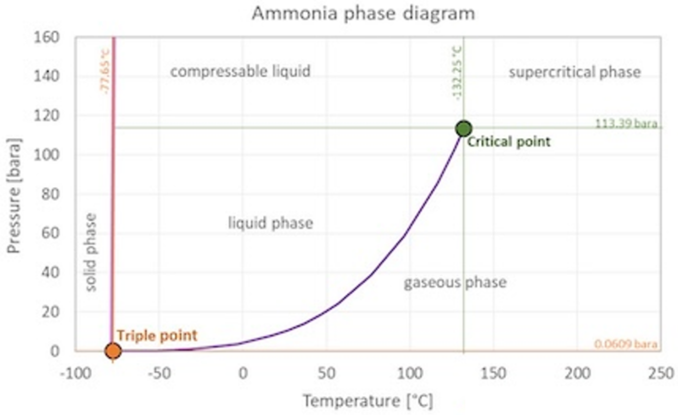


Figure 13.6: Ammonia phase diagram (source: footnote 12)

If the temperature of the ammonia becomes 120 °C, the pressure required to keep ammonia liquid is around 100 bar. The pipes need to be big enough to allow a reasonable amount of mass flow without getting too high flow speed. The following relation relates the mass flow of coolant and the temperature difference between fuel cell and radiators to the heat flow required.

$$\dot{Q} = \dot{m}c\Delta T \tag{13.11}$$

Where $\dot{Q}$ is the heatflow in W, $\dot{m}$ is the mass flow in kg/s, c is the specific heat and ΔT the temperature difference between two elements. Reshuffling and filling in the right values for ammonia gives:

$$\dot{m} = \frac{\dot{Q}}{c\Delta T} = \frac{(2515 + 300) \cdot 1.5}{2200 \cdot 20} \approx 0.1713\text{[kg/s]} \tag{13.12}$$

Note that a safety factor has been added to account for thermal loss and other inefficiencies. The mass flow of ammonia can be related to flow speed in the pipe using:

$$V_{flow} = \frac{\dot{m}}{\rho\pi r^2} = \frac{0.1713}{580 \cdot \pi \cdot 0.005^2} = 2.11\text{[m/s]} \tag{13.13}$$

where ρ is the density of ammonia in kg m^{-3} and r is the radius of the pipe's cross-section. The radius of 5 mm was taken as it is a standard measure. With an inner radius of 0.5 mm the flow speed of the ammonia is still reasonably low. Knowing the radius of the pipe, the hoop stress of a pipe can be calculated with [63]:

$$\sigma_{hoop} = \frac{Pr}{2t} \tag{13.14}$$

By rewriting the equation and selecting aluminium as the material for the pipe, the minimum required thickness can be determined. Instead of filling in a hoop stress, a maximum yield stress of aluminium was used ($\sigma_{hoop} \rightarrow \sigma_y$). The pressure of 100 bar is discussed as the limit for the ammonia coolant and is required to keep the ammonia liquid.

$$t = \frac{Pr}{2\sigma_y} = \frac{100 \cdot 10^5 \cdot 0.005}{2 \cdot 300 \cdot 10^6} = 0.083\text{[mm]} \tag{13.15}$$

Since this thickness is very small, the thickness of the aluminium pipes was increased to 2 mm. This is to ensure the pipes will be easier to handle and install, since pipes with very small thickness are very vulnerable. The total amount of pipes is calculated by summing up the distances between the most important elements:

$$2(L_{tanks,fuelcell} + L_{rad,fuelcell} + L_{cabin,fuelcell}) = 2(2 + (1.8\pi + 2) + 1) = 21.3\text{[m]} \tag{13.16}$$

The factor of two is present for redundancy, all pipes are installed twice. The designs of the coolant storage tanks will be discussed in the second-order detailed design phase after the 10-week mark. The storage tanks are sized for 5 liters of fluid and have a weight of 10 kg assigned to them. Furthermore, an insulation layer of 10 mm around the pipes is added to account for the heat loss. The insulation layer is made out of aerogel and the mass for this insulation is calculated to be 4.8 kg.

Radiators

The radiator is one of the most important elements of the active thermal control. The radiator panels of the LTS are made from aluminium and coated in graphite powder to increase their radiative properties. The radiators are based on the radiator design of the ISS [8]. The ISS does give a size and performance value for the radiators. However, the thickness of the radiators is never mentioned. Based on the weight and assuming the plates are made out of 50% aluminium and 50% air, a calculation can be made to estimate the thickness of the radiators.

$$t_{rad} = \frac{m_rad}{A_rad\rho} = \frac{1122.64}{1450 \cdot 23.3 \cdot 3.4} = 0.00977[\text{m}] \quad (13.17)$$

The final value assumed for the thickness of the panels is 0.01 m.

For redundancy, the panels have been simulated with one panel detached. The temperature of the panels becomes 348.30 K. This is still under the temperature limit for 100 bar of pressure, Figure 13.6. Note, the total area of the radiators with one panel detached is 3.64 m^2.

The position of the radiators is chosen to be perpendicular to the sun, which means they are required to rotate around their vertical axis. Furthermore, the position on top of the rover was selected since it has to most optimal performance, furthest away from the lunar surface and the abrasive dust and also always capable of being perpendicular to the LTS, the lunar surface and the sun.

Sensors & Valves

The active thermal control system needs input and feedback to control the temperature. This is where the temperature and flow speed sensors are required. It is assumed that all electronics have their own thermal sensors and will not be further specified. Extra thermal sensors are required per active thermal element. The overview of thermal sensors required is shown in Table 13.7. Note that the allocated number of sensors is preliminary.

Furthermore, the flow speed sensors are required to detect any malfunction in the plumbing of the coolant. The minimum sensors required are specified in Table 13.8. Note that also these numbers are preliminary. Also, the amount of valves was calculated using one valve before and after a heat exchanger of radiator. Also note that the pipes are all installed twice which means twice the amount of sensors.

Table 13.7: Thermal sensors overview

Element	Sensor Quantity
Heat exchangers	4
Radiator panels	8
Coolant pipes	20
Cabin	3
Structure	8
H_2/O_2 tanks	2
Total	45

Table 13.8: Valves overview

Element	Flow Speed Sensor Quantity
Coolant pipes	10
Radiators	8
Near every valve	20
Total	38

As mentioned before, to improve modularity of the system and control over the heat flows, several valves are placed to close off the pipes that are malfunctioning or need replacement. The total additional weight of the sensors and valves is estimated to be no more than 10 kg. Selection of the valves and sensors will be discussed in the second-order detailed design phase after the 10-week mark.

13.5. Redundancy, Modularity & Sustainability

As mentioned in Section 13.4, per element redundancy has been built into the system. The following measures were taken to ensure redundancy:

- Every coolant pipe is installed twice
- All pumps and heat exchangers have a backup
- Several sensors are installed to detect malfunctions
- Several valves are installed to handle coolant flows
- The radiators have been designed such that one panel can fail without endangering the mission or vehicle performance
- Coolant reserves are available in the event of a fluid loss

In terms of modularity, the thermal control subsystem will be designed in such a way that the valves have multiple functions. They can control the coolant flow, but will also be conveniently placed such that swapping parts is made possible. Also, the coolant reserves allow for coolant loss when swapping out parts.

The thermal control subsystem is designed to not expel any waste; this means the lunar environment stays clean and unchanged. Furthermore, when repairs need to be conducted, the thermal system can take advantage of its modular design to only replace a small part of the subsystem. Also, the radiator panels, as well as the heat exchangers, can be repurposed towards thermal control systems in lunar bases after the lifespan of the LTS has ended.

13.6. Thermal Subsystem Performance

To summarize the capabilities of the thermal control system, some critical characteristics can be mentioned. First of all, the thermal control subsystem can resist lunar days and nights. Initially the thermal system was designed to resists only lunar days. After analyzing the design, it was found that with good insulation the LTS could easily survive in the lunar shadow regions. The insulation was also beneficial for the power requirement of the heat pumps during the lunar days. Keeping the power consumption low for the LTS was crucial, thus the insulation was sized up.

Furthermore, the currently selected heat exchangers of the active thermal control can handle heat flows up to 7 kW. Also, the radiators can handle 4290 W of heat flow without exceeding a temperature of 100 °C which is important for the coolant inside the radiators. Finally, the thermal control subsystem is designed to be adjustable. This way, the crew can select a comfortable temperature within the range of 20-25 °C.

13.7. Final Thermal Subsystem Budgets

To create an overview of the full thermal control subsystem, a budgeting table has been constructed. Table 13.9 lists all parts with their mass, volume and power consumption. The cost on a part level is not documented due to time constraints. The mass and volume are per one part, thus some things will need to be multiplied with the number of parts to find, for example, total mass. The power is documented as the total average power required in operational conditions.

Table 13.9: Thermal control system budget

Part	Quantity parts	Mass [kg]	Volume [mm x mm x mm]	Electrical Power Required [W]
Heat pump compressors	2	11.7	485 x 385 x 330	563
pumps	8	3.65	200 x 200 x 100	10
Heat exchanger liquid-liquid	2	41.28	635 x 530 x 203.2	N.A.
Heat exchanger liquid-air	2	6.569	266.7 x 147.7 x 10.2	50
Radiator	4	7.37	600 x 800 x 10	N.A.
Coolant pipes	1	4.658	5 x 21300	N.A.
Sensors & Valves	$\geq$ 50	0.2	5 x 10 x 10	2
Storage tanks	3	10	100 x 160	N.A.
Cabin insulation	1	984	50 x 65.6E6	N.A.
Fuel cell insulation	1	1.1	50 x 7.25E4	N.A.
Fuel cell shell	1	1.5	5 x 1.0E5	N.A.
Fuel cell shell insulation	1	0.9	30 x 9.0E4	N.A.
H_2/O_2 tanks insulation	1	63	30 x 7E6	N.A.
Motor regolith insulation	6	2.6	5 x 2.86E5	N.A.
Pipe insulation	1	4.8	10 x 6.69E5	N.A.
Total		1293	3.70E9	625

The number of sensors and valves are summed up in the table, since depending on the final configuration, the number of sensors and valves changes. The number 50 comes from an estimate of 20 temperature sensors, 20 flow speed sensors and at least 10 valves. Furthermore, the coolant pipes measurements and the storage tanks measurements are given in [radius x length], and the insulation is all given in [tickness x area]. All measurement units are in mm. Note that the thickness and size of the thermal insualtion for the cabin are very high. Further research should be done into lowering the size and mass of the insulation. Materials such as multilayer insulation (MLI) are commonly used to improve the insulation against radiative heatflow. This could be applied to the LTS design as well.

13.8. Compliance Matrix

To verify the performance of the thermal control subsystem, a compliance matrix is used. Table 13.10 lists all thermal requirements and also evaluates if the design adheres to the requirements.

Table 13.10: Compliance matrix of the thermal control subsystem

Requirement ID [REQ-SSYS-TC-#.#.#.#]	**Description**	**Method**	**Compli-ance**	**Justification**
REQ-SSYS-TH-2.2.	The thermal subsystem shall ensure the air temperature inside the pressure vessel is within the range of 293-298 [K]	Analysis, Testing	Yes	Testing will be conducted in a later stage
REQ-SSYS-TH-2.2.	The thermal subsystem shall ensure the temperature of components that the astronauts are exposed to are within the range of 293-298 [K]	Analysis, Testing	Yes	Testing will be conducted in a later stage
REQ-SSYS-TH-2.3.	The thermal subsystem shall have a backup system	Analysis/Inspection	Yes/TBD	
REQ-SSYS-TH-5.1.	The thermal subsystem shall ensure the fuel cell of the LTS is within the range of 243-373 [K]	Analysis, Testing	Yes/TBD	Testing will be conducted in a later stage
REQ-SSYS-TH-5.1.	The thermal subsystem shall ensure the battery of the LTS is within the range of 218-398 [K]	Analysis, Testing	Yes/TBD	Testing will be conducted in a later stage
REQ-SSYS-TH-5.1.	The thermal subsystem shall ensure the electric components of the LTS are within the range of 218-398 [K]	Analysis, Testing	Yes/TBD	Testing will be conducted in a later stage
REQ-SSYS-TH-5.1.	The thermal subsystem shall ensure the electric motors of the LTS are within the range of 23-673 [K]	Analysis, Testing	Yes/TBD	Testing will be conducted in a later stage
REQ-SSYS-TH-5.1.	The thermal subsystem shall ensure the outside structure is within the range of 20-673 [K]	Analysis, Testing	Yes/TBD	Testing will be conducted in a later stage
REQ-SSYS-TH-5.1.	The thermal subsystem shall ensure the O2-tank is within the range of 233-358 [K]	Analysis, Testing	Yes/TBD	Testing will be conducted in a later stage
REQ-SSYS-TH-5.1.	The thermal subsystem shall ensure the H2-thank of the LTS is within the range of 233-358 [K]	Analysis, Testing	Yes/TBD	Testing will be conducted in a later stage
REQ-SSYS-TH-5.1.	The thermal subsystem shall ensure the antenna of the LTS is within the range of 153-393 [K]	Analysis, Testing	Yes/TBD	Testing will be conducted in a later stage
REQ-SSYS-TH-5.2.	The thermal subsystem shall employ a modular design	Analysis, Inspection	Yes/TBD	Inspection will be conducted when the first prototype is constructed
REQ-SSYS-TH-5.5.	The thermal subsystem shall be recyclable to the extent specified in the End Of Life strategy	Analysis	Yes	
REQ-SSYS-TH-5.7.	The development costs of the thermal subsystem shall not exceed [TBD] Euros	Analysis	TBD	Due to time constraints, no detailed cost requirement were made.
REQ-SSYS-TH-5.7.	The manufacturing cost of the thermal subsystem shall not exceed [TBD] Euros	Analysis	TBD	-
REQ-SSYS-TH-5.7.	The maintenance cost of the thermal subsystem shall not exceed [TBD] Euros	Analysis	TBD	-

14 | Electrical Power Subsystem

Author: Ishaan

In this chapter, the electrical power subsystem is sized. The means of power generation is decided through a trade-off, then the main power generation system and ancillary systems are designed. Finally, a failure mode analysis and verification and validation of the design tool and subsystem as a whole is conducted.

14.1. Overview

The electrical power subsystem (EPS) will use a fuel cell system for nominal power deployment, with an additional battery that will be used during peak power needs. The fuel cell system will use hydrogen and oxygen as its fuel; four hydrogen tanks and four oxygen tanks in the frame of the LTS will be used to store the fuel. Additionally, two storage tanks placed in the walls of the cabin will be used to store the water produced by the fuel cells. The tanks will be made using T1000 carbon fiber composites. A feed system has been designed to ensure safe distribution of reactants from the tank to the fuel cells and a power management system has been designed to ensure safe distribution of power from the fuel cells to components. The total mass of the EPS equals 430 kg.

14.2. Power Generation Trade-off

For the means of power generation, three options were considered: power generated by solar arrays, energy stored batteries which is then used to deliver power to all components and power generated by fuel cells. Nuclear sources of power were not considered, because a lot of additional design considerations would be needed to make the LTS safe for human use, which would outweigh the benefits of nuclear power. There are many design options within each choice, but for the purpose of evaluation of the different options, Lithium-ion batteries, Polymer-electrolyte membrane (PEM) fuel cells using gaseous hydrogen and oxygen, and solar arrays using TJ-GaAs cells were considered. Specifications for the solar array and batteries were taken from [66], and specifications for the fuel cells were taken from the U.S Department of Energy [1].

For the purpose of making a decision on the type of power generation system, the exact power needed was not used because a comparison between the choices can still be made. It was assumed that each option should generate 6000 W, to be used over two days. The following table shows the capabilities of each option.

