LEAP

Lunar Exploration Access Point

DSE Spring 2017: Final Report

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Settling on the Moon by 2035

The conceptual design of a lunar habitat for 4 astronauts staying on the Moon for 1 year

DSE Spring 2017: Final Report

by

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This DSE is confidential and cannot be made public until July 5, 2017.

Cover image: Victor Gutgesell

Preface

Group 26 of the Spring Design Synthesis Exercise 2017 proudly presents the final report of LEAP's lunar habitat design. All 10 group members with a passion for space have chosen to work on this assignment out of 25 alternatives. For the past ten weeks, we have worked hard to come up with a feasible design and mission plan for a lunar habitat that will host four astronauts for one year over a period of 10 years. Now we are excited to say that it is in fact possible! If we put our mind to it, we can build a settlement on the Moon. Of course, provided that our Lunar Exploration Access Point (LEAP) is used.
We could not have achieved this without the valuable support of our tutor Prof. dr. ir. Sybrand van der Zwaag and coaches Dr. ir. Ali Elham and Ir. Bas Blank, who have sacrificed their precious time to share their knowledge with us and give us the feedback and support we needed to accomplish this. We would like to thank you all for this input in this project that is so different from the usual DSE project. Furthermore, we would like to thank the OSCC for organising this DSE, teaching us about PMSE and allowing this project to take place in the first place.

Summary

This report is the fourth in a series of reports covering the design of a lunar habitat as part of the "Lunar Exploration Access Point" (or LEAP). LEAP aims at setting up a lunar village in order to gather more information about the Moon itself, but also give more insight about life on other celestial bodies, which can be useful for future space colonisation. The focus of this project lays on the design of the habitat, and determination of necessary auxiliary units which are necessary for the deployment and operation of the lunar village, whereas the actual design of these auxiliary units will be considered in detail in future projects. The focus of this report lies in the detailed design of the habitat. All the other aspects that play a role in the design such as the lunar environment, logistics, technical risks, and a market analysis are also included in the report in order to give a better insight into the driving factors of the final design.

After the Midterm Report, it was found that a synergy of two concepts was the optimal solution, combining one hard shell and two inflatable modules. The hard shell, or simply The Shell, acts as the central module, while the two inflatables, The Hive and The Nest, respectively serve as a storage and a sleeping facility. The detailed design of the three modules covers in depth all the subsystems ranging from very technical calculations such as for the power subsystem to more human requirements analysis needed for the interior design.
LEAP covers a large time span after this Design Synthesis Exercise. More research in specific fields needs to be conducted in order for the mission to be a success. After the DSE, the detailed design of the mission will continue, after which the construction phase can begin. The mission is expected to start in 2030 with the launch of Small Step One. The entire mission will require a total of 23 launches, including the resupply and crew change missions, which will be executed with a variety of launch modules, each specially selected for the mission phase depending on the payload weight, aiming at minimising the launch costs. The total cost is estimated at 58.3B EUR including the resupply missions, but also the research and development of all the auxiliary units and exploration vehicles needed to set up the lunar habitat. Finally the first manned mission, and thus making the habitat fully operational is expected in 2035.

Lists of Symbols and Abbreviations

List of Abbreviations

A5V	Angara 5V
AAA	Avionics Air Assembly
AC	Alternating Current
ADU	Automated Driving Unit
ARPCS	Atmospheric Revitalising and Pressurisation Control System
ATH-LETE	All-Terrain Hex-Limbed Extra-Terrestrial Explorer
BEAM	Bigelow Expandable Activity Module
BER	Bit Error Rate
CATIA	Computer Aided Three-dimension Interactive Application
CBS	Cost Breakdown Structure
CCAA	Common Cabin Air Assembly
CDHS	Communication and Data Handling System
CDRA	Carbon Dioxide Removal Assembly
CERN	Conseil Européen pour la Recherche Nucléaire (European Organization for Nuclear Research)
CP	Cold Plates
DC	Direct Current
DDCU	DC-to-DC Conversion Units
DSE	Design and Synthesis Exersice
EATCS	External Active Thermal Control System
EC	Electrolizer Cell
EDRS	European Data Relay System
ESA	European Space Agency
ESP	Emittence Solar Particle
FBS	Functional Breakdown Structure
FEM	Finite Element Method
FFD	Functional Flow Diagram
GCR	Galactic Cosmic Radiation
HEPA	High Efficiency Particle Atmosphere
HZE	High atomic number and energy
IATCS	Internal Active Thermal Control System
IHE	Interface Heat Exchanger
IMV	Intermodule Ventilation
ISDS	International Standard Docking System
ISO	International Organization for Standardisation
ISS	International Space Station
LADEE	Lunar Atmosphere and Dust Environment Explorer
LAF-PMS	Lunar Air Filtration with Permanent Magnet System
LEAP	Lunar Exploration Access Point
LED	Light-Emitting Diode
LEO	Low-Earth orbit
LISA	Lightweight Inflatable Structural Airlock
LLCD	Lunar Laser Communication Demonstration
LLGT	Lunar Lasercom Ground Station
LLST	Lunar Lasercom Space Terminal
LOLA	Lunar Orbiter Laser Altimeter
LRO	Lunar Reconnaissance Orbiter
LSR	Lunar Sintering Rover
MBSU	Main Bus Switching Unit
MCA	Major Constituent Analyser
MLI	Mulit-Layer Insulation
MLR	Multifunctional Lunar Rover
MMOD	MicroMeteorite and Orbital Debris
MPPT	Maximum Peak Power Tracking
MTB	Multishock thermal blanket
NASA	National Aeronautics and Space Administration
OGA	Oxygen Generation Assembly
OPALS	Optical PAyload for Lasercomm Science
OSCC	Organising Committee
PCA	Pressure Control Assembly
PCU	Power Control Unit
PEEK	Polyether Ether Ketone
PEM	Proton Exchange Membrane
PEN	Polyethylene Naphthalate
PET	Polyethylene Terephthalate
PMSE	Project Management and System Engineering
PrCU	Pressure Control Unit
PSY-CHIC	Prediction of Solar particle Yields for Characterising Integrated Circuits
RAMS	Reliability, Availability, Maintenance and Safety
RF	Radio Frequency
SBR	Scouting and Beaconing Rovers
SCR	Solar Cosmic Radiation
SEV	Space Exploration Vehicle
SLS	Space Launch System