Table 14.1: Comparison of different power generation options

	Solar arrays	Batteries	Fuel cells
Power required to be generated $[W]$	6000	6000	6000
Energy required to be generated over a two day mission $[Wh]$	288000	288000	288000
Specific power $[W/kg]$	115	-	2000
Power density $[W/m^2]$	276	-	-
Volumetric power $[W/dm^3]$	-	-	3100
Specific energy $[Wh/kg]$	-	133	16666.67
Volumetric energy $[Wh/dm^3]$	-	321	-
Mass $[kg]$	52	2165	343
Volume $[m^3]$	-	0.897	0.863
Area $[m^2]$	21.7		
Operational temperature $[℃]$	-70 to 30	-20 to 30	-30 to 100

Table 14.1 shows that having only batteries to store the required energy for a two-day mission is unfeasible. The weight required for all the batteries would result in the total weight of the vehicle exceeding the design weight of 5 tonnes and resulting in a snowball effect which would result in increasing power requirements as the drive-train subsystem would need more

[1] https://www.energy.gov/eere/fuelcells/hydrogen-and-fuel-cell-technologies-office

power to move the heavier LTS, again increasing the weight and so on. Note that the design weight varies from the actual final weight of the LTS. Having only solar arrays producing the required power was also deemed unfeasible because a complicated array system would be needed with a relatively large surface area of solar arrays. Furthermore, a constant power deployment cannot be guaranteed with solar arrays, as power could only be produced during the day, which also varies relative to the inclination of the Sun. The Sun at the South Pole is at a relatively low inclination casting long shadows, meaning continuous generation of high amounts of power would be difficult. A system that constantly adjusts the solar arrays to make sure they face the Sun would be needed, which is an added layer of complexity.

For the fuel cells, PEM fuel cells were used as they are most commonly used for vehicle applications on Earth. The specifications for the fuel cells were taken from the fuel cell stack used on the hydrogen-powered 2014 Toyota Mirai. To get a first estimate of the required amount of reactants, a conservative efficiency of 50% was assumed [2]. Hydrogen has specific energy of 120 MJ/kg [3]. With the efficiency, it was assumed that 60 MJ of energy can be produced per kg of hydrogen. Along with hydrogen, oxygen is also required. Per kg of hydrogen, 8 kg of oxygen are needed, and 9 kg of water are produced. This water will be collected on the LTS for later electrolysis on the Lunar Basecamp, thus allowing for continuous supply of fuel for the LTS. Using this method, a total mass for the fuel cell system excluding storage of reactants and ancillary systems of 343 kg was determined.

The main problem with fuel cells is that hydrogen has a low volumetric energy, which means that it needs to be stored at high pressures to allow for manageable storage volumes. To get a first estimate of the volume of gas needed for each reactant, it was assumed they will be stored on Earth and brought up to the Moon, meaning the temperature of the gas inside the tank will be at the ambient temperature of 20 $℃$. The hydrogen was kept at 700 bar and the oxygen at 300 bar, and all the volume of water produced will also be stored on the LTS, resulting in a total volume of 0.863 m^3.

Based on these first estimates, fuel cells are looking like the most viable option for power generation. The mass estimates don't include the mass of the tanks, but the added weight is likely not going to make the entire system as heavy as the battery-powered alternative. A combination of fuel cells and solar arrays is also possible. In order to judge the feasibility of this option, it was assumed that 1000 W was produced by the solar arrays and the remaining 5000 W was produced by the fuel cells. Doing so required an array size of about $4m^2$, which is manageable and reduced the mass of the fuel cells and reactants by 53 kg, but the mass of the solar array itself would be 9 kg. As previously explained, in order to get continuous power out of the solar arrays, a system that ensures it is always pointing at the Sun is needed, and it was deemed the added complexity of such a system outweighs the 40 kg weight reduction, therefore solar arrays will not be used at all for the LTS.

As a result, it was decided that fuel cells will be used as the main source of power generation on the LTS. It will be designed for average loads during nominal operations, while a battery will be used for rapid deployment of power during peak loads.

[2] https://www.energy.gov/eere/fuelcells/articles/fuel-cells-fact-sheet

[3] https://www.energy.gov/eere/fuelcells/hydrogen-storage

14.3. Fuel Cell System

Now that it has been decided that the LTS will use fuel cells for nominal power usage, the specifics of the design can be explored. There are three main things that need to be considered, the sizing of the fuel cell stack, the sizing of the reactant storage tank and the feed system between the tanks and the fuel cells.

Power requirement

To properly size the fuel cell system, the peak nominal power required by each subsystem is needed. The peak nominal power is defined as the most power that a system or component will continuously consume during the two day mission. This is shown in Table 14.2.

Table 14.2: Peak nominal power requirements of each subsystem

Subsystem/component	Peak nominal power requirement [W]
Propulsion and suspension	2000
Life support	1088
Thermal	570
Cargo handling	368
Cargo storage	200
Telecommunication+ GNC + User Interface + Internal communications	586
Airlock	1000

Along with that, different subsystems are active during different operational modes, therefore, the required power is different. These different modes are shown in Table 14.3.

Table 14.3: Different operational modes and their power usage

Operational Mode	Subsystems required	Power usage (W)
Ferrying-empty	Telecommunications, Thermal, GNC, Propulsion + Suspension, Internal Communications	3156
Ferrying-cargo	Telecommunications, Thermal, GNC, Propulsion + Suspension, Cargo storage, Internal Communications	3356
Ferrying-cargo + crew	Telecommunications, Thermal, GNC, Propulsion+Suspension, Cargo storage, Life Support, Internal Communications	4444
Ferrying-crew	Telecommunications, Thermal, GNC, Propulsion + Suspension, Life Support, Internal Communications	4244
Idle	Telecommunications, Thermal, GNC, Internal Communications	1156
Idle+humans	Telecommunications, Thermal, GNC, Life Support, Internal Communications	2244
Idle+cargo handling+humans	Telecommunications, Thermal, GNC, Life Support, Cargo Handling, Cargo Storage, Internal Communications	2812
Idle + cargo handling + airlock + humans	Telecommunications, Thermal, GNC, Life Support, Cargo Handling, Cargo Storage, Airlock, Internal Communications	3812
Ferrying + airlock + humans + cargo storage	Telecommunications, Thermal, GNC, Life Support, Cargo Storage, Airlock, Propulsion+Suspension, Internal Communications	5444

It was decided that the airlock will not be used while the LTS is moving, therefore, the operational mode that requires the most power is ferrying cargo and crew, and therefore, the fuel cells will be sized to be able to continuously provide 4888 W for two days, accounting for a 10% margin.

Fuel Cell Stack

The fuel cell stack to be used on the LTS is based on the fuel stack used by the 2014 Toyota Mirai. The Mirai uses a fuel stack with 370 PEM cells connected in series producing 114 kW[4] with an output voltage of 360 V[5]. The total weight and volume of the fuel cell stack included fasteners is 56 kg and 37 dm^3 respectively.

Since these cells are connected in series, the current through each cell is the same as the output current but the voltage per cell is different. The total power output of the fuel cell stack can be expressed as [67]:

$$P_{stack} = I_{stack} \cdot V_{stack} = n_{cells} \cdot V_{cell} \cdot I_{stack} \tag{14.1}$$

The voltage of each cell is then found through:

$$V_{percell} = V_{stack}/n_{cells} \tag{14.2}$$

This yielded a voltage and power per cell of 0.937 V and 308 W, respectively, operating at a current of 316.67 A. With the required power of 4888 W, a total of 16 fuel cells are needed, meaning the fuel stack used on the LTS will occupy a volume of about 1.6 dm^3 and weigh 2.4 kg. Along with electrical power, fuel cell stacks generate heat, given by [67]:

$$P_{heat} = P_{stack}\left(\frac{1.48}{V_{cell}} - 1\right) \tag{14.3}$$

Using Equation 14.3, 2547 W of heat is produced by the fuel cell stack, which can be used by the thermal subsystem for temperature control of the LTS.

Reactants Sizing

The fuel cell stack uses hydrogen and oxygen as its source of energy. The required mass flow of hydrogen to the fuel cell stack is given by [68]:

$$\dot{m}_{H_2} = 1.05 \cdot 10^{-8} \cdot \frac{P_{stack}}{V_{cell}} \tag{14.4}$$

This yielded a required mass flow to produce 4888 W of $5.3 \cdot 10^{-5}$ kg/s. Along with this, extra hydrogen is needed to ensure the pressure of the storage tank is equal to the required pressure of the fuel cells, however this was assumed to be negligible relative to the total gas mass needed. Finally, a reserve amount of hydrogen will be included that will only be used in case of emergency and will allow for an additional five hours of nominal operations. Thus, the total mass of required hydrogen is:

$$m_{total} = \dot{m}_{H_2} * (t_{two-days} + t_{five-hours}) \tag{14.5}$$

Using Equation 14.5, the required mass of hydrogen comes to be 10.06 kg. The required mass of oxygen can be determined from the chemical reaction taking place in the fuel cells:

$$H_{2_{(g)}} + \frac{1}{2}O_{2_{(g)}} = H_2O_{(l)} \tag{14.6}$$

This shows that for every mol of hydrogen gas, 0.5 mol of oxygen gas is needed, meaning that every kg of hydrogen gas needs 8 kg of oxygen gas. Along with the required mass of oxygen for the fuel cells, the oxygen tanks will also hold oxygen required by the life support subsystem, which is 15.3 kg. The total mass of oxygen then becomes 95.8 kg. A total of 90.6 kg of water will be produced by the fuel cell once the entire mass of hydrogen is used.

[4] https://www.smartcirculair.com/wp-content/uploads/2020/06/Toyota-Mirai.pdf

[5] https://clustercollaboration.eu/sites/default/files/2021-03/Transport%20Unit%2004%20High%20voltage%20components-EN.pdf

Storage Tank Sizing

Now that the mass of reactants has been determined, the storage tanks can be sized. These fuels will be stored as gases at high pressures to allow for a large mass of gas to be stored in a relatively small volume. The hydrogen will be stored at 700 bar, as this is standard for hydrogen powered vehicles on Earth [6]. The phase diagram of oxygen, shown in **??** was used to determine the pressure to store it at [7].

The higher the pressure of the gas, the smaller the volume it occupies, but the thicker the walls of the storage tank need to be to hold the gas, which leads to increased mass. Since the molecular weight of oxygen is considerably higher than hydrogen, it does not need to be stored at as high pressures to allow for reasonable volumes of gas. Therefore, a pressure of 300 bar was chosen to store oxygen. At this pressure, oxygen will stored as a supercritical fluid. Furthermore, the gases will be treated as ideal gases, which allows for the use of the ideal gas equations.

The tanks are assumed to be cylindrical tanks with hemispherical ends on either sides. This shape was chosen as it allows for long, thin tanks, which can be stored in the lower section of the frame. The inner volume of the tank will equal the volume of gas needed to be stored, expressed through the equation:

$$\pi r^3 \left(4/3r + l\right) = \frac{nRT}{p} \tag{14.7}$$

Where r is the inner radius, l is the length of the cylindrical section of the tank, n is the number of moles of gas in the tank, R is the universal gas constant, T is the temperature of the gas and p, the nominal pressure of the gas. The length of the cylindrical section depends on the allowable space in the frame of the LTS. The length was constrained to 2 m by the structures subsystem. Once the inner volume is determined, the thickness of walls of the tank can be found. For the cylindrical section, it experiences stresses in the hoop and longitudinal direction, while the hemispherical ends experience a constant stress in all directions. The stress in the longitudinal direction of the cylinder is the same as the stress in the end-caps, while the stress in the hoop direction is double that in the longitudinal direction[69]. Having a pressure vessel of constant thickness would make the manufacturing easier and therefore, the storage tank was sized for a constant thickness of:

$$t = \frac{Pr}{\sigma} \tag{14.8}$$

Here, P is the maximum pressure of the gas, r is the inner radius of the tank and σ is the allowable stress that the material of the tank can handle. Once the thickness has been calculated, the entire mass of the tank can be calculated with:

$$M = \rho\left(\left(\pi(r+t)^2 - \pi r^2\right) \cdot l + \frac{4}{3}\pi((r+t)^3 - r^3)\right) \tag{14.9}$$

Here, ρ is the density of the material used.

Values Used for Tank Sizing

The inner volume of the tanks will be sized for nominal pressures and temperature but the thickness of the walls will be sized for the maximum pressure that the tank will experience. The maximum pressure can be determined by the maximum permissible temperature of the gas

[6] https://www.energy.gov/eere/fuelcells/articles/hydrogen-storage-fact-sheet
[7] https://www.chemix-chemistry-software.com/school/phase-diagram/oxygen-phase-diagram.html

inside the tank. It is assumed that the storage tanks for these gases will first be made on Earth and transported to the Moon. This means that during handling on Earth, the temperature of the gas inside the tank would equal the ambient temperature of where it is being stored. Therefore, sizing of the inner volume of the tank will assume the gases are kept at 20 degrees Celsius and at 700 bar for hydrogen and 300 bar for oxygen. For the sizing of the wall thickness, it is assumed that the volume of gas in the inner tank is subjected to the maximum allowable temperature for hydrogen tanks on Earth of 85 degrees, thereby increasing the pressure of the gas inside. The values used for sizing of the tanks are summarised in Table 14.4. The pressure to store the water was chosen as 11.38 bar, which comes from a NASA paper describing the designed of a fuel cell reactant storage system [70].

Table 14.4: Table showing values used to size storage tanks

	Nominal tank temperature [$^\circ C$]	**Nominal pressure [bar]**	**Maximum pressure [bar]**
Hydrogen	20	700	855
Oxygen	20	300	367
Water	20	11.38	11.38

Material Choice

The mass of the storage tanks are heavily influenced by the choice of material. In order to minimize mass, a material with a high allowable stress and low density needs to be chosen. One such material that satisfies these requirements is the T1000 carbon fibre composite, which has a tensile strength of 3040 MPa and a density of 1.56 g/cm^3. This material has previously been used to produce similar storage tanks known as composite over-wrapped pressure vessels (COPVs) for the storage of high pressure gases like hydrogen [71]. Due to this reason, this material was chosen for the storage tanks on the LTS. An ultimate safety factor of two was used when designing the storage tanks as is standard for composite structures according to [20]. This safety factor essentially means that the allowable stress of the storage tanks becomes 1520 MPa, and hence this is σ used in Equation 14.8.

Number of tanks

Another important design consideration when sizing the tanks is number of tanks used to store the required mass of reactants. Figure 14.1 shows the effect on the total mass of all the hydrogen tanks as the number of tanks increase. As can be seen, increasing the number of tanks decreases the total mass of all tanks, although the decrease is marginal.

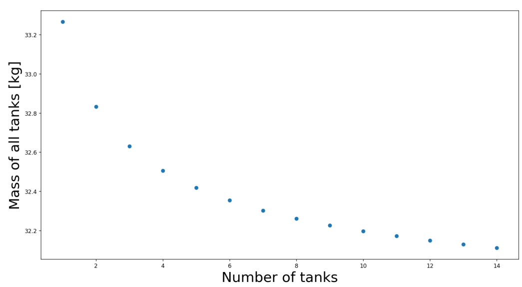


Figure 14.1: Graph show the effect of increasing number of hydrogen tanks on total mass of all tanks

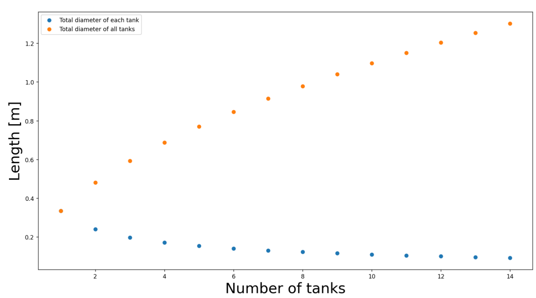


Figure 14.2: Graph show the effect of increasing number of hydrogen tanks on cross-sectional dimensions

Figure 14.2 shows how the total diameter of each hydrogen tank and the total diameter of all the tanks change with increasing number of tanks. The total diameter of each tank decreases by increasing the number of tanks, which was to be expected as each tank is holding less gas. The total diameter of each tank decreases slightly with increasing number of tanks, but the total diameter of all the tanks explodes. Therefore, this shows that increasing the number of tanks does not produce significant benefits. However, along with a length constraint, the structures subsystem also imposed a diameter constraint, where the maximum diameter of each tank cannot exceed 20 cm. This means that only having one tank is not feasible. Furthermore, by having more than one tank, redundancies are kept in place, ensuring that failure of one tank would not cripple the LTS. Therefore, the number of tanks was chosen such that the maximum diameter is below 20 cm, and an even number of tanks will be chosen to ensure symmetry is maintained for the location of the center of gravity. The above method was applied to both the hydrogen tanks and oxygen tanks and as a result, it was chosen that four hydrogen and four oxygen tanks would be used for the LTS.