SPENVIS	SPace ENVironment Information System
SPEs	Solar Particle Events
TBD	To Be Decided
TCCS	Trace Contaminant Control Sub-system
TCU	Thermal Control Unit
TLI	Trans Lunar Injection
TRL	Technical Readiness Level
UHF	Ultra High Frequency
VALID	Verifiable, Achievable, Logical, Integral and Definitive
WS	Whipple Shield

List of Symbols

α	deflection angle	$[deg]$
α_a	absorptance	$[-]$
β	angle between coils	$[deg]$
δd_C	critical projectile diameter with MLI	$[cm]$
ΔT_{pl}	temperature difference between in- and outside of a plate	$[K]$
$\dot{m}_{H_2}$	hydrogen mass flow	$[kg/s]$
$\dot{m}_{H_2O}$	water mass flow	$[kg/s]$
$\dot{m}_{O_2}$	oxygen mass flow	$[kg/s]$
ϵ	emmisivity	$[-]$
η_{cell}	solar cell efficiency	$[-]$
η_{EC}	efficiency of the electrolyser cells	$[-]$
η_{fc}	fuel cell efficiency	$[-]$
$\eta_{inv_{fc}}$	invertor efficiency	$[-]$
$\eta_{inv_{Sol}}$	degradation of photovoltaic cells	$[-]$
ρ_b	bumper density	$[g/cm^3]$
ρ_p	projectile density	$[g/cm^3]$
ρ_w	rear wall density	$[g/cm^3]$
ρ_{reg}	regolith density	$[g/cm^3]$
ρ_s	resensitivity	$[\Omega * m]$
σ_B	Boltzmann constant	$[m^2 kg s^{-2} K^{-1}]$
σ_{iw}	stress induced due to weight of inflatable	$[Pa]$
σ_n	normal stress	$[Pa]$
σ_{S-B}	Stefan–Boltzmann constant	$[W/(m^2 * K^4)]$
σ_w	rear wall yield stress	$[ksi]$
σ_h	hoop stress	$[Pa]$
σ_y	yield stress	$[Pa]$
θ	impact angle from target normal; θ = 0 deg is normal to the target.	$[deg]$
A_{ex}	area exposed to heat flux	$[m^2]$
$A_{solarcells}$	solar cells area	$[m^2]$
A_l	leg area	$[m^2]$
A_p	projected area	$[m^2]$
A_r	radiating area	$[m^2]$
B	magnetic field strength	$[T]$
B_r	magnetic field strength in radial direction	$[T]$
B_x	magnetic field strength in axial direction	$[T]$
BHN	Brinell Hardness factor	$[-]$
c	speed of light	$[m/s]$
c_b	front bumper constant	$[-]$
C_{cap}	circumference cap	$[m]$
C_{Cu}	copper heat capacity	$[J/K]$
C_{H_2}	specific heat hydrogen	$[kJ/(kgK)]$
C_{H_2O}	specific heat water	$[kJ/(kgK)]$
C_L	low velocity coefficient	$[cm^3/g]$
c_{N-K}	intermediate bumper constant	$[-]$
C_{O_2}	specific heat oxygen	$[kJ/(kgK)]$
c_o	rear wall constant	$[-]$
c_w	rear wall coefficient	$[s/km]$
c_w	rear bumper constant	$[-]$
C_t	speed of sound within a material	$[km/s]$
$c1, c2$	heat capacity	$[J/K]$
d_c	critical projectile diameter	$[cm]$
d_{des}	design projectile diameter	$[cm]$
d_p	projectile diameter	$[cm]$
d_l	diameter leg	$[m]$
dp	momentum increment	$[kgm/s]$
E	particle energy	$[J]$
E_{stored}	Energy stored by fuel cells	$[kWh]$
E_y	Young's modulus	$[Pa]$
F	packing factor	$[-]$
F_{crit}	critical buckling load	$[N]$
F_L	magnetic force	$[N]$
f_{MLI}	protective capability factor for MLI	$[-]$
F_n	normal force per leg	$[N]$
G	G-factor	$[-]$
g_M	gravitational acceleration on the Moon	$[m/s^2]$
h_{cap}	height cap	$[m]$
h_{UHF}	height UHF antenna	$[m]$
I	current	$[A]$
I_l	second moment of inertia of a leg	$[m^4]$
k	thermal conductivity	$[W/(mK)]$
K_{H-SW}	high-velocity coefficient for stuffed whipple shield	$[km^{1/3} s^{-1/3}]$
K_{Hi-SW}	high-intermediate velocity coefficient for stuffed whipple shield	$[-]$
K_{L-SW}	low-velocity coefficient for stuffed whipple shield	$[g^{0.5} km^{2/3} cm^{-3/2} s^{-2/3}]$