The sizing of the water tanks was done slightly differently to the sizing of the reactant tanks. It was decided by the structures subsystem that the water tanks would be stored in the wall of the cabin, instead of the frame. Therefore, a radius constraint of 0.12 m was imposed and the length and thickness were sized accordingly, resulting in the choice of two water tanks being used on the LTS to store water produced by the fuel cell.

Results of Tank Sizing

Table 14.5: Table showing results of tank sizing

	Total diameter of each tank [m]	**Wall thickness [mm]**	**Total length of each tank (including end caps) [m]**	**Total mass of all tanks + fluid [kg]**
Hydrogen (4 tanks)	0.171	4.58	2.09	42.6
Oxygen (4 tanks)	0.195	2.30	2.10	115
Water (2 tanks)	0.240	0.011	1.07	90.6

Table 14.5 shows the dimensions and total mass of all the tanks for each fluid being stored. The values were determined using the methodology described in the previous sections. For the water tank, the same material was used for the reactants tank, yielding a theoretical thickness of 0.011 mm, and the masses were calculated using this value. However, this would make manufacturing extremely difficult, therefore, a thickness of 0.5 mm [8] should be used for the water tanks and a different material can also be used, since the material does not need to be very strong to hold in the water. In the next iteration of the design, the masses can be recalculated to account for this change.

Feed System

A feed system is required between the tanks and the fuel cell because the reactants will be stored at a given pressure, but the fuel cells have an optimal operational temperature and pressure. For PEM fuel cells, the optimal pressure is between 3 and 4 bar [72], and the optimal temperature is between 60 and 90 ℃ [73]. Temperature control of the fuel cell will be handled by the thermal management subsystem and will be maintained at about 80 ℃ but a feed system is required to ensure the fuel cells are kept within the nominal operating pressure.

[8] https://www.carbon-composite.com/en/About-carbon/Carbon-basics/

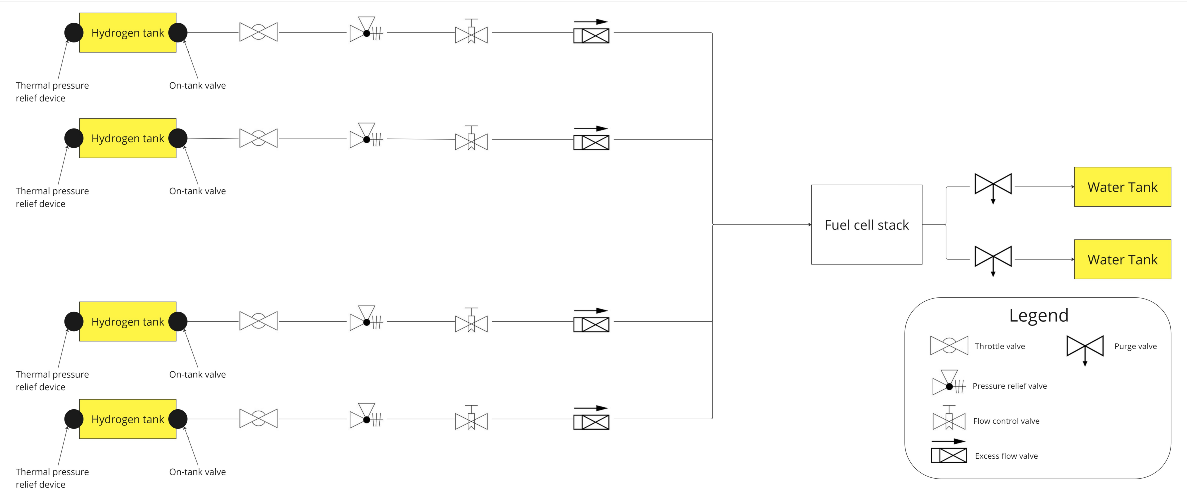


Figure 14.3: Feed system used by hydrogen tank sets

The same feed system shown in Figure 14.3 will also be used for the four oxygen tanks. Each tank will have an on-tank valve, which allows for safe refuelling and defuelling of the tanks, while also containing a temperature sensor and pressure sensor. If the temperature inside the tank ever reaches unsafe levels, the thermal pressure relief valve will empty the tanks before a fire can start [9]. In accordance with this, throttle valves will be used to reduce the pressure of the reactants. Throttle valves allow the gas to be pushed through a small hole, with no heat and work exchanged with the environment, meaning the pressure of the gas is reduced under isenthalpic conditions. Since ideal gases were assumed, this means that there is no temperature change after the gas pressure is reduced [10].

For the hydrogen feed system, two trottle valves will be used for each tank; one high-pressure valve that brings the pressure down from 700 bar to 100 bar will be used, and a medium-pressure valve that brings the pressure from 100 bar to 4 bar. For the oxygen system, only the medium-pressure valve will be used that brings the pressure down from 300 bar to 4 bar. Following the trottle valves, a pressure relief valve will be in place in case the pressure following the throttle valves is higher than 4 bar.

After the pressure relief valve, the gas will flow to a flow-control valve which ensure that the correct mass-flow of reactants is entering the fuel cell depending on the current power requirement. Similar to the pressure-relief valve, an excess flow valve will be in place to ensure the mass flow going into the fuel stack is not higher than it needs to be. Each set of hydrogen and oxygen tanks will have their own feed system as these operate independently from each other. Finally, after the fuel cell stack, a purge valve will be used to drain the fuel cell of the water produced and direct the water to the water tanks for storage. Existing market components were used to estimate the mass of the feed system, shown in Table 14.6. The total mass of the feed system comes out to 81.42 kg. For the mass of piping, an upper limit of 15 kg is kept, which would mean about 90 meters of piping is used.

[9] https://teesing.com/en/library/hydrogen/on-tank-valve-for-700-bar-hydrogen-tanks
[10] https://home.iitk.ac.in/ gtm/thermodynamics/lecture14/14-2.htm

Table 14.6: Mass budget for feed system

Component	Number	Mass (kg)	Total mass (kg)
TOPAQ On Tank Valve OTV 2.0	8	2.50	20.0
Argo-Anleg GmbH Thermal pressure relief device	8	0.920	7.36
Pressure Tech H875 Hydrogen Pressure Regulator (High pressure regulator)	4	1.50	6.00
Hydrogen Fuel Cell Pressure Regulator - LW438 (Medium pressure regulator)	8	0.425	3.40
VRH 30 Pressure relief valve	8	1.50	120
Flow control valve Type G PC P 040	8	0.420	3.36
ER25 Excess flow valve	8	1.75	14.0
Purge / Drain valve Type G PC S 028	2	0.150	0.300
TOPAQ H2 Pipes for Hydrogen Distribution			15.0

Battery

Along with the fuel cells, a battery will be used for peak loads. For the powertrain and mobility subsystem, a peak load of 7000 W for 10 seconds is needed. In addition to those 7000 W, the suspension system has a peak load of 2000 W to each wheel, and up to three wheels can be needed during peak power deployment. This power however will only be needed instantaneously (over one second). To meet these requirements, a 5.38 kWh battery produced by PowerTech Systems will be used [11]. This battery has a peak power deployment of 10.2 kW, which can be deployed over 30 seconds, a continuous power deployment of 6.12 kW and an instantaneous power deployment of 25.6 kW. The weight of this battery is 37.5 kg. If the battery is continuously discharged, it can provide 6.12 kW for 52 minutes. Alternatively, if the entire battery were to be used to provide power only to the powertrain and mobility subsystem, 277 deployments would be possible, and if only to the suspension system, assuming all three wheels needed the power, 3228 deployments would be possible. Furthermore, a system will be in place that can recharge the battery using the power from the fuel cells if needed, allowing for continuous usage of the battery.

14.4. Refueling

For the first LTS, the hydrogen and oxygen for the fuel would be brought to the Moon from the Earth. Afterwards, it was decided that the water onboard the the LTS will be electrolysed on the Lunar Basecamp to allow for a continuous supply of gaseous hydrogen and oxygen. Electrolysers work by using a high voltage and current to split water into its gaseous constituents. To do so, a large amount of power would be required, which is part of the reason it was decided not to include such a system on the LTS. The infrastructure needed to realise such a plan would involve a large industrial electrolyser, such as those used for the production of hydrogen and oxygen on Earth. This would be powered by the Nuclear plant NASA is planning on constructing for the Lunar Basecamp. Along with the electrolyser itself, safe storage of the gases will need to be ensured through pressurised tanks, although since volume is not that big of an issue on the Basecamp, this doesn't necessarily need to be kept at as high pressures as on the LTS. However, the refuelling stations at which the LTS will dock will need to have compressors that can compress the gas to the required pressures, along with cooling systems as gases experience increases in temperature when getting compressed.

[11] https://www.powertechsystems.eu/home/products/48v-lithium-ion-battery-pack/48v-105ah-5-38kwh-lithium-ion-battery-pack-powerbrick/

Furthermore, the refuelling stations will also need to have a system in place that empties the water tanks onboard the LTS, and collects them for electrolysis, and another system in place that correctly and safely controls the fuelling of gases, and stores them in their respective tanks onboard the LTS. Such refuelling stations would be able to empty and refuel the LTS in around 15 to 20 minutes, considering it takes about 3-5 minutes to refuel hydrogen cars on Earth, which only have one tank that needs to be filled [12].

14.5. Power Management System

Figure 14.4 is the electrical block diagram, showing the power management system for the LTS. Bus modularity was used whereby, each component of a subsystem would be connected to its SBC and a bus bar. This allows for easy adaptation if new components are needed for a subsystem.

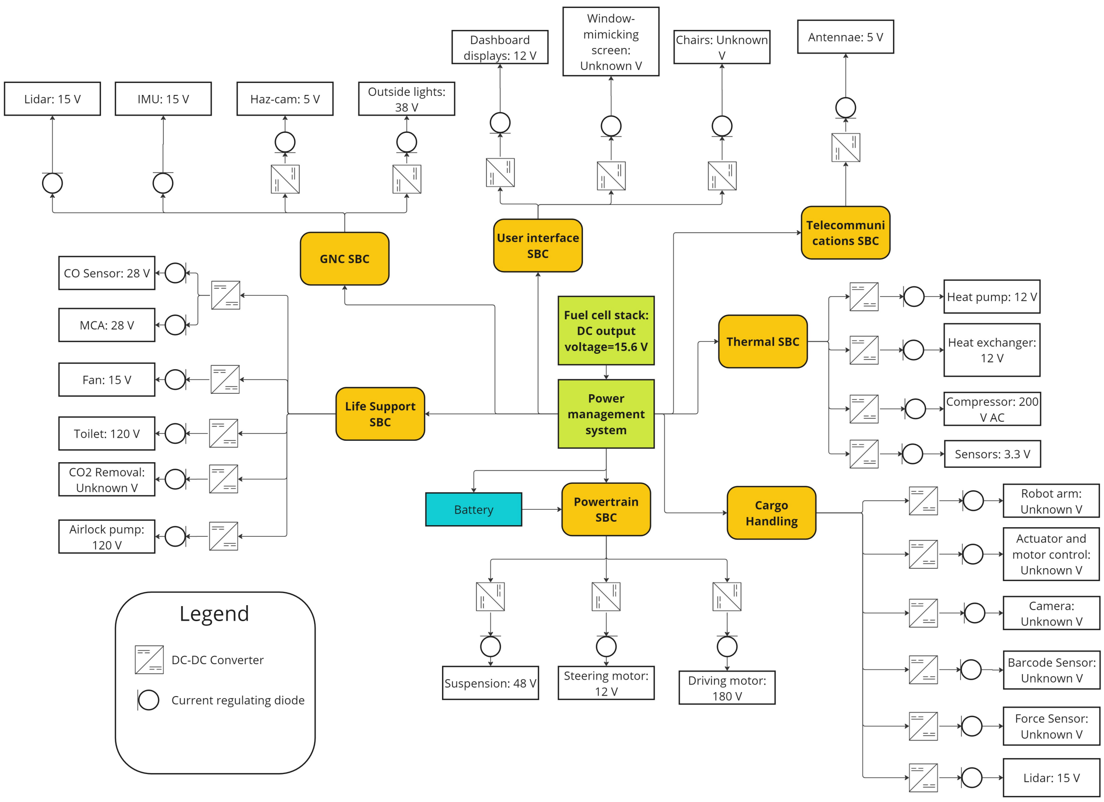


Figure 14.4: Electrical block diagram showing power management system for the LTS

The power generated by the fuel cells will have an output voltage and current of 15.6 V and 317 A, respectively. The general architecture is that the power from the fuel cells will go to the power management system, which would ensure a safe and constant output voltage and current from the fuel cells. From that, the power will be sent to the SBC's of the respective subsystems, which will then send the power to the required components. All components run on direct currents. Components that require a direct current, at a voltage different from the output voltage, have a DC-DC converter after the SBC, which ensures the components are receiving the correct voltage. After the converters, each

[12]https://driveclean.ca.gov/hydrogen-fueling#: :text=Fueling %20is%20Easy%20and%20Fast,from%20other%20electric%20car%20types.

component also has a current regulating diode, which ensures the correct current is entering the element. Certain elements' voltage are still not known, but they would also employ the same system as described. Do note that this block diagram does not show the number of components, but the main framework for each component will be the same as shown. Furthermore, there will be multiple SBCs assigned to each subsystem as redundancies.

As an estimate for the mass of wiring needed on the LTS, the average mass of wiring in road cars is taken, which is said to be about 60 kg [13].

14.6. Final Budget of Subsystem

The final budget of the EPS is shown in Table 14.7.

Table 14.7: Mass budget of subsystem

Component	Mass (kg)	Volume (m^3)
Tanks + reactants + water	248.1	0.542
Feed system	81.4	-
Power management system	60	-
Battery	37.5	0.0259
Fuel cell stack	2.4	0.00156
Total	430	0.569

14.7. Failure Modes

As part of a reliability analysis, a failure mode analysis was conducted for each element of the EPS. This can be seen in Table 14.8. As can be seen, single failures of all components is not critical and will not result in mission failure, except the failure of the fuel cell stack. This is to be expected as there are redundancies in place for each component except for the fuel cell stack. More research needs to be done into the reliability and integration of the fuel cell stack.

Table 14.8: Failure modes of the internal communications subsystem.

Component	Failure Mode	Effect	Prevention/Response	Criticality
Hydrogen tanks	Single Failure	Less reactant onboard to be used for power generation. Range and duration is impacted.	Singular tank is emptied of contents. Multiple tanks on board for redundancies. Single failure of tank not critical.	2
	Complete Failure	LTS no longer can produce power.	All tanks emptied of contents. Failure of all tanks is critical. Mission will need to be aborted.	4
Oxygen tanks	Single Failure	Less reactant onboard to be used for power generation. Range and duration is impacted.	Singular tank is emptied of contents. Multiple tanks on board for redundancies. Single failure of tank not critical.	2
	Complete Failure	LTS no longer can produce power.	All tanks emptied of contents. Failure of all tanks is critical. Mission will need to be aborted.	4
Water tanks	Single Failure	All water produced from fuel cell can no longer be stored if all reactants is used up. Range and duration is impacted.	Other tank that has not failed will be the only tank being used to collect water.	2

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[13] https://blogs.sw.siemens.com/ee-systems/2020/07/28/
wiring-harness-development-in-todays-automotive-world/#: :text=
Some%20modern%20vehicles%20contain%20close,weigh%20approximately%20132lbs%20(60kg).

Table 14.8 – *Continued from previous page*

Component	Failure Mode	Effect	Prevention/Response	Criticality
	Complete Failure	Water produced from fuel cell can no longer be stored. Fuel cells would stop functioning properly, power generation impacted.	Failure of all tanks is critical. Mission will need to be aborted.	4
On-tank valves	Single Failure	Singular tank unable to fuel/defuel. Range and duration impacted.	Each tank has their own on-tank valve for redundancies. If one on-tank valve fails, that tank will no longer be used.	2
	Complete Failure	All tanks unable to fuel/defuel. Power generation impacted.	All on-tank valves failing is a critical failure and mission will need to be aborted.	4
Thermal pressure relief valves	Single Failure	No prevention against fire for singular tank in case of tank failure	Each tank has their own thermal relief pressure valve for redundancies. If one valve fails, that tank will no longer be used.	2
	Complete Failure	No prevention against fire for all tanks in case of tank failures	Failure of all valves will not prevent operational use of fuel cells. However, no safety measures will be in place to prevent fires. Failure is not critical, but could be problematic.	2
Throttle valves	Single Failure	No pressure regulation for single tank from tank to fuel cells. Range and duration impacted.	Each tank has its own set of throttle valves for redundancies. In case a throttle valve breaks, that tank will no longer be used.	2
	Complete Failure	No pressure regulation for all tanks from tank to fuel cells. Power generation impacted.	Failure of all throttle valves is critical. No tanks can be used for power generation.	4
Pressure relief valves	Single Failure	No prevention against too high pressures after pressure regulator for single tank. Fuel cell efficiency impacted.	Each tank has its own pressure relief valve for redundancy. Failure of a single valve will not hinder use of tank but LTS should return to Basecamp to be fixed.	2
	Complete Failure	No prevention against too high pressures after pressure regulator for all tanks. Fuel cell efficiency impacted.	Failure of all pressure relief valves also does not hinder operations, but the LTS should return to Basecamp to be fixed.	2
Flow control valves	Single Failure	No prevention against too high mass flows after pressure relief valve for single tank. Fuel cell efficiency impacted.	Each tank has its own mass flow valve for redundancy. Failure of a single valve will mean that tank is no longer being used.	2
	Complete Failure	No prevention against too high mass flows after pressure relief valve for all tanks. Power generation impacted.	Failure of all valves is a critical failure and mission must be aborted.	4
Excess flow valves	Single Failure	No prevention against too high mass flows after control valve for single tank. Fuel cell efficiency impacted.	Each tank has its own excess flow valve for redundancy. Failure of a single valve will not hinder use of tank but LTS should return to Basecamp to be fixed .	2
	Complete Failure	No prevention against too high mass flow after control valve for all tanks. Fuel cell efficiency impacted.	Failure of all excess flow valves also does not hinder operations, but the LTS should return to Basecamp to be fixed.	2
Purge valves	Single Failure	No draining of water from fuel cell to single stack. Fuel cell efficiency impacted.	Two purge valves are used for redundancy. Failure of single valve will not stop operations immediately, but LTS should return to Basecamp to be fixed.	2

Continued on next page

Table 14.8 – *Continued from previous page*

Component	Failure Mode	Effect	Prevention/Response	Criticality
	Complete Failure	No draining of water from fuel cell. Power generation impacted.	Failure of all purge valves will mean that the fuel cells will be flooded with water, hindering operations. Mission will need to be aborted.	4
Battery	Complete Failure	No peak power deployment	Failure of battery is not critical. Peak power loads cannot be provided for, but the LTS would still be able to function on level ground. After failure of battery, LTS should return to Basecamp for repairs.	2
Fuel cells	Complete Failure	Single failure of cell can lead to failure of entire stack, since all are connected in series. Power generation impacted.	More research needs to be conducted on the integration of fuel cell systems to ensure reliability. Reliability of fuel cell stack themselves could be designed such that single fuel cell stack would not fail over the course of 10 years. Redundant fuel cell stacks could also be used, whereby in case of a failure in a single stack, the secondary stack will be used in place.	4

14.8. Verification and Validation

Design Tool

For the sizing of the EPS, a design tool was made which calculates all the dimensions of the tank. As inputs, the total power required, specifications of the fuel cells used, mass of reactants, nominal pressure at which it is stored, the maximum pressure that the tank will experience, the number of tanks, and material properties of the tank are needed. The outputs are then the inner diameter, total length, wall thickness and mass of each tank and of all the tanks combined. The code also outputs Figure 14.1 and Figure 14.2 for each reactant. Three main functions were created for use in the code. Two functions were used to calculate the dimensions of tanks based on inner pressure and temperature, mass of reactants and number of tanks. One function was used to output the radius, given a length constraint and another was used to output the length given a radius constraint. The final function calculates the mass of the tank given all dimensions. The outputs of the function were checked based on the inputs through the use of hand verification, whereby the inputs were used on the formulas in the function, and they were checked if they are the same.

Sensitivity Analysis

Adaptability is a key aspect with regards to the design on the LTS. This means that the LTS should be easily adapted to be able to perform in a variety of different use cases and environments. In addition, a sensitivity analysis is used to assess the robustness of the design. To do so, the power requirement and the trip duration will change and the impact on the diameter of the hydrogen tanks will be investigated. Changing the power requirements mainly impacts the fuel cell stack, while the trip duration impacts the total amount of reactants being stored onboard the LTS. Only increasing power requirements were analysed, because decreasing power requirements would simply mean a smaller fuel cell stack, and smaller and fewer tanks, which would not be difficult to accommodate for on the LTS. Therefore, the impact of increasing the power requirement to 6 kW of continuous power delivered by the fuel cells was investigated. The number of tanks was kept constant at four tanks for each reactant, because increasing the number of tanks was found not to be beneficial; the decrease in mass is marginal, but added tanks means added complexity to the feed system, and the total mass of the entire system would then increase significantly.

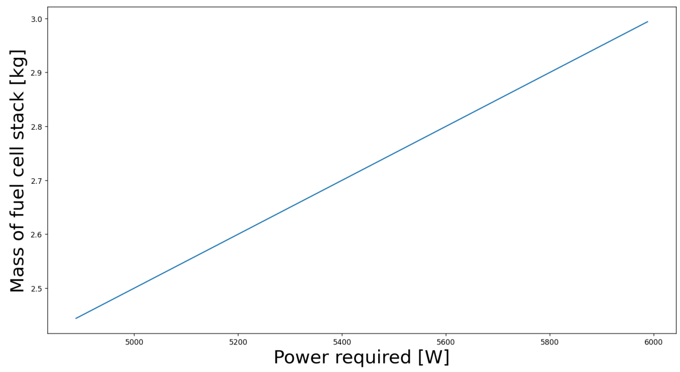

Figure 14.5: Impact of increasing power requirement on the mass of the fuel cell stack

Figure 14.5 shows the impact of increasing the power requirement on the mass of the fuel cell stack. As can be seen, there is a linear relationship, and a stack that provides 6 kW, would only weigh 3 kg and occupy a volume of 3 dm^3 showing that accommodating a bigger fuel stack would not be unfeasible.

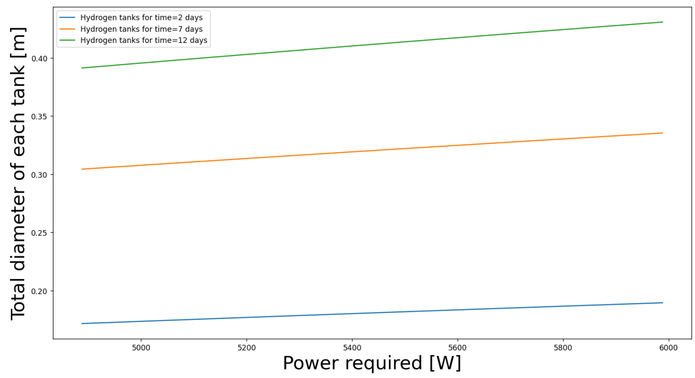

Figure 14.6: Increasing power requirement vs the total diameter of each tank for different trip durations

Figure 14.6 and shows the impact of increasing power requirements on the tank systems for trips lasting 2 days, 7 days and 12 days respectively. Increasing the power requirement for the two day trip slightly increases the total diameter of each tank, which is beneficial as that means that the frame of the LTS would need to be modified to be made slightly bigger in order to fit slightly bigger tanks to hold enough reactants for the increased power requirement. However, when the total trip duration is increased, it shows that significantly larger tanks are needed, because a lot more reactants are needed to be stored. For a 12 day trip, the total mass of the hydrogen and the tanks, given the 4888 W used for the sizing of the fuel cells, would be equal 240 kg, and the total mass of oxygen and tanks would equal 550 kg. The total diameter for each hydrogen tank then increases to around 37 cm. This shows that if the LTS were to be designed for significantly longer trip durations, the frame and suspension would need to be redesigned to accommodate the larger tanks and increased mass of the entire tank system. However, to show the robustness of the design, if the trip duration were increased to three days and requiring 5000 W of power to be produced, the total diameter for each hydrogen tank would slightly increase to 21 cm and the total hydrogen tank system would weigh 62 kg. These changes can easily be accommodated with the current design of the LTS, showing that the design is robust.

For a more complete sensitivity analysis, along with the power requirement and trip duration, the type of fuel cell could also be varied to determine the effect it has on the capabilities of the LTS. Changing fuel cells configurations would also mean that the thermal subsystem would be receiving a different amount of heat which can be used for temperature control. This would in turn mean that a different mass of insulation may be required, changing the mass of the entire LTS as whole, which would impact the structures and locomotion subsystem as they need to design for a different mass. A more complete sensitivity analysis would take into account the interactions between subsystems, and how changes in one would influence the overall design of the LTS.

Compliance Matrix

Table 14.9 shows the compliance matrix for the EPS. Requirements related to power requirements have been satisfied, however, budgetting requirements would need a further in-depth analysis that is currently not possible, as explained in Chapter 17.

Table 14.9: Compliance matrix of the EPS

Requirement ID [REQ-SSYS-TC-#.#.#.#.#]	Description	Method	Compli-ance	Justification
1.1.1.2.1	EPS shall provide 2000 W for each wheel with up to three wheels during peak loads for the suspension system	Analysis	Yes	See Section 14.3
1.1.1.2.2	EPS shall provide 7000 W during peak loads for the powertrain and mobility subsystem	Analysis	Yes	See Section 14.3
1.6.1.6.1	EPS shall accommodate an instantaneous energy discharge rate of 6000 W	Analysis	Yes	See Section 14.3
3.2.1.1.1	EPS shall allow for autonomous refuelling/recharging capabilities	Test	TBD	Infrastructure would be needed for electrolysis of water into hydrogen and oxygen gas. Refuelling would then be possible.
3.2.1.2.1	EPS shall accommodate recharging/refuelling within 30 minutes	Test	TBD	Actual refuelling times would likely vary depending on the infrastructure and system employed for refuelling on the Moon, but based on current estimates it should be around 15 to 20 minutes.
5.1.1.1.1	EPS shall have a probability of failing during its operational life of less than [TBD]	Analysis	TBD	A reliability analysis would be needed to be conducted.
5.1.1.7.1	EPS shall supply 69 W to the telecommunication subsystem	Analysis	Yes	See Section 14.3
5.1.1.7.2	EPS shall supply 1000 W to airlocks in the LTS	Analysis	Yes	See Section 14.3
5.1.1.7.3	EPS shall supply 171 W to the Guidance, Navigation and Control subsystem	Analysis	Yes	See Section 14.3
5.1.1.7.4	EPS shall supply 368 W to the cargo handling subsystem	Analysis	Yes	See Section 14.3

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Table 14.9 – *Continued from previous page*

Requirement ID [REQ-SSYS-TC-#.#.#.#]	Description	Method	Compli-ance	Justification
5.1.1.7.5	EPS shall supply 80 W to the TTC and data handling subsystem	Analysis	Yes	See Section 14.3
5.1.1.7.6	EPS shall supply 2000 W to the powertrain and mobility subsystem	Analysis	Yes	See Section 14.3
5.1.1.7.7	EPS shall supply 1088 W to the life support subsystem	Analysis	Yes	See Section 14.3
5.1.1.7.8	EPS shall supply 570 W to the thermal control subsystem	Analysis	Yes	See Section 14.3
5.7.1.1.1	Development cost of the power subsystem shall not exceed [TBD] Euros	Analysis	TBD	Parametric estimation of subsystem can be done using NAFCOM tool. See Chapter 17
5.7.1.2.1	Manufacturing costs of the power subsystem shall not exceed [TBD] Euros	Analysis	TBD	Parametric estimation of subsystem can be done using NAFCOM tool. See Chapter 17
5.7.1.3.1	Maintenance costs of the power subsystem shall not exceed [TBD] Euros	Analysis	TBD	Parametric estimation of subsystem can be done using NAFCOM tool. See Chapter 17
5.7.1.4.1	Operational costs of the power subsystem shall not exceed [TBD] Euros	Analysis	TBD	Parametric estimation of subsystem can be done using NAFCOM tool. See Chapter 17

15 | Internal Communications

Author: Dani

The Internal Communications subsystem's function is to provide a communication channel for the subsystems, it basically acts as the vehicle's nervous system, sending information and commands to all parts of the system. In the chapter, first the system architecture will be discussed, after which a design and failure analysis is taking place. Finally, a compliance matrix is present to verify requirements.

15.1. Overview

It was decided to employ one on board computer and several single board computers to create a decentralized system which allows for high redundancy and modularity. For the data connections, a Classical CAN bus was chosen to ensure functionality of the system even if one component fails.

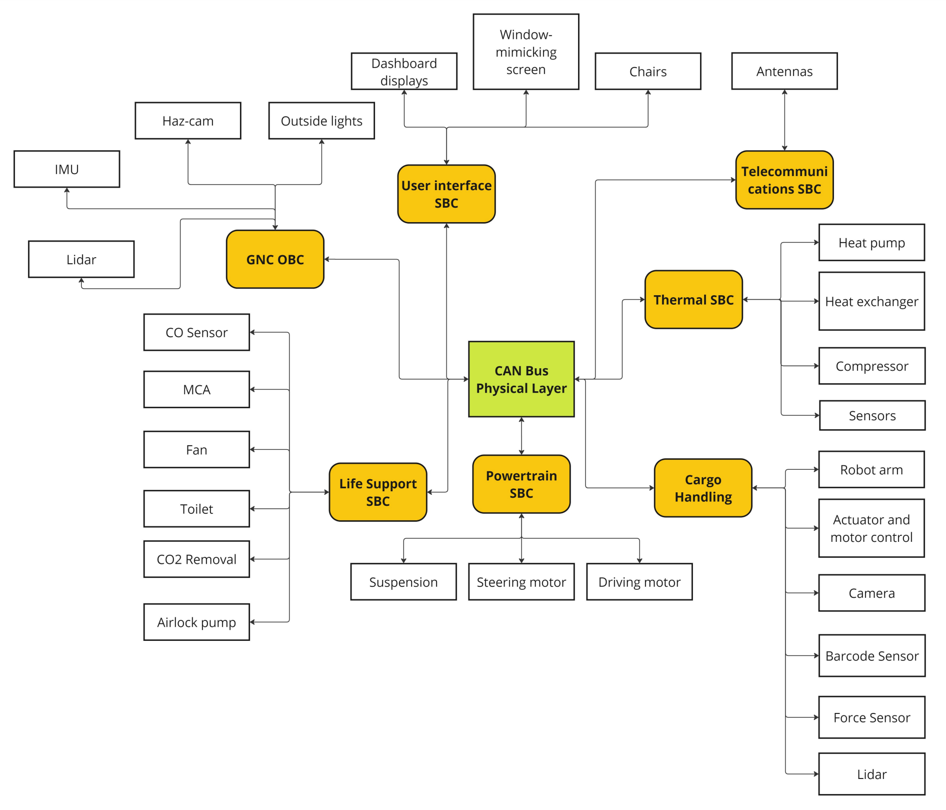

Figure 15.1: Data handling block diagram

15.2. System Architecture

As this subsystem's responsibilities are to receive, process and send data and commands to different components, it is very important to employ a redundant and robust system. For this reason, the decision was made to use a decentralized data system instead of having one big centralized on-board-computer. The most common decentralized systems employ CAN buses to allow direct communication between each individual computer board. In modern automotive engineering, it is almost standard to use as many single computer boards as possible (up to even 90), as they are very light, small and have low power consumption. Using a CAN bus is also beneficial for the prioritizing of the information that needs to be sent first. In order to estimate a first-order sizing, the CAVU OBC-1[1] by a British company, Cavu Aerospace, was chosen as a component for the single board computers.