k_l	thermal conductivity of a layer	$[W/(wK)]$
K_{Li-SW}	low-intermediate velocity coefficient for stuffed whipple shield	$[g^{0.5}km^{2/3}$ $m^{-3/2}]$
k_{MLI}	MLI constant	$[cm^3/g]$
k_{pl}	thermal conductivity of a plate	$[W/(wK)]$
k_{tot}	total thermal conductivity	$[W/(wK)]$
l_{cap}	total coil length	$[m]$
L_{tot}	total length of inflatable	$[m]$
L_{UHF}	side length of landing square	$[km]$
m_{arrays}	solar array mass	$[kg]$
$m_{b-total}$	total bumper areal density	$[g/cm^2]$
m_{cells}	solar cell mass	$[kg]$
$m_{cryotank}$	cryotank mass	$[kg]$
m_{Cu}	copper mass	$[kg]$
$m_{FC\&EC}$	total mass of fuell cells and electrolyser cells	$[kg]$
m_{foam}	areal density of foam	$[g/cm^2]$
m_{H_2}	hydrogen mass	$[kg]$
$m_{intermediate}$	intermediate tanks mass	$[kg]$
m_{MLI}	areal density of insulation blanket	$[g/cm^2]$
m_{MLI}	areal density of MLI layer	$[g/cm^2]$
$m_{Nextel-Kevlar}$	mass ratio Nextel-Kevlar	$[-]$
m_{O_2}	oxygen mass	$[kg]$
M_p	projectile mass	$[g]$
m_{panels}	solar panaal mass	$[kg]$
m_{tank}	tank mass	$[kg]$
m_{water}	water mass	$[kg]$
m_0	invariant mass	$[kg]$
m_b	areal density of front bumper	$[g/cm^2]$
M_i	mass inflatable	$[kg]$
m_w	areal density rear wall	$[g/cm^2]$
n	number of coils	$[-]$
n_{H_2}	amount of hydrogen	$[mol]$
n_{O_2}	amount of oxygen	$[mol]$
P	power required per coil	$[W]$
$P/infty$	penetration depth	$[km]$
p_{H_2}	hydrogen pressure	$[Pa]$
P_{H_2}	dissipated thermal power for hydrogen	$[kW]$
P_{H_2O}	dissipated thermal power for water	$[kW]$
$P_{internal}$	internal pressure	$[Pa]$
$P_{max_{EC}}$	maximum power electrolyser cells	$[kW]$
P_{night}	required power at night	$[kW]$
p_{O_2}	oxygen pressure	$[Pa]$
P_{O_2}	dissipated thermal power for oxygen	$[kW]$
P_{peak}	peak power	$[kW]$
P_{solar}	needed solar power	$[kW]$
P_{tot}	total required coil power	$[W]$
p_1	initial particle momentum	$[kgm/s]$
Q	heat	$[W]$
q	particle charge	$[C]$
R	shielding reliability	$[-]$
r_{cyl}	radius inner cylinder	$[m]$
r_{in}	inner radius	$[m]$
R_{Moon}	Moon radius	$[km]$
r_1	coil inner radius	$[m]$
r_2	coil outer radius	$[m]$
R_i	ideal gas constant	$[J/Kmol]$
r_l	radius leg	$[m]$
S_{b-w}	internal offset between the front bumper and rear wall	$[cm]$
S_{MLI}	gap distance from back of bumper to front of MLI	$[cm]$
S_R	solar irradiance	$[W/m^2]$
SF	safety factor	$[-]$
T	temperature	$[K]$
t	proton travel time	$[s]$
t_b	thickness front bumper	$[cm]$
T_{cryo}	temperature of the cryogenic storage	$[K]$
t_{cyl}	thickness cylinder	$[m]$
T_{FC}	fuel cell temperature	$[K]$
T_{H_2}	hydrogen temperature	$[K]$
T_{in}	incoming gas temperature	$[K]$
t_l	thickness of a layer	$[m]$
t_{min}	minimal thickness	$[mm]$
T_{O_2}	oxygen temperature	$[K]$
t_{pl}	thickness of a plate	$[m]$
t_{res}	restrainer thickness	$[mm]$
T_{room}	ambient room temperature	$[K]$
t_{sp}	thickness sphere	$[m]$
t_{tank}	tank thickness	$[m]$
t_{tot}	total thickness of layers	$[m]$
t_w	rear wall thickness	$[g/cm^3]$
t_{wall}	habitat wall thickness	$[m]$
t_l	thickness of a leg	$[mm]$
u_0	permeability	$[H/m]$
V	projectile velocity	$[km/s]$
V_{fast}	speed of high velocity projectiles	$[km/s]$
V_{H_2}	hydrogen volume	$[m^3]$
V_{med}	speed of medium velocity projectiles	$[km/s]$
V_n	normal component of projectile velocity	$[km/s]$
V_{O_2}	oxygen volume	$[m^3]$
V_{slow}	speed of low velocity projectiles	$[km/s]$
v_1	initial particle velocity	$[m/s]$