However, it was also decided to use one on-board-computer to facilitate the most resource heavy tasks, like guidance and navigation. To estimate what an on-board-computer would mean in terms of mass, power and capabilities, The Next Generation On Board Computer[2], developed by Beyond Gravity was chosen to act essentially as a placeholder. This is a very robust and reliable computer which certainly can handle the tasks assigned to it. As it is relatively big, it was decided to be placed under the dashboard of the vehicle which is closely located to a lot of the systems that it needs to be connected to.

[1]https://satcatalog.s3.amazonaws.com/components/1432/SatCatalog_-_CAVU_AEROSPACE_-_OBC1_Satellite_On-Board_Computer_-_Datasheet.pdf?lastmod=20240110113443

[2]https://www.beyondgravity.com/sites/default/files/media_document/2023-11/Next-Generation-On-Board-Computer.PDF

The overall architecture can be seen in Figure 15.1, which is essentially the same as the electrical block diagram, due to the fact that the sensors and boards also need electricity of course.

15.3. Failure Modes

As the Internal Communication subsystem ensures that everything works properly, it needs to be highly reliable. One of the most common ways to analyse component reliability is the FMEA method which ensures that all failures can be tracked, isolated or prevented. The Internal Communication subsystem's failure mode analysis is presented in Table 15.1. This is not as extensive as it will be once the detailed design is done, however for a preliminary design it is good starting point which later on can be further developed.

Table 15.1: Failure modes of the internal communications subsystem.

Component	Failure Mode	Effect	Prevention / Response	Criticality
On Board Computer	Single Failure	Some GNC function might be lost	Go back to Basecamp to repair, while some tasks can be redistributed to other controllers	1
	Multiple Failure	Multiple component in OBC stops working	Probably a sign of serious damage, needs to be repaired at Basecamp as soon as possible	1
Single Board Computers	Single Failure	One component or subsystem stops working	Depending on which board stopped working either immediately go back to Basecamp for repair or it can wait until the LTS arrives back to it as scheduled	2
	Multiple Failure	Multiple component stops working	Probably a sign of serious damage, needs to be repaired at Basecamp as soon as possible	1

15.4. Design Analysis

By employing multiple single board computers, modularity is guaranteed, and the maintenance of any sensor or subsystem becomes very easy to do. With the on board computer positioned at the dashboard, it allows for easy access for maintenance and repair either from the inside or the outside.

15.5. Compliance Matrix

Table 15.2: Compliance matrix of the internal communications subsystem.

Requirement ID [REQ-SSYS-TD-#.#.#.#.#]	Description	Method	Compliance	Justification
1.5.1.2.1	The TTC shall continuously collect health and status from all other subsystems.	Analysis/Test	Yes/TBD	The system was designed to collect data, however testing is needed to confirm functionality.
1.5.1.2.2	The TTC shall continously process command to relevant subsystems.	Analysis/Test	Yes/TBD	The system was designed to process commands, however testing is needed to confirm functionality.
1.5.1.2.3	The TTC shall have automatic health check and alert for anomalies.	Analysis/Test	Yes/TBD	The system was designed to analyse data, however testing is needed to confirm functionality.
2.3.1.1.1	The TTC subsystem must include redundant components to ensure continuous operation in case of hardware failure	Analysis	Yes	See Section 15.2.
2.3.1.1.2	The C&DH shall have quantization error less than 0.1 %	Analysis	Yes	See Section 15.2.
2.3.1.1.3	The C&DH shall encrypt data	Analysis	Yes	See Section 15.2.
2.3.1.1.4	The C&DH shall have data rate of TBD	Analysis	Yes	See Section 15.2.
2.3.1.1.5	The C&DH shall have a handling speed of TBD MIPS	Analysis	Yes	See Section 15.2.
5.1.1.8.1	The C&DH shall not exceed 5215 bits between sync words	Analysis	Yes	See Section 15.2.
5.1.1.8.2	The C&DH shall have maximum data rate of 16 kpbs	Analysis	Yes	See Section 15.2.
5.1.1.8.3	The C&DH shall have error in data rate less than 10e-5 %	Analysis	Yes	See Section 15.2.
5.1.1.8.4	The C&DH shall have data storage capacity of 250 k-bits for data storage memory	Analysis	Yes	See Section 15.2.
5.7.1.1.1	Development cost of the TTC&Datahandling will not exceed [TBD] euros	Analysis	TBD	Stakeholder meetings concerning monetary budget constraints are planned to take place after the 10-week mark.
5.7.1.2.1	Manufacturing cost of the TTC&Datahandling will not exceed [TBD] euros	Analysis	TBD	See 5.7.1.1.
5.7.1.3.1	Maintenance cost of the TTC&Datahandling will not exceed [TBD] euros	Analysis	TBD	See 5.7.1.1.
5.7.1.4.1	Operational cost of the TTC&Datahandling will not exceed [TBD] euros	Analysis	TBD	See 5.7.1.1.

16 | Assembly and Integration

Authors: Jan, Aleksei

In this chapter the integration and assembly of the LTS is presented. In Section 16.1, an overview of the launch configuration is presented. Section 16.2 shows all the parts that need to be assembled, where to attach them and with what method.

16.1. Launch Configuration

One of the main reasons to employ a stowed configuration during launch is the launch loads; the forces and vibrations the cargo will undergo during launch are significantly larger than the ones they are designed for during operations on the moon. This major discrepancy means that all parts in the LTS impose increased loads where ever they are connected; this would mean that the LTS requires a lot of additional structural supports during launch. To mitigate this, as many parts as possible are disassembled from the LTS during launch, limiting the required additional supports.

The main reason for not removing a part is if it would add significant complications when assembling in situ; this means that these parts do need additional supports during launch. Additionally, any parts that does not impose any significant additional loads is also not removed.

The parts of the LTS that will be assembled on earth are the following:

- The shell cabin including the insulation, wiring and plumbing
- The frame and cabin will be connected.
- The supporting structure around the cabin is connected.

All remaining parts are either transported individually or in smaller assemblies. Some of the notable ones are:

- Empty radiation shielding panels, which are to be filled with lunar regolith
- The wheel and motor assemblies
- The suspension wishbones and shock-absorbers
- Toilet system including toilet waste tanks
- Smaller life support system components that can easily be disconnected and fit through the door
- Cargo handling subsystem
- O_2 and H_2 tanks

16.2. Assembly Plan

In Section 16.2, the top level cabin assembly plan is presented. This plan consists of the sequential operations that need to be performed to assemble all subsystems into and onto the pressure cabin.

Figure 16.1 presents the exploded cabin assembly of the LTS. All the different subsystem parts are grouped based upon color. These different subsystems are presented in Table 16.1. While each individual part needs to be assembled on a specific location inside of the LTS, Table 16.2 presents the sequence of the larger subsystem modules that are to be assembled. Notable is that:

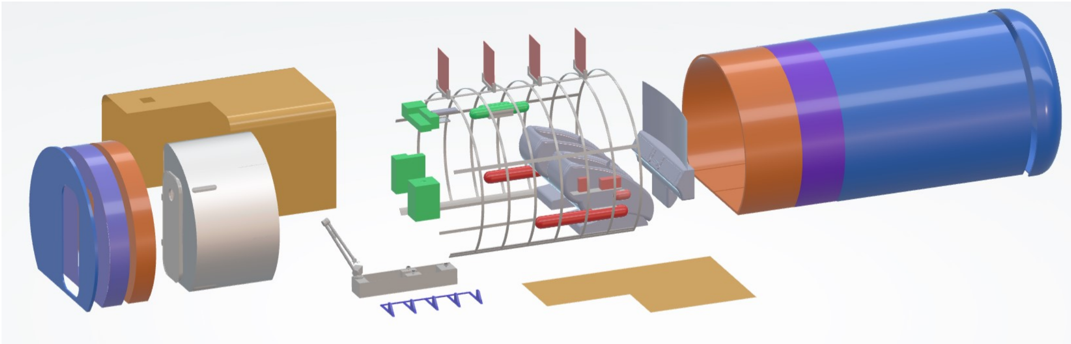

Figure 16.1: Exploded cabin assembly

Table 16.1: Subsystem assembly overview

Acronym	Subsystem	Color	Assembly place
PC	Pressure Cabin	Purple	Earth
TCI	Thermal control insulation	Orange	Earth
IC	Internal Cabin	Gold	Earth
UI	User Interface	Blue Grey	Earth
AS	Airlock Subsystem	White	Earth
SS	Structural support	Grey	Earth
RP	Radiation protection	Blue	Moon
LS	Life support	Green	Moon
TC	Thermal control	Light red	Moon
PM	Power management	Red	Moon
CH	Cargo Handling	Grey	Moon

Table 16.2: Assembly sequence of LTS subsystems

height Order	Color	Subsystem	Part	Attachment	Connection
1	Purple	PC	Main cabin	Onto chassis frame	Bolt
1.1	Orange	TC	Main Thermal insulation	Inside main cabin	Adhesive
1.2	Grey	IC	Internal cabin stiffeners	Inside main cabin	Adhesive
1.3	Gold	IC	Internal cabin floor	Onto internal cabin stiffeners	Bolt
1.4	Gold	IC	Internal cabin frame	On top of internal cabin floor	Bolt
1.5	White	AS	Airlock subsystem	Onto internal cabin stiffeners	Bolt
1.7	Blue Grey	UI	Window screen	Onto internal cabin frame	Bolt
1.8	Blue Grey	UI	Chairs	Onto cabin floor	Bolt
2	Purple	PC	Cabin bulkhead	Onto main cabin	Welding
2.1	Orange	TC	Bulkhead thermal insulation	Onto cabin bulkhead	Adhesive
3	Grey	SS	Structural support	Outside main cabin and cabin bulkhead	Bolt
4	Green	LS	Life support machinery and tanks	On top of internal cabin frame	Bolt
4.1	Green	LS	Toilet System	On main cabin floor/inside cabin frame	Bolt
4.2	Red	PM	Water capture tanks	On front right cabin frame walls	Bolt
4.3	Light Red	TC	Thermal control machinery	Onto front right cabin frame walls	Bolt
5	Blue	RS	Main cabin radiation shielding	Onto structural support	Bolt
5.1	Blue	RS	Front bulkhead radiation shielding	Onto structural support	Bolt
5.2	Blue	RS	Bulkhead radiation shielding	Onto structural support	Bolt

Besides the cabin in- and externals, a few other parts are to be assembled. These are:

- Suspension: the wishbones and shock-absorbers need to be connected to the frame using pin connections at the designated connection points.
- Wheel and motor assembly: the wheel and motor assembly needs to be connected to the wishbones and shock-absorbers, again using pin connections to ensure free movement.

- O_2 and H_2 tanks: the tanks need to be placed in between the webs in the frame and connected to the feed system. After installation they need to be covered by the matching insulation covers.

16.3. Final internal cabin layout

In this section, the final LTS design will be presented, including its internal cabin layout. Also, a brief reflection on the design limitations is presented.

In Figure 16.2, the complete assembly is presented from the outside. On top of the cabin, the radiators are visible. On the side, the cargo handling system is presented, including the robot arm. Finally, on the bottom, the chassis is presented onto which the wheels, suspensions and motors are attached.

Figure 16.2: Assembled LTS

In Figure 16.3, the internal cabin layout is visible. In the back left of the cabin, the toilet including its tanks is visible. In the front of the cabin, the fold-able chairs and window screen are presented. Finally, the airlock system is visible on the right.

Similarly to the external cabin layout, the internal cabin is driven by minimising the needed volume. The following constraints were taken into account while designing the LTS:

- At least one horizontal sleeping surface is required with the area of 800 mm x 2000 mm.
- The internal cabin height shall be at least 2000 mm for comfortable movement.
- A toilet is essential to be on board.
- Each main pathway shall be at least 500 mm wide.
- An unobstructed door is essential to connect the airlock with the cabin.

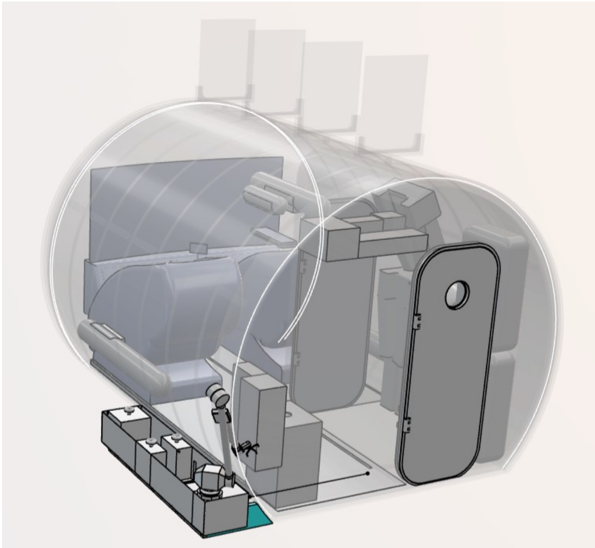

Figure 16.3: Ghost view integrated cabin

The following improvements are essential for further development of the LTS design:

- Electronic wiring, sensors and heat piping are not modeled in this design but have been taken into account for adding tolerances.
- For multiple electronic and thermal components attached on the outside of the LTS, cutouts are expected to be necessary. This is not modeled yet and could add local complexities to the design.
- Cabinets, a water dispenser and a table are not implemented into the design but in between the internal cabin wall and the thermal insulation layer, much more space could be utilised for this.
- Many parts, such as radiation protection and thermal insulation are modeled as one structure. In reality, these are subsystems consisting of multiple parts fitting inside of these dimensions.

17 | Budget breakdown

Author: Ishaan

In this chapter, the mass, power budgets will be shown and cost estimates will be introduced.

17.1. Mass and Power Budgets

Table 17.1: Table showing mass and power requirement of each subsystem

Subsystem	Power requirement [W]	Mass [kg]	Volume [m^3]
Powertrain and mobility	2000	544	-
Life support	1088	1771	1.43
Thermal	570	1178	3.70
Cargo handling (only storage)	368	21.2	0.135
Telecommunication	72	3.09	-
Guidance, navigation and control	171	28	-
User interface (only chairs)	266	210	1.66
Internal communication	82	12.6	-
Structures (entire LTS)	0	1390	34.14
Power	0	430	0.569

Table 17.1 shows a summary of the total masses and nominal power requirements of each subsystem. These requirements are handled by the fuel cell system. Peak loads are not included, as these are managed by the onboard battery. A range budget is not provided, because the LTS was designed for a range of 300 km over a two day trip. A delta V budget is also not provided because it was deemed that the transit to the Moon would be handled by the Starship, and therefore, is out of the scope of this project. Using Table 17.1, the total mass comes out to be 5796.5 kg. The total power requirement cannot be inferred from this table, as different subsystems are active during different operational modes, and the fuel cells were sized for the most limiting case. Similarly, the total volume occupied by the LTS is simply the volume of the structures subsystem, since all the other subsystems are within the LTS, therefore the total volume of the LTS is 34.14 m^3. More specific information regarding power can be seen in Chapter 14.

17.2. Cost Budget

A cost-estimate relationship for the development and manufacturing of a single unit was found for an in-space habitat, developed by the Georgia Institute of Technology [74]. The relationship to determine the development cost is given by:

$$C = 1457.7 \cdot W^{0.0856} \tag{17.1}$$

Here, W is the mass of the vehicle, and C is the cost in million USD in FY2012. The relationship to determine the manufacturing cost of one unit is given by:

$$C = 46.624 \cdot W^{0.2146} \tag{17.2}$$

This relationship was deemed to be representative of the cabin of the LTS, as the in-space habitat is used to house a crew of four people in the vacuum of space, which means that it would include thermal protection, radiation protection and life support capabilities to keep humans safe, similar to what is required on the LTS. To account for the locomotion subsystem, which is not included in the in-space habitat, an additional 376 million USD are present in the development budget. This figure comes from the published development cost of NASA's LRV [1], as it was deemed appropriate to compare the development cost of the locomotion subsystem of the LTS to that of an unpressurized Lunar rover, whose main function was simply to move two astronauts on the Lunar environment. This was 38 million USD in 1971, resulting in the mentioned 376 million USD accounting for inflation.