Contents

1
Introduction

For centuries, mankind has had the idea of living on other planets. This idea might very soon turn into reality. The advancement of technology and science has enabled mankind not only to survive in space, like on the International Space Station, but also travel to our celestial neighbours. The Moon is an object of interest to gain a first experience of living on another celestial body. The Moon's relatively close distance, as well as the well-known conditions, make it an ideal target for a first permanent extraterrestrial base. The mission of this project is to explore the option of designing a semi-permanent habitat for such a base on the Moon. This lunar habitat will host 4 astronauts for the duration of 1 year and operate reliably for ten years. Furthermore, the habitat is designed to be part of a larger lunar village with many auxiliary buildings. Thus, the objective of the project is to design and study the feasibility of a habitat mission, which incorporates the transport and also the assembly of the modules on-site.

This report is the third and final issue of the series featuring the design of a lunar habitat and its mission operations. It concludes the design process of the lunar habitat itself with a detailed mass, cost and power estimation of all subsystem needed to provide a safe environment for the astronauts. Further, the mission operations are defined to ensure long-term mission success. A detailed functional and logistic plan regarding the set-up, deployment and operation of the habitat is created to pinpoint the resources needed and to be able to create a mission timeline. All design decisions regarding the habitat and its mission operations are taken keeping sustainability in mind. Sustainability is an important criterion for success and does not only include the environmental factor. In order to ensure economic sustainability, a market and cost analysis gives insight into the finances of the mission.

The presentation of the design is initiated with a description of the challenges of lunar settlement and the LEAP habitat design in in chapter 2. The design will be explored in more detail in the following chapters which are treating the subsystems needed to provide a safe environment for the astronauts. In chapter 3, chapter 4 and chapter 5 in particular, the subsystems of the habitat which create a life sustaining environment are described: the environmental protection subsystem, the structure and the bioastronautic subsystem, respectively. The communication subsystem is described in chapter 6 while the interior design can be found in chapter 7. In order to run the habitat, a means of power generation is needed which is described in chapter 8. To conclude the habitat design and for convenience, a subsystem factsheet in chapter 9 contains a summary of all subsystems design parameter. The mission operations of the habitat are initiated with the transportation and lander configuration in chapter 10. Further, chapter 11 contains a detailed functional analysis and the logistics approach of the LEAP missions. In order to ensure mission success, a technical risk assessment is performed and documented in chapter 12, a technical verification and validation process is described in chapter 13 and a requirements compliance matrix is presented in chapter 14. Further, the LEAP mission and its market analysis can be found in chapter 15 and chapter 16, respectively. Finally, the sustainability and development strategy is given in chapter 17 and concluded remarks are documented in chapter 18.

2
Executive Summary

In 1969, Apollo 11 proved that mankind can send astronauts to the Moon and safely bring them back to Earth. The International Space Station (ISS) proves that it is possible to sustain life in space for extended durations. The advancement of technology and mankind's inner drive to explore bring us to new frontiers and enable us to overcome new challenges. The challenge of today is to settle outside of our planet's protection and see if the achievements of Apollo and the ISS can be combined to establish a semi-permanent outpost on the Moon.

LEAP's mission:
Explore the option of living on the Moon by designing a semi-permanent lunar habitat operating for at least ten years, as part of a lunar base.

Our objective:
Design and study the feasibility of a lunar habitat, incorporating transport and assembly, hosting four astronauts for one year, by ten students in twelve weeks.

The Moon is a desolate place and holds a hostile environment for man and machine. If humans are to stay on the Moon for a year the environment and the challenges of the travel have to be understood and accounted for. This chapter presents an analysis of exactly these challenges.

The environment of the Moon differs vastly from the environment that we are used to on Earth. Most significantly, the lack of an atmosphere complicates operations, as the astronauts permanently have to be held in an artificial atmosphere. Similar to other manned space missions, this calls for a means of creating an internal pressure and sustaining a breathable, non-toxic air mixture using machines or organisms.

The absence of a protective bubble around the Moon poses further hazards than the mere problem of being able to breathe. Whereas Earth's atmosphere burns up most meteorites and objects that would otherwise impact the surface, meteorites that are on collision course with the Moon, reach its surface with incredible velocities. There is a constant influx of micrometeorites, meteorites that have a diameter of up to 1cm, that perforate and corrode whatever is on the lunar surface. With velocities of up to 45km/s the energy that such a meteorite can have may be lethal to an astronaut and terminal for a mission. Proper shielding needs to be employed to mitigate the risk of a fatality or a mission abort.

Even on a smaller scale, particles impact and penetrate objects on the lunar surface. Radiation, for example, is energy travelling through space as high-speed atomic particles and electromagnetic waves. Ionising radiation can damage material and living tissue. In long terms, it can be the cause of cancer and cerebral degeneration, and in short terms, it can cause radiation sickness, a condition which normally ends with death. Radiation is particularly difficult to shield against, as the particles can simply fly through the empty space in between atoms. Outside the protection of Earth's atmosphere and magnetic field, astronauts and the habitat will constantly be exposed to dangerous levels of radiation.

The lunar surface is mainly a desert of regolith, or lunar dust. Any kind of movement can stir up this dust that infiltrates machines or ends up in the respiratory system of an astronaut. The fine particles can harm systems, corrode material, cause breathing problems, trigger asthma and damage lung tissue. Reports of former Moon missions confirm that dust was brought into the vehicle after a Moon walk [1]. Regular cleaning may not be effective as some particles are too small to be seen. Additionally, dust was reported to be a hazard even on the lunar surface. Astronauts have reported that dust laying on the firm ground of the Moon poses an injury hazard as it is very slippery. The coarse underground is also not homogeneous in density. These differences can be seen in the form of soft spots, spots where machines and astronauts can sink in.

Finally, the unhindered influx of the Sun during the day and the lack of an atmosphere cause vast temperature changes of objects on the Moon. A significant amount of energy has to be spent to keep thermal conditions inside the habitat liveable. The Moon itself also radiates heat and the Sun reflects from the Earth to the lunar surface. This makes for a complex thermal environment that has to be accounted for when designing. All of the above-listed aspects are a shortened version of the findings of a detailed literature study which was performed for the Midterm Report of this series [2].

By now it is clear that a proposed lunar habitat has many key functions and utilises a variety of systems to safeguard the life of its inhabitants. The larger such a habitat becomes, the heavier these systems are and the more power they consume. In a one-year mission, 4 human beings require a significant area to live, to not suffer from adverse psychological effects and to perform their tasks in an effective manner. Without quantifying one can imagine that there is probably no transportation vehicle to send a singular entity which can fulfil all of the demands for such a habitat. Further, the settlement of the Moon is a long term endeavour and features a variety of complex tasks and operations. This calls for a thorough mission planning and poses a huge logistical challenge to the people involved. The utilisation of different launch vehicles, and production and research facilities, as well as their time-wise context, are aspects which cannot be neglected in the engineering process of the habitat. This is why part of this report also focuses on conceptually solving the issues that arise in these aspects.

To address all these challenges the LEAP habitat is designed to be a modular structure, constructed at near the landing site of Apollo 11 and brought to the Moon in several stages. Several mission elements are featured to construct an operational habitat on the Moon, all of them will be explained in detail in the coming chapters of this report. An artist's impression of the LEAP habitat can be found in Figure 2.1. Figure 2.2 shows a schematic functional diagram of the habitat and its auxiliary units. The habitat itself consists of 3 modules called The Shell, The Nest and The Hive. The Shell is a high-tech aluminium structure which acts as a central node to The Nest and The Hive. These 2 modules are inflatable structures, made from new-age space-grade fabrics. The combination of these 2 types of structures is chosen to preserve their individual strengths. The protection of a rigid hard-shell structure and the space-to-weight ratio of the inflatables make an ideal combination for the mission at hand.

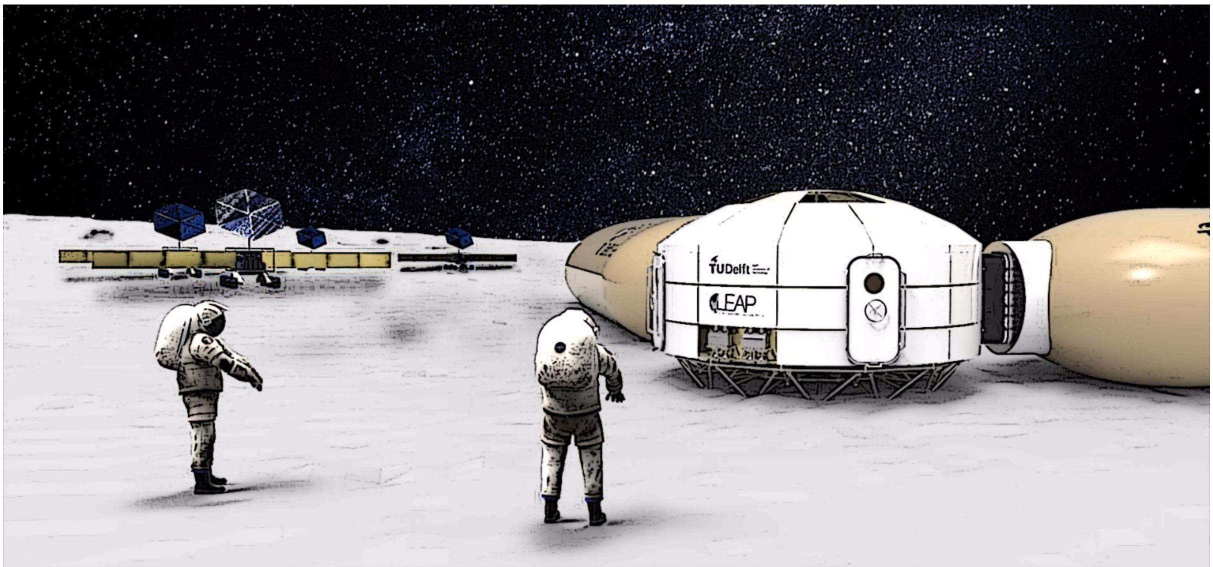


Figure 2.1: LEAP artist's impression.