Using Equation 17.1 and Equation 17.2, and accounting for the cost of locomotion and inflation between 2012 and 2024, the total development cost can be estimated as 4.6 billion USD and a manufacturing cost of 410 million USD. However, the data points used to determine the parametric equation have a weight between 10 and 12 tonnes, while the LTS weighs around 6 tonnes, and as said previously, the in-space habitat is used to house a crew of four, while the LTS houses a crew of two. This means that the budget is likely an overestimate, therefore, to account for these differences a lower margin of 50% should be used. This means the budget for the development of the LTS would be around 2.28 to 4.57 billion USD, and the cost per LTS would be between 205 and 410 million USD. Launch costs can also be estimated. It was decided that Starship will be used to transport the LTS to the Moon, which would cost 3500 USD/kg, which means it would cost 20.3 million USD to transport the entire LTS to the Moon [75].

Further breakdown of the budget to a subsystem level is not possible at this point,

[1] https://nssdc.gsfc.nasa.gov/planetary/lunar/apollo_lrv.html

because appropriate parametric equations, representative of the LTS were not found. To determine a more accurate budget , the NASA-Air Force Cost Model (NAFCOM) could be employed. NAFCOM employs an extensive database of historical space missions and various cost-estimating methods to determine costs of subsystems [76]. Using this tool would involve choosing previous manned spaceflight missions that are representative of the LTS, inputting the work-breakdown structure of the project and system-level requirements. By using the work-breakdown structure, NAFCOM is able to provide user specific results. The cost-estimating relationships also take into account complexity and technological readiness levels, which allow for more accurate estimates [77].

18 | Verification and Validation

Author: Thyme, Bart

Verification and Validation (V&V) are essential processes to increase the confidence in the proposed design. Different approaches can be used, in this case, the V-model is followed. In the respective chapters of each subsystem, a compliance matrix was presented, functioning as subsystem requirement verification. In this chapter, the verification of the system and mission requirements will be described, along with the validation of the system.

18.1. Verification of System Requirements

Verifying the system requirements is essential in the V&V of a design. Table 18.1 presents each of these requirements, along with the verification method exploited, the design's compliance to it, and a justification for it.

Table 18.1: Compliance matrix of system requirements

Requirement ID [REQ-SYS-#.#.#.#]	Description	Method	Compliance	Justification
1.1.1.1	Per sortie and under nominal operating conditions, the LTS shall have a range of at least 300 km	Analysis/Testing	Yes/TBD	Testing to be done on Earth by prototype
1.1.1.2	The LTS shall be able to store energy up to 400 kWh	Inspection	Yes	Total energy stored in battery/fuel tanks is 400 kWh
1.2.1.1	The LTS shall be able to support a cargo payload up to 50 kg	Inspection	Yes	There is support for 5x10 kg
1.3.1.1	The LTS shall be able to support a cargo payload volume of at least 0.135 m^3	Inspection	Yes	The total payload volume is 0.135 m^3
1.4.1.1	The LTS shall be able to maintain two way communication with the crew	Inspection/Testing	Yes/TBD	Test will be for two way communication capability together with 1.4.1.2
1.4.1.2	The LTS shall be able to maintain two way communication with the base camp	Inspection/Testing	Yes/TBD	Test will be for two way communication capability together with 1.4.1.1
1.4.1.3	The LTS shall communicate with the base camp every 30 minutes	Inspection	Yes	The Telecommunication subsystem was designed in a way that telemetry data can be sent at any time.

Continued on next page

Table 18.1 – *Continued from previous page*

Requirement ID [REQ-SYS-#.#.#.#	Description	Method	Compliance	Justification
1.5.1.1	The LTS shall be able to be teleoperated from a distance of 150 km	Inspection/Testing	Yes/TBD	LTS can be tele-operated via LunaNet, therefore basically anywhere on Earth/Moon. Testing will be done on Earth without crew
1.5.1.2	The LTS shall be able to detect the need for maintenance tasks on itself	Inspection/Demonstrat	Yes/TBD	Demonstration still to be done
1.6.1.1	The LTS shall be able to traverse terrain with an longitudinal inclination of up to 20 degrees	Analysis/Testing	Yes/TBD	Testing will be done on Earth with assembled prototype
1.6.1.2	The LTS shall be able to traverse terrain with a lateral tilt of 20 degrees	Analysis/Testing	Yes/TBD	Testing will be done on Earth with the prototype
1.6.1.3	The LTS shall not topple over	Analysis/Demonstratio	Yes/TBD	Demonstration will be done with the prototype
1.6.1.4	The LTS shall have a minimum nominal operating speed of 1.94 m/s	Analysis/Testing	Yes/TBD	Testing shall be done with the prototype
1.6.1.5	The LTS shall have a minimum turning rate of 20 deg/s	Analysis/Testing	Yes/TBD	Testing will be done with the prototype
1.6.1.6	The LTS shall have a peak energy discharge rate up to 30 kW	Demonstration/Analys	TBD/Yes	Power subsystem will demonstrate it can deliver peak power on Earth
1.6.1.7	The LTS shall have a maximum breaking distance smaller than 5 m	Analysis/Testing	Yes/TBD	Testing will be done by the prototype
1.6.1.8	The LTS shall have a maximum turn radius of 4 m	Analysis/Demonstratio	Yes/TBD	Demonstration will be done on Earth by the prototype
1.7.1.1	The LTS shall be able to operate for up to 2 Earth days	Analysis/Demonstratio	Yes/TBD	Demonstration could be done on Earth with the prototype
2.1.1.1	The LTS shall be able support a crew of 2 members	Inspection	Yes	
2.2.1.1	The LTS's crew shall be exposed to pressures in the range between 26.2 kPa and 103 kPa	Analysis/Testing	Yes/TBD	Pressure cabin will be tested "pressure proof" and burst tested
2.2.1.2	The LTS shall meet the functional anthropocentric accommodation standards specified in item 4102 of NASA-STD-3001 Standards	Inspection	Yes	
2.2.1.3	The atmosphere the LTS's crew is exposed to shall contain diluent gas of at least 30%	Inspection	TBD	Inspection to be done before launch
2.2.1.4	For pressure differences greater than 1.0 psi, The LTS's crew shall not be exposed to pressure rates greater than 13.5 psi/min	Inspection	TBD	Inspection to be done before launch
2.2.1.5	The LTS's crew shall be exposed to temperatures in the range 20-25 degrees Celsius	Analysis	Yes	
2.2.1.6	The LTS's crew shall be exposed to a relative humidity in the range 30-60%	Analysis	Yes	
2.2.1.7	The LTS shall ensure a ventilation rate that is sufficient to avoid CO2 and thermal pockets, in accordance with item 6107 of the NASA-STD-3001 standards	Inspection	TBD	Inspection to be done before launch
2.2.1.8	The LTS shall provide aesthetically-acceptable, potable water that is chemically and microbiological safe for human use, including drinking, food re-hydration, personal hygiene, and medical needs in accordance with item 6026 of the NASA-STD-3001 standards	Inspection	Yes	

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Table 18.1 – *Continued from previous page*

Requirement ID [REQ-SYS-#.#.#.#	Description	Method	Compli-ance	Justification
2.2.1.9	The LTS shall ensure a quantity of potable water per crew member per day in accordance with item 6109 of the NASA-STD-3001 standards	Inspection	Yes	Water tanks available are sufficient
2.2.1.10	The LTS shall provide water at temperatures in accordance with item 6110 of the NASA-STD-3001 standards	Analysis/Inspection	Yes/TBD	Inspection to be done before launch
2.2.1.11	The LTS shall not exceed translational load limits set by item 6064 of the NASA-STD-3001 standards	Inspection	Yes	
2.2.1.12	The LTS shall not exceed the rotational rate limits set by item 6065 of the NASA-STD-3001 standards	Inspection	Yes	
2.2.1.13	The LTS shall limit crew exposure to radiation in accordance with item 6097 of NASA-STD-3001 standards	Testing/Inspection	TBD/TBD	Testing of radiation shield, testing of integrated radiation shield on prototype and inspection of radiation shield on final LTS to be done before launch
2.2.1.14	The LTS shall provide food with a quality and quantity specified in items 7001 to 7003 of the NASA-STD-3001 standards	Inspection	TBD	Inspection of food to be done before launch
2.2.1.15	The LTS shall handle waste in accordance with item 7020 to 7029 of the NASA-STD-3001 standards	Inspection	Yes	
2.2.1.16	The LTS shall provide medical capabilities in accordance with item 7043 of the NASA-STD-3001 standards	Inspection	Yes	
2.2.1.17	The LTS shall limit the vibrations such that the total vibrational loads for frequencies between 0.5 Hz and 80 Hz are with the limit established by item 6090 of the NASA-STD-3001 standards	Analysis/Testing	TBD/TBD	A complete vibrational analysis is still to be done after which the vibrations test can be done.
2.2.1.18	The LTS shall use no chemicals which endanger the health of the crew and can not be contained	Analysis/Inspection	Yes/TBD	Inspection to be done before launch
2.2.1.19	The LTS shall be able to detect and extinguish fires	Inspection/Demonstrat	Yes/TBD	System is sized for, demonstration to be done on Earth
2.2.1.20	The LTS shall provide the crew sleeping accommodations	Inspection	Yes	
2.2.1.21	The LTS shall have a noise level below 75 dB	Inspection	TBD	Actual noise level inside cabin needs to be inspected.
2.3.1.1	The LTS system and subsystem design shall be such that no single point failure shall abort the mission and no second failure should endanger the crew	Analysis/Inspection	TBD/TBD	Complete analysis to be done + complete inspection on Earth before launch
2.4.1.1	Following the accessibility standards from NASA-3001 the LTS shall be accessible by the crew on the lunar surface	Inspection	Yes	
3.1.1.1	The LTS shall have autonomous capability (corresponding to level 5 in terrestrial cars	Demonstration	TBD	Demonstration of autonomous driving to be done on Earth before launch

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Table 18.1 – *Continued from previous page*

Requirement ID [REQ-SYS-#.#.#.#	Description	Method	Compli-ance	Justification
3.1.1.2	The LTS shall be able to deduce its location on the lunar surface with an accuracy of 2 m	Analysis/Testing	TBD/TBD	A more detailed analysis is to be done to determine actual accuracy of deducing location after which a test can be done before launch
3.2.1.1	The LTS shall have autonomous refueling/recharging capabilities	Demonstration	TBD	Demonstration to be done by prototype
3.2.1.2	The LTS shall be able to "refuel/recharge" within 30 min	Demonstration	TBD	Demonstration to be done by prototype
3.3.1.1	The LTS shall have autonomous cargo handling capabilities	Demonstration	TBD	Demonstration to be done by prototype
3.4.1.1	The LTS and its systems shall be able to go in and out of hibernation mode at the discretion of the user	Inspection	Yes	
3.5.1.1	The LTS shall have a Loss-of-Motion probability of less than 0.001	Analysis	TBD	A more detailed analysis is necessary once design is more mature
3.6.1.1	The LTS shall have a Loss-of-Communication probability of less than 0.0001	Analysis	TBD	A more detailed analysis is necessary once design is more mature
4.1.1.1	The LTS shall be able to withstand launch loads of 6 g in stowed configuration	Analysis/Testing	TBD	Analysis in FEA + Testing of structure in stowed configuration
4.2.1.1	The LTS shall be able to be assembled on the Moon within a time of 14 days	Demonstration	TBD	Demonstration on Earth
5.1.1.1	The LTS shall have a probability of failing during its operational life of less than [TBD]	Analysis	TBD	A more detailed analysis is required once design is more mature
5.1.1.2	The LTS shall be able to withstand temperatures in the range between 40 -200 Kelvin	Analysis/Testing	TBD/TBD	Analysis on the maximum environmental temperature to be done, Testing of exposed temperatures of invidual components
5.1.1.3	The LTS subsystems shall have an operational life of at least 10 years	Analysis	TBD	Analysis to be done once design is more mature
5.1.1.4	The LTS shall prevent dust from adhering to the vehicle	Analysis/Testing	Yes/TBD	
5.1.1.6	The LTS shall have a natural frequency higher than 80 Hz	Analysis/Testing	TBD/TBD	Analysis to be done using FEA software + Testing
5.1.1.7	The LTS shall generate a continues total power of 6 kW	Inspection	Yes	
5.1.1.8	The LTS shall be able to store data up to [TBD]	Inspection	Yes	
5.2.1.2	The LTS shall allow the replacement of modular part within [TBD] hours/days	Demonstration	TBD	Demonstration of replacing modular part to be done
5.2.1.3	The LTS shall be accessible for inspection and maintenance purposes	Inspection	Yes	
5.4.1.1	The LTS shall not discharge any waste during operation	Inspection	Yes	
5.5.1.1	at EOL, the LTS should be recyclable to Chapter 21	Demonstration/Analys	TBD/TBD	Analysis to be done, Demonstration on how parts are recycled to be done
5.7.1.1	The cost of the development cost of one LTS shall not exceed 4.6 billion USD	Inspection/Analysis	Yes	
5.7.1.2	Manufacturing cost of one LTS shall not exceed 410 million USD	Inspection/Analysis	Yes	

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Table 18.1 – *Continued from previous page*

Requirement ID [REQ-SYS-#.#.#.#	Description	Method	Compliance	Justification
5.7.1.3	Operational cost of one LTS shall not exceed [TBD] Euros	Inspection/Analysis	TBD	This cost are still TBD
5.7.1.4	Maintenance cost of one LTS shall not exceed [TBD] Euros	Inspection/Analysis	TBD	
5.7.1.5	The mass shall not exceed 100 tons	Analysis	TBD	This cost are still TBD

Except for a few exceptions, the verification of the system requirements can be split up into two categories: prototype and inspection. In the following section, these will be addressed.

Prototype

It was found that to verify some requirements, building a prototype is necessary. The requirements planned to be tested with a prototype are listed in Table 18.2. The detailed scope of the prototype will need to be refined at a later stage, but it will only consist of parts necessary for the testing such as the structure and power train for example.

Table 18.2: Prototype testing list

Requirement ID	Test description
REQ-SYS-1.1.1.1	The prototype will test if its range is 300 km on Earth in an artificial environment imitating the lunar environment.
REQ-SYS-1.5.1.1	The prototype will be tested for teleoperation in an imitated lunar environment.
REQ-SYS-1.6.1.1 - REQ-1.6.1.5 + 1.6.1.7 & 1.6.1.8	The prototype will be tested for longitudinal and lateral inclination, whether it topples over in extreme conditions, its speed, its turning rate, braking distance at maximum operating speed, and maximum turn radius, all in the same test campaign.
REQ-SYS-3.2.1.1, REQ-3.2.1.2 & REQ-3.3.1.1	The prototype will demonstrate autonomous refueling/recharging in the same test campaign.
REQ-SYS-2.1.1.17 & REQ-5.1.1.6	The prototype will undergo a vibrations test to determine if it sufficiently dampens frequencies between 0.5 and 80 Hz, and to find its natural frequency.
REQ-SYS-2.2.1.19	The prototype will demonstrate the detection and extinguishing of fires
REQ-SYS-4.2.1.1	The prototype will be tested to determine how quickly it can be assembled.
REQ-SYS-5.1.1.4	The prototype will be tested to evaluate whether the design effectively prevents dust from adhering to the vehicle.
REQ-SYS-5.2.1.2	The prototype will demonstrate that the replacement of a modular part is possible and will measure the time required for replacement.

Facilities needed

To conduct the test for the prototype several facilities are necessary. One of the most important required facilities required is something imitating the lunar environment. With this facility most of the test from the prototype test plan can be conducted.

Inspections

For some requirements, verification is only possible by inspecting the manufactured product. These are listed in Table 18.3.