The habitat modules have very similar functions, yet they differ in the specific implementation of subsystem configuration. Nonetheless, the set of subsystems for both is the same and the habitat overall consists of 5 different subsystems: the structural subsystem, the environmental protection subsystem, the power subsystem, the bioastronautic subsystem and the communication and data handling subsystem. The mass distributions of the subsystems is shown in Figure 2.4. As the name implies, the structural subsystem provides the habitat with space and structural integrity. The environmental protection system shields the inhabitants of the habitat from the harsh environment on the Moon. The power subsystem mainly distributes the power from the auxiliary modules amongst the habitat subsystems. Bioastronautics is the subsystem which addresses the needs of the human inhabitants, both physically and psychologically. The communication and data handling subsystem ensures contact with Earth, auxiliary modules and expeditions on the Moon, and a smooth data process inside the habitat. When combining the three modules and their subsystems one obtains the full habitat. The total habitat mass budget is shown in Figure 2.4 and a detailed listing is shown in Table 2.1.

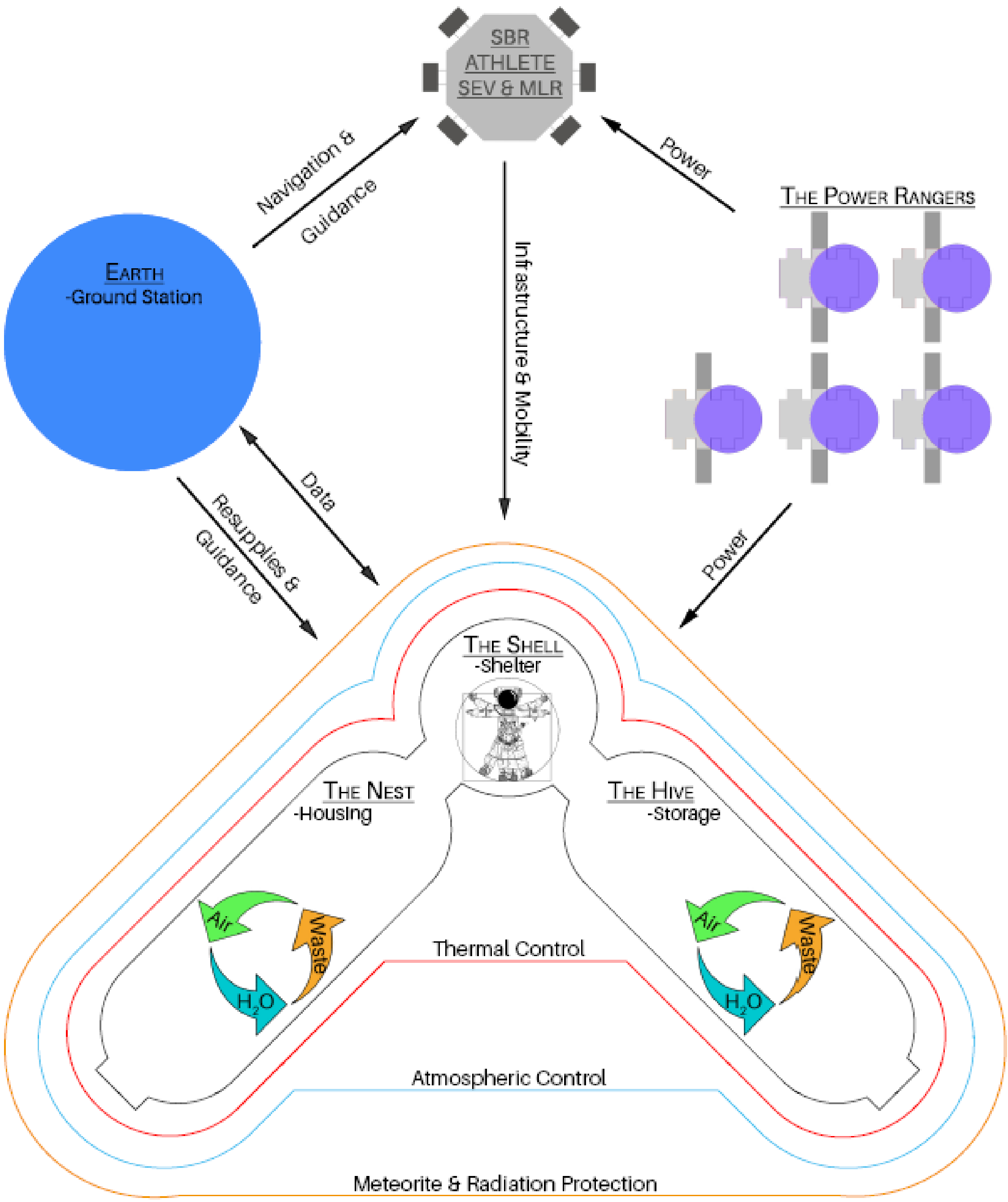


Figure 2.2: Schematic functional diagram of LEAP Habitat mission operation.

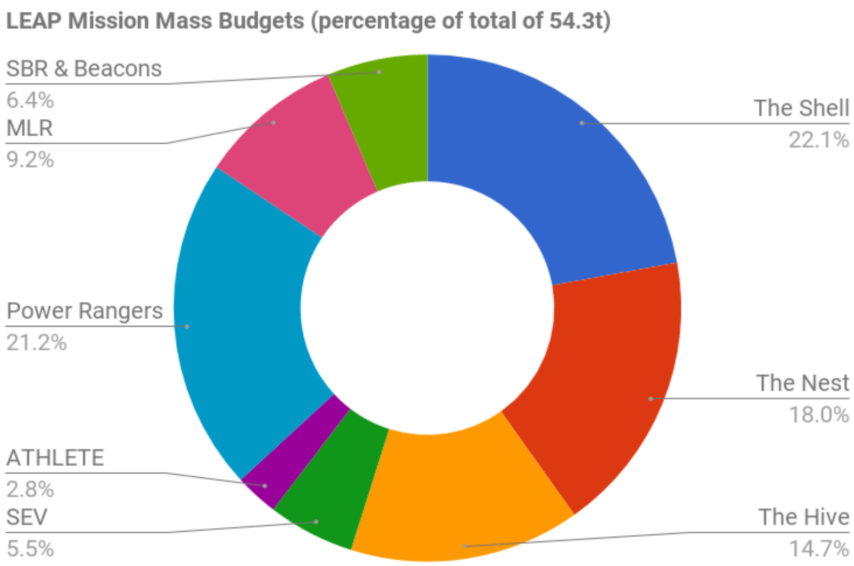


Figure 2.3: LEAP mass budget.

The auxiliary systems of the mission form another very important part of the habitat design. The key to any operation on the Moon are the 5 Power Rangers, autonomous rovers with the ability to produce and store solar energy during the lunar days and to release the energy during the lunar night. The Power Rangers, however, are only one type of the 5 rover types LEAP envisions to send. The SEV (Space Exploration Vehicle, a NASA concept) transports astronauts over large distances to explore the Moon. The Multipurpose Lunar Rover (MLR) is a means of transportation over short distances for men and material. It is designed to carry heavy loads and can thus carry the inflatable modules to The Shell for assembly. The ATHLETE (All-Terrain Hex-Legged Extra-Terrestrial Explorer, a NASA concept), on the other hand, is an infrastructure rover sent to the Moon to sinter the lunar dust into a foundation for the habitat. The SBRs (Scouting and Beaconing Rovers) are the first rovers to be sent to the Moon. It is their function to search a suitable construction site for the habitat and mark the landing zone with beacons.

In total, the LEAP habitat will feature 5 initial missions, before the arrival of the first astronaut crew and with that, it will send a total of 54.3 tonnes to the Moon. A general listing of all the mass division can be found in Figure 2.3. In Figure 2.5 a similar diagram about the peak power budget of the mission is shown.

LEAP's first mission, Small Step 0 is planned for 2020 and sends a reconnaissance orbiter to the Moon to collect data for a further detailed design of the lunar habitat and its auxiliary modules. Small Step 1 follows in 2029, with four scouting and beaconing rovers (SBRs) which map and research the landing site of the habitat. The rovers shall autonomously select a site and place beacons for future landings on a 4-4km area. In 2032, Small Step 2 sends the first payload with a Space Launch System (SLS) to the Moon. This mission will make way for the landing of the habitat by sending the machines needed to create the infrastructure on the Moon. One ATHLETE robot starts sintering lunar soil to build the foundation for the habitat to stand on. The MLR, a rover to level the ground and move payloads is sent with one SEV and two Power Rangers, autonomous power trucks to provide the power for all vehicles and the habitat in the future. Giant Leap 1, finally, delivers the central module of the habitat to the Moon in 2034; a high-tech aluminium structure which acts as a central node for the habitat and its systems. In the same year, Giant Leap 2 delivers two inflatable structures which are first docked to the central module and later inflated to generate sufficient living space and protect the astronauts against the hostile lunar environment. When this process is concluded, the first crew of astronauts is expected to arrive in 2035. On the final page of this chapter a visualisation of the launch schedule is shown for the first 5 initial missions. Nonetheless, the total mission cost 58.3B EUR already entails the operations of the habitat too. During these operations annual resupplies ensure that the astronauts have everything they need and an annual exchange of the astronaut crew ensures a healthy continuation of the mission. After

10 years, the end of the mission is reached and a decision ahs to be made on whether to decommission the habitat or continue operations.
Lastly, sustainability is a key driver in Project LEAP. With a total estimated mission cost of around 58.3B EUR, the placement of humans on another celestial body and the environmental impact of rocketry, the project bares great responsibilities not only to the Earth and humanity, but also the Moon and its future inhabitants.