Table 18.3: Inspection list

Requirement ID	Inspection description
REQ-2.2.1.3	Diluent gas concentration inspection >30%
REQ-2.2.1.4	Pressure rates Inspection <13.5 psi/min
REQ-2.2.1.5	Temperature inspection 20-25 degrees Celsius
REQ-2.2.1.6	Humidity inspection 30-60%
REQ-2.2.1.7	Ventilation inspection
REQ-2.2.1.8	Drinking water inspection
REQ-2.2.1.9	Drinking water quantity inspection
REQ-2.2.1.10	Water temperature inspection
REQ-2.2.1.14	Food quality inspection
REQ-2.2.1.18	Dangerous chemical inspection
REQ-2.2.1.19	Fire detection inspection
REQ-2.3.1.1	Points of failure inspection

Other Tests

Some of the following requirements are exceptions to the previous 2 categories. This section deals with those requirements.

Points of failure:
REQ-SYS-2.3.1.1: A more refined analysis on single points of failure is necessary, and when this is done, a new inspection list should be created where necessary.

Two-way communications:
REQ-SYS-1.4.1.1 & **REQ-SYS-1.4.1.2**: The communication system will be tested for two-way communication.

Pressure cabin:
REQ-SYS-2.2.1.1: The pressure cabin needs two tests: a burst pressure test and a pressure proof test. The burst pressure test will verify the pressure at which the pressure cabin will burst. The pressure proof test will verify whether the pressure cabin can handle the operational pressure difference. This should also be done with the final version before launch.

Radiation:
REQ-SYS-2.1.1.13: The radiation panels will be tested as individual components and as an assembly on the prototype.

18.2. Mission Validation

Table 18.4: Compliance matrix of Mission requirements

Requirement ID [REQ-MIS #.#.#.]	Description	Method	Compliance	Justification
1.1.1	The LTS shall have a range per sortie of at least 300 km	Testing	TBD	Testing done by prototype
1.2.1	The LTS shall be able to support a cargo payload of up to 50 kg	Inspection	Yes	
1.3.1	The LTS shall be able to support a cargo payload of up to 0.135 m^3	Inspection	Yes	
1.4.1	The LTS shall be able to maintain two way communication	Demonstration	TBD	Will be done by prototype

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Table 18.4 – *Continued from previous page*

Requirement ID [REQ-MIS #.#.#.]	Description	Method	Compliance	Justification
1.5.1	The LTS shall be able to be teleoperated from a distance of [TBD]	Demonstration	TBD	Will be done by prototype
1.6.1	The LTS shall be able to move on the lunar surface	Testing	TBD	Testing will be done on Earth
1.7.1	The LTS shall be able to operate for up to 2 Earth days	Analysis	Yes	
2.1.1	The LTS shall be able to support a crew of up to 2 members	Inspection	Yes	
2.2.1	The LTS shall meet the NASA-STD-3001 standards for crew health	Inspection	TBD	Inspection to be done with inspection list
2.3.1	The LTS system and subsystem design shall be such that no single-point failure shall abort the mission and no second failure shall endanger the crew	Analysis/Inspection	TBD	Complete analysis to be done
2.4.1	The LTS shall be accessible by the crew on the lunar surface	Inspection	Yes	
3.1.1	The LTS shall have autonomous capability (corresponding to level 5 in terrestrial cars)	Demonstration	TBD	Demonstration by prototype
3.2.1	The LTS shall have autonomous refueling/recharging capabilities	Demonstration	TBD	Demonstration to be done by prototype
3.3.1	The LTS shall have autonomous cargo handling capabilities	Demonstration	TBD	Demonstration to be done by prototype
3.4.1	The LTS and its systems shall be able to be turned on and off at the discretion of the user	Inspection	Yes	
3.5.1	The LTS shall have a Loss-of-Motion probability of less than 0.001	Analysis	TBD	Analysis to be done at a later design stage
3.6.1	The LTS shall have a Loss-of-Communication probability of less than 0.0001	Analysis	TBD	Analysis to be done at a later design stage
4.1.1	The LTS shall be able to withstand 6 g launch loads in stowed configuration	Testing	TBD	Testing of structure still to be done
4.2.1	The LTS shall be able to be assembled on the Moon within a time of 14 days	Demonstration	TBD	Demonstration done by prototype
5.1.1	The LTS shall have an operational life of at least 10 years	Analysis	TBD	More detailed analysis necessary
5.2.1	The LTS shall employ a modular design	Inspection	Yes	
5.4.1	The LTS shall not discharge any waste during operation	Inspection	Yes	
5.5.1	At EOL, the LTS should be recyclable to Chapter 21	Inspection	Yes	
5.7.1	The cost of the total mission shall not exceed [TBD] Euros	Inspection	Yes	

With the verification of the mission requirements finally addressed, the total system can be validated with the mission needs. To reiterate the mission needs:

- **SN-1**: The LTS needs to enable exploration of the lunar surface
- **SN-2**: The LTS needs to be able to accommodate the crew
- **SN-3**: The LTS needs to have autonomy level 5
- **SN-4**: The LTS needs to be manufacturable
- **SN-5**: The design & development process needs to be sustainable

SN-1 is met if the LTS is able to move across the lunar surface, budgeting for exploration equipment. SN-2 is met if the LTS has a pressurised cabin in which the crew can survive in for 2 Earth days, including sleeping chairs and basic needs of human beings. SN-3 is met by having autonomous driving/navigating, cargo handling and refueling all to be demonstrated first on Earth. SN-4 is met through the production plan. SN-5 is met, for the structure for example, by choosing materials which are available on the Moon, giving the design future potential to be sustainable for the Moon.

19 | Technical Risk Assessment

Author: Lee

In order to judge the importance of a risk, two metrics will be used, namely, the severity of a risk and its likelihood. The severity of a risk will be judged using the scale shown in Table 19.1 and using the risk's impact on the overall progress of the entire project. The likelihood of a risk occurring will be judged using the scale in Table 19.2. The categorized risks, based on their severity and likelihood, will be visually presented via a risk map in Table 19.6 for the risks before mitigation strategy and in Table 19.7 for the risks after mitigation strategy. Lastly, the relationships between causes, mitigation, risk, contingency, and outcomes will be presented via bowtie diagrams in Section 19.4.

19.1. Risk Identification

Table 19.1: Risk Severity Score

Score	Severity	Consequence of Severity
4	Catastrophic	Mission failure or severe non-achievement of performance. This would mean that at least one of the previously listed requirements will surely not be met.
3	Critical	Mission success is questionable or some reduction in technical performance. This means that the achievement of at least one requirement is questionable.
2	Marginal	Degradation of secondary mission or small reduction in technical performance. None of the requirements are not met, but practicality is heavily affected.
1	Negligible	Minor inconvenience or non-operational impact.

After each risk has been identified, a mitigation strategy will be employed to ensure that the occurrence of a risk is less likely. Based on this mitigation strategy, each risk will have a post-strategy risk magnitude, again determined using Table 19.6. In the event that a risk does occur, a contingency plan is needed which aims at reducing the impact when a risk actually does play out.

Table 19.2: Risk Likelihood Score

Score	Likelihood Label	Likelihood
E	Very high	Greater than or equal to 70 %
D	High	Greater than or equal to 50 % and lower than 70 %
C	Moderate	Greater or equal to 30 % and lower than 50 %
B	Low	Greater or equal to 1 % and lower than 30 %
A	Very low	Lower than 1 %

19.2. Risk Register

The risk register serves both as a tool and a database to identify and store the potential risks and uncertainties during the project. To combat these risks, a handful of actions can be taken. **Reduce/avoid** the impact of the risk to ensure a close to risk-free environment. **Transfer** the risk to another source or organization to keep the required resources on risk mitigation to a minimum. In case a risk is not manageable in any way, it can be **accepted**, and a proper contingency plan should be drawn up for this possibility. To reduce the likelihood of a risk, it can be **controlled**, to ensure lowered chance of the risk happening. Finally, the risks can be **monitored** to keep up with their likelihood and status, in order to change plans depending on the updated situation. With these actions in mind, mitigation methods and contingency plans can be made, completing the risk register with all the information one might need to ensure a successful project.

As the risks from RSK-TRM-001 to RSK-TRM-024 have been previously identified and mitigated, these risks were taken into account for designing the subsystems. After the subsystem design, the failure or damaging of three subsystems were considered to be most critical for the mission operation: cabin, power, and communication. Damage to the cabin structure, including the potential for fire or pressure vessel puncture, is directly linked to crew safety and mission success. Also, battery and fuel cell malfunction in the power system lead to a loss of power which is critical for various subsystems, causing mission failure or even endangering crew safety. Lastly, failure in the communication system, such as hardware and software malfunctions, causes mission failure due to communication failure, which is essential for data handling and guidance, navigation & control(GNC). The new risks related to these critical subsystems will be addressed in this section, and entire risks including previously identified ones will be visually presented in Section 19.3 and Section 19.4.

Table 19.3: Risk of cabin damage

Identifier: RSK-TRM-025	**Risk Title:** Cabin damage		**Responsible Member:** Life support engineer
Risk Impact: The electrical malfunction and failure of thermal control system and structure damage the cabin and endanger life of crew			
Likelihood	**Severity**	**Risk Index**	**Risk Magnitude**
D	4	D4	**HIGH**
Risk Action: **Reduce**	**Risk Mitigation Method:** i) Electrical insulation ii) Redundant thermal control system iii) Thermal insulation of cabin structure iiii) Radiation shielding of cabin structure		
Reduced Risk Index:	**A4**	**Reduced Risk Magnitude:**	**VERY LOW**
Risk Contingency Plan: Shut down the system with electrical malfunction, Use back up thermal control system, Emergency patching for cabin structure			

Table 19.4: Risk of power system failure

Identifier: RSK-TRM-026	**Risk Title:** Power system failure		**Responsible Member:** Power engineer
Risk Impact: Battery & fuel cell malfunction and electrical component failure lead to power system failure, which lead to loss of power supply for various subsystem and potential fire			
Likelihood	**Severity**	**Risk Index**	**Risk Magnitude**
D	4	D4	**HIGH**
Risk Action: **Reduce**	**Risk Mitigation Method:** i) Use thermal resistant material ii) Regular maintenance, iii) Redundant wiring, iiii) Dust shielding,iv) Redundant power source		
Reduced Risk Index:	**A4**	**Reduced Risk Magnitude:**	**VERY LOW**
Risk Contingency Plan: Use of back up battery or other redundant power source, Isolate the part with malfunction and repair			

Table 19.5: Risk of communication system failure

Identifier: RSK-TRM-027	**Risk Title:** Communication system failure		**Responsible Member:** Data handling engineer
Risk Impact: Malfunction for hardware such as antenna and sensors and communication software lead to communication system failure, leading to failure of mission			
Likelihood	**Severity**	**Risk Index**	**Risk Magnitude**
E	4	E4	**VERY HIGH**
Risk Action: **Reduce**	**Risk Mitigation Method:** i) Redundant hardware ii) Thermal shielding for hardware iii) Dust shielding for the hardware, iiii) Rigorous testing & Redundant software		
Reduced Risk Index:	**B3**	**Reduced Risk Magnitude:**	**LOW**
Risk Contingency Plan: Use redundant hardware, Repair, Utilize failover mechanism for software failure			

19.3. Risk Map Overview

Table 19.6: Risk Map: Before mitigation

Likelihood \ Severity	1	2	3	4
E	LOW	MEDIUM RSK-TRM-016	HIGH RSK-TRM-001 RSK-TRM-003 RSK-TRM-006	VERY HIGH RSK-TRM-004 RSK-TRM-027
D	LOW	LOW	MEDIUM RSK-TRM-010	HIGH RSK-TRM-025 RSK-TRM-026
C	VERY LOW	LOW RSK-TRM-005 RSK-TRM-013 RSK-TRM-021 RSK-TRM-023	MEDIUM RSK-TRM-002 RSK-TRM-008 RSK-TRM-022 RSK-TRM-024	MEDIUM RSK-TRM-020
B	VERY LOW	VERY LOW	LOW RSK-TRM-007	LOW RSK-TRM-009 RSK-TRM-011 RSK-TRM-015 RSK-TRM-017 RSK-TRM-018
A	VERY LOW	VERY LOW	VERY LOW	VERY LOW RSK-TRM-012 RSK-TRM-014 RSK-TRM-019

Table 19.7: Risk Map: After mitigation

Likelihood \ Severity	1	2	3	4
E	LOW	MEDIUM	HIGH	VERY HIGH
D	LOW	LOW	MEDIUM	HIGH
C	VERY LOW	LOW RSK-TRM-003 RSK-TRM-006 RSK-TRM-016	MEDIUM	MEDIUM
B	VERY LOW	VERY LOW RSK-TRM-004 RSK-TRM-005 RSK-TRM-013 RSK-TRM-021 RSK-TRM-023	LOW RSK-TRM-001 RSK-TRM-002 RSK-TRM-010 RSK-TRM-027	LOW
A	VERY LOW	VERY LOW RSK-TRM-017	VERY LOW RSK-TRM-007 RSK-TRM-008 RSK-TRM-009 RSK-TRM-011 RSK-TRM-018 RSK-TRM-019 RSK-TRM-022 RSK-TRM-024	VERY LOW RSK-TRM-012 RSK-TRM-014 RSK-TRM-015 RSK-TRM-020 RSK-TRM-025 RSK-TRM-026

19.4. Bowtie Diagram

The Bowtie Diagram offers a clear and visual representation of the relationship between potential hazards, their causes (threats), and the resulting consequences. Based on the risks identified in Section 19.2, this Bowtie digram will be presented in Figure 19.1,Figure 19.2, and Figure 19.3 for the risks of cabin, power, and communication respectively.

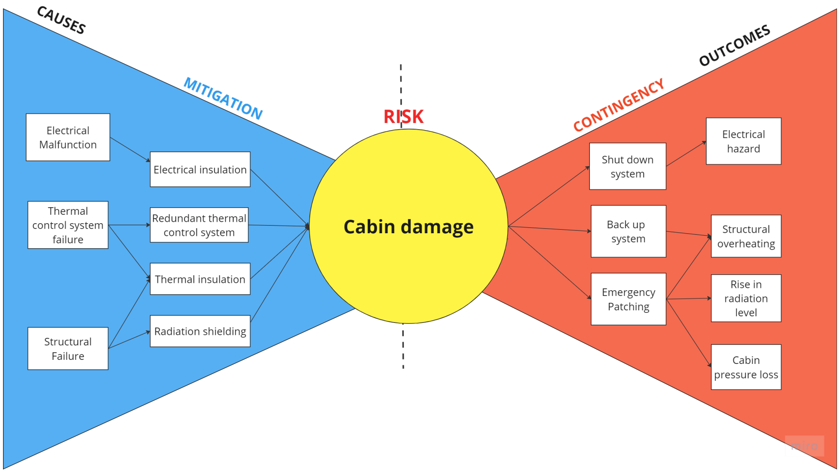

Figure 19.1: Bow Tie Diagram for cabin risk

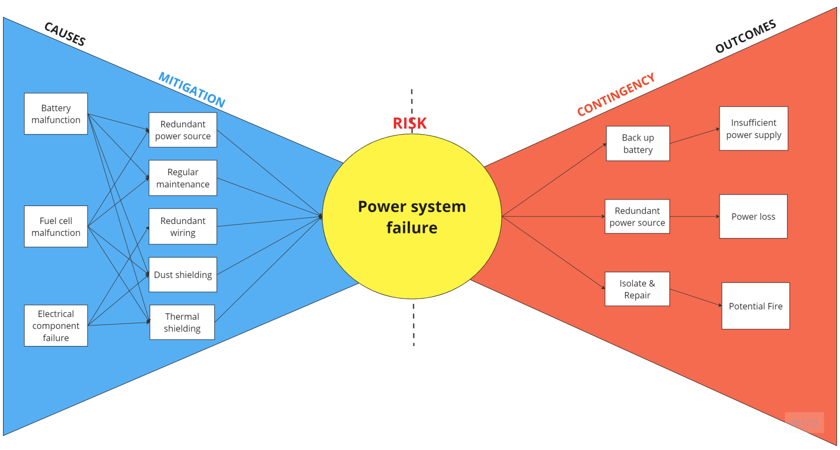

Figure 19.2: Bow Tie Diagram for power risk

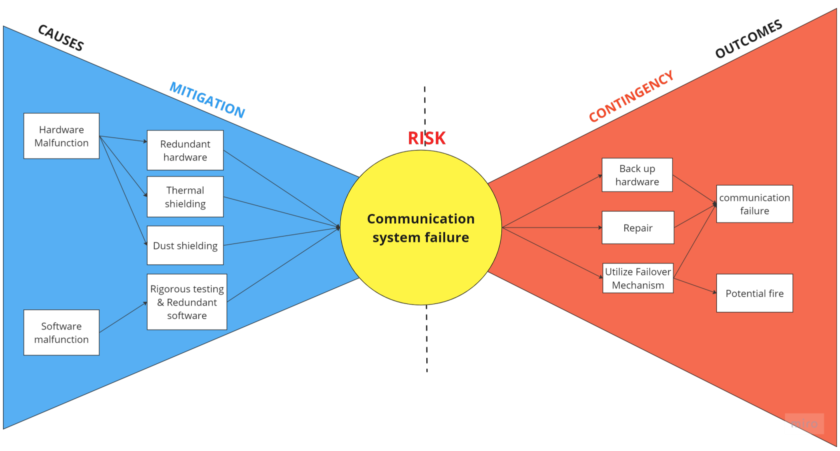

Figure 19.3: Bow Tie Diagram for communication risk

20 | Market Analysis

Author: Ishaan

A detailed market analysis is integral to find a gap in the market, which can be filled by Lunar Industries. First, a summary of what has been done with regards to the market analysis so far will be presented and then costs and potential returns on investment will be explored.

20.1. Current Market Position

Four primary competitors have been identified. These are operating within the same market segment, namely: Toyota-JAXA, Intuitive Machines, Lunar Outpost and Venturi Astrolab. JAXA is making a vehicle that is capable of continuously driving for over 1000 km and 30 days, while the others are making a vehicle for shorter ranges, between 20 km. Therefore, a gap in the market for mid-range vehicles, of around 300 km, was found. In addition to this, it was found that there is a market for a highly-adaptable vehicle, one that is capable of being easily altered to be operational in a wide range of use cases and environments. Lunar Industries aims to address both of these market gaps.

Do note that, from the market analysis no specific performance metrics were determined. The market analysis was mainly conducted to find the gap within which the LTS can be put into. The main requirement that arose from the market analysis is REQ-MIS-3.3.1, which states the LTS shall have autonomous cargo handling capabilities, because all our competitors have autonomous cargo handling system, and also the fact that the LTS needs to be teleoperated, which means it can be used to transport cargo without crew and therefore, it needs to have autonomous cargo handling capabilities.

20.2. Stakeholders

Along with the competitors, stakeholders were also identified, and categorised based on their influence on the design, and interest in the design. This can be seen in the stakeholder map, shown in Figure 20.1

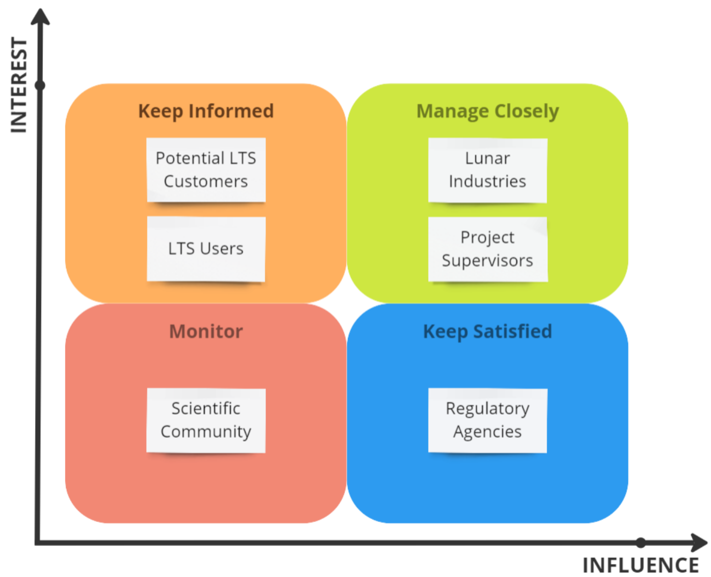

Figure 20.1: Stakeholder Map

20.3. Return on Investment

Based on Chapter 17, it was found that the total cost of development for the LTS would be between 2.28 and 4.57 billion USD, and the cost per LTS would be between 205 million USD and 409 million USD. In order to determine the return on investment, a price must be set for the LTS. It was decided by Lunar Industries that the development cost should be recuperated after 10 LTS have been sold. Using this, the cost per LTS should be set between 435 million and 867 million USD. After which, the return on investment per LTS would be between 228 million USD and 457 million USD.

21 | Concept of Operations

Authors: Lee, Thomas, Max

This chapter will define the concept of operations of the LTS. In Section 21.1, an overview of the most important capabilities of the LTS will be given. Afterwards, how the LTS will be transported to the Moon will be explained in Section 21.2. A general overview of the maintenance process will be given in Section 21.3. Subsequently, in Section 21.4, the end of life strategy of the LTS will be described. Finally the societal impact of the LTS will be described in Section 21.5

21.1. Capabilities

In this section of the concept of operations, an overview of all the capabilities of the LTS will be given. These capabilities are:

- Range of 300 km
- 48 hour nominal mission duration
- EVA capable
- Maximum speed of 10 km/h
- Ability to operate during lunar night
- Drive on a maximum incline of 20°
- Drive on a maximum tilt of 20°
- Level 5 autonomous capability
- Ability to be teleoperated
- Autonomous cargo handling

21.2. Transporting the LTS

In order for the LTS to operate on the Moon, it will first have to be transported there. This will be done using the Starship launch vehicle developed by SpaceX. In order to minimize the impact of the launch loads, which can reach up to 6g in axial acceleration and 3.5g in lateral acceleration [78], the LTS will not be fully assembled while being launched. The stowed configuration before launch has been presented in Section 16.1. Furthermore, a support structure will have to be developed to limit the movement of the LTS during launch.

21.3. Maintenance

After each use of the LTS, inspection will have to be performed on critical components of the LTS, such as the wheels, suspension system, fuel cells and radiation shielding, in order to determine whether maintenance is needed. Next to this inspection, the LTS will have to be cleaned in order to remove the lunar dust which has accumulated during the use of the LTS. If during the inspection damage has been found, the damaged part will have to be replaced. For example, if a radiation shield panel has been damaged by micrometeorites, this panel will be removed by one of the crew members. After the panel has been removed, a replacement panel will be placed in the spot where the damaged panel has been removed. The crew member will also inspect the damaged panel and determine whether it can be repaired, or if it has to be decommissioned.

21.4. End of Life

To be able to decommission and recycle the LTS effectively, an End-of-life (EOL) strategy has to be defined. Once the LTS has reached its EOL, either because it has been damaged beyond repair, or it has become obsolete due to the development of a new version, a detailed assessment will have to be performed. During this assessment the LTS' condition will be analysed, including wear and tear, remaining functionality and potential hazards. The function of this assessment is to determine which components of the LTS, such as shielding panels, fuel cells, batteries, and structural components, can still be of use to other LTSs or other structures on the Moon.

After the assessment of the LTS is done, and all the reusable components have been inventoried, the LTS can be disassembled. First, the wheels will be detached, parts of the wheels will be made of a 3D-printed composite, so these could in potential be melted down and reused for 3D printing. Furthermore, the wheels contain titanium spokes and a titanium core which could also either be reused to produce new wheels or could be processed and used for other structures. The driving and steering motors could possibly be used in different LTSs. After removing the wheels, the suspension system can be removed. The structural rods and springs in the suspension system are made of materials which can be melted down and processed in to different parts. Likewise, the magnets in the suspension system can be processed and eventually used to create new magnets.

The next part which will be removed is the frame, which connects the suspension system to the cabin. This frame also has the hydrogen and oxygen tanks of the electrical power system attached to it. Furthermore, the fuel cells will also be attached to the frame. Both the tanks and the fuel cells can be detached and either reused or processed and decommissioned. Next to the fuel cell, the ammonia to water heat exchanger will be detached. This heat exchanger can be repurposed for other thermal control systems. The frame will be made of aluminium which can be processed and reused for other structures. Now that the wheels, suspension system and frame have been detached, only the cabin and its radiation shielding, and the things that are attached to the outside of the LTS, remain. The radiation shielding of the LTS is made of separate panels, which can be easily removed and recycled. Like the heat exchanger, the radiator panels, which are attached to the top of the LTS, can be repurposed towards other thermal control systems on the Moon. The antennas and all the sensors which are attached to the outside of the cabin could also potentially be recycled. The final system which is attached to the outside of the LTS is the cargo handling system. Both the arm and the cargo storing compartment can be removed and recycled for use in different LTSs.

For the cabin, firstly, the inside will be made completely empty. Inside the cabin there are a lot of tanks, which could either be reused in different LTSs, or the material they are made of could be processed and used to make new tanks. Another heat exchanger can also be found inside the cabin, which can be repurposed for other thermal control systems. The Universal Waste Management System in the cabin can be removed and possibly reused in other LTSs. It is also possible to remove the chairs from the cabin and store these, to be able to replace the chairs in a different LTS. The screens which display necessary information to the crew will also be removed and repurposed for different functions. Furthermore, all the wiring which is used in the LTS can be recycled. Lastly, the life support systems and sensors which can be found inside the cabin will also be removed and reused in different LTSs. All the leftover structural elements of the LTS will be made of aluminium; this material can be processed to be reused to manufacture different parts of lunar infrastructure.

21.5. Societal Impact

The design of the innovative LTS, which integrates level 5 autonomy, modularity and sustainability, will impact the society positively by encouraging technological advancement, fostering economic growth and promoting sustainable engineering. The developed technology for the LTS can be applied to different vehicles for space exploration, enabling safer and more efficient missions. The designed LTS does not only benefit space vehicles but can also enhance vehicles on Earth by introducing technological advancement in autonomous technology, structural& material science and energy production. The innovative technology of the LTS creates demand for specialized expertise in various sectors such as aerospace engineering, software engineering, materials science and sustainable energy. This creates more jobs and generates more funds from government and corporation, resulting in long-term economic growth and technological advancements. Environmentally, the emphasis on recyclability, modularity and adaptability of the designed LTS will promote sustainable practices in all planetary transportation, minimizing the waste and environmental footprint. In conclusion, the designed LTS and its innovative technology does not only lead to technological advancement but it also fosters economic growth and promotes sustainable engineering practices.

21.6. Prospects for In-Situ Manufacturability

Prospectively, the structures subsystem can almost entirely be made using in-situ resources; only minor adhesives and crew suits which are comprised in the structures mass budget cannot. Consequentially, 1200 kg from a total of 1390 kg can thus be made using in-situ aluminium for structures and titanium for bolts.

The powertrain and mobility subsystem too offers some promising prospects for future in-situ resource utilization. First, each wheel, made out of a 10 kg titanium core surrounded by layers of an elastomeric composite totalling 5 kg, can thus partly be made using the titanium available on the Moon. This means that 60 kg of the total 90 kg of the wheels can be made using in-situ resources. Further, the composite layers offer at least the possibility to be manufactured on the Moon using additive manufacturing techniques; however, the composite material would need to be brought from Earth. The suspension system also partly allows for in-situ resource utilization. In fact, its aluminium spring can be manufactured on the Moon using the available aluminium, while its copper electromagnetic coils can also be made using the Moon's copper. This would allow for 15 kg of the total 20 kg of a single suspension to be made using in-situ resources. Hence, overall 150 kg of the total 544 kg of the powertrain and mobility subsystem can prospectively be manufactured using in-situ resources available on the Moon.

Furthermore, the current design of the life support system already makes use of in-situ resources, using regolith in between aluminium plates to shield from radiation. However, one can go even further and produce the aluminium plates using in-situ aluminium. Doing so, would allow for 1500 kg of the total 1771 kg of the life support subsystem to be made using in-situ resources.

Moreover, the aluminium compartments of the cargo handling subsystem can also be manufactured using in-situ resources. This means that 5 kg of the total 21 kg can be manufactured using in-situ resources.

It was explored to produce the tanks used in the life support and electrical power subsystems using in-situ titanium or aluminium instead of the selected composite. However, calculations yielded that this would drastically increase the weight of the tanks. For instance, the four hydrogen tanks from the electrical power subsystem witnessed an increase from 32 kg to over 740 kg in total. If the resource utilization in the production of the tanks is worth the increase in weight to the stakeholders, the tanks could prospectively be made employing in-situ resources. This might, however, require alterations in the design to accommodate for bigger tanks.

In summary, the first design of the LTS will be entirely manufactured on Earth. Nonetheless, the design of the LTS currently allows for future use of in-situ resource utilization of up to almost 50 percent of the total mass, as can be seen in Table 21.1. Exploration performed by the first LTS and increased technological knowledge certainly can increase this conservative estimate. Further, if there is a willingness to increase the total mass of the LTS, even more components could be made using in-situ resources. Of course, it is assumed that all the infrastructure necessary to extract the required resources, to process them and to manufacture them will be available on the Moon.

Table 21.1: Identified potential for future in-situ resource utilization, per subsystem

Subsytem	Mass [kg]	In-Situ mass [kg]	Total Percentage
Structures	1390	1200	20.705%
Cargo Handling	21.2	5	0.086%
Internal Communications	12.6	0	0.000%
User Interface	211	0	0.000%
Guidance, navigation and control	28	0	0.000%
Thermal Control	1178	0	0.045%
Life Support	1771	1500	25.881%
Powertrain & Mobility	544	150	3.106%
Payload	210	0	0.000%
Power Subsystem	429	0	0.863%
Telecommunication	3.09	0	0.000%
TOTAL	**5795.71**	**2855.00**	**49.26%**

22 | Conclusion

Author: Luca

In this report, a preliminary design of the LTS was presented. Lunar exploration is becoming the focus of spaceflight once again, and Lunar Industries is ready to jump into the Lunar Vehicle market with the LTS, a Lunar Transportation Service able to transport a crew of two on the lunar surface, thereby enabling its exploration and colonization.

The engineering team of Lunar Industries delved in great detail into how each subsystem was designed and integrated. An innovative yet reliable powertrain and mobility system will allow the LTS to traverse the lunar surface with ease and agility, without compromising the comfort of the crew. A spacious and reliable pressurized cabin will provide the crew with all necessities, from breathable air to food and water. A well-designed airlock system will not only allow the crew to manage the pressure differences while entering and exiting the LTS, but also to function as a dust removal hub. The LTS is a fully autonomous vehicle: an avant-garde GNC system, equipped with different types of cameras and sensors, will function as the "eyes" of the LTS, allowing it to drive on its own, without any required input form the crew, which will then be able to focus on the exploration missions. Furthermore, the LTS is equipped with an autonomous cargo system. It contains a robotic arm able to manipulate all kinds of cargo. Finally, the entire vehicle is sustainably powered by fuel cells. The integration of, not only all these subsystems, but also additional ones, required the correct functioning of the vehicle, creating a working system, able to accomplish its mission with utmost efficiency and reliability, without compromising sustainability and crew comfortability.

The entire design process is backed by thorough research. Firstly, the lunar environment was analysed. The LTS will operate along with Artemis, NASA's first step in their Moon to Mars campaign. It will therefore operate in the Artemis Exploration Zone, and exploit the infrastructure planned to be put in place by NASA, namely the LunaNet telecommunication service, Basecamp, asn habitable hub and a nuclear fission reactor, needed as energy source during the two-week-long nights. Furthermore, surface conditions within the AEZ were analysed, including altitudes, slopes and temperature ranges. These proved useful in the generation of requirements, which was the following step. In fact, analysing stakeholder needs, functions and the market gap, it was possible to derive system and subsystem requirements, which then formed a base for the design of the LTS' subsystems.

Finally, the proposed preliminary design was analysed. Tools used to design it were verified, and the requirements were validated. Doing this increased the confidence of the team in its proposition. Later, a plan for the next phases of the LTS' development was put in place, ranging from production, testing and finally decommissioning. Additionally, the market potential of the vehicle was analysed, and its ROI was evaluated.

In conclusion, this report described the preliminary design of the LTS, including the design choices made, the research done to back them up and the plan for the next phases. Furthermore, it proved the ability of this design to be not only feasible, but also sustainable, profitable and useful.

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