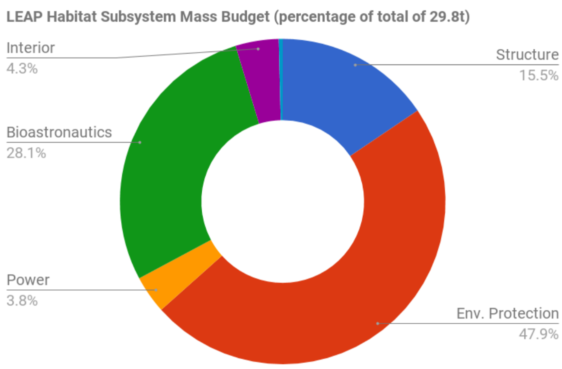


Figure 2.4: LEAP habitat mass budgets.

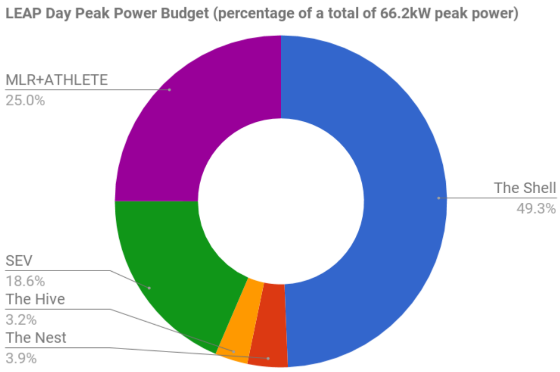
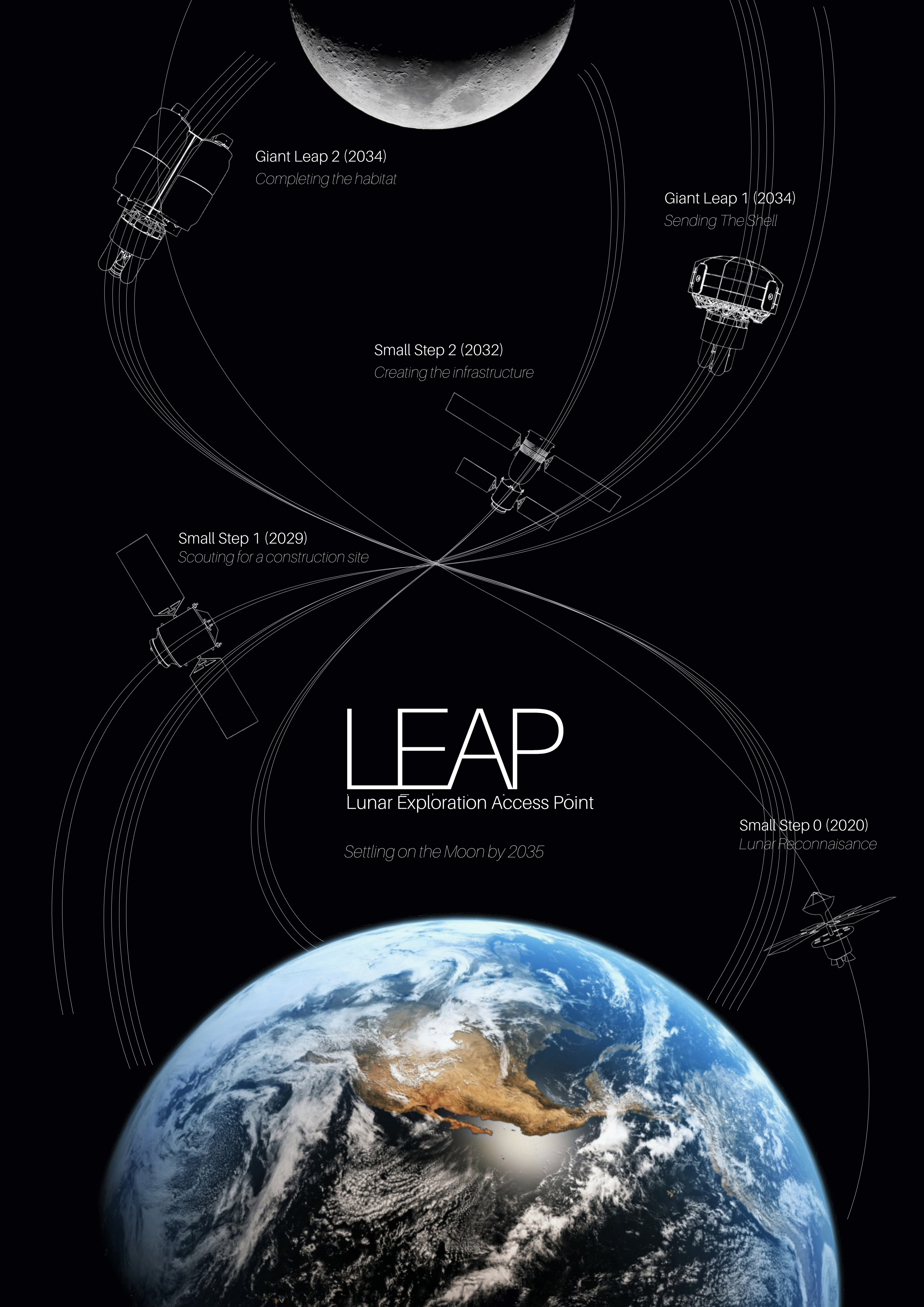


Figure 2.5: LEAP peak power budget during the day.

Table 2.1: Overall mass budget of the modules, in kg

Subsystem	**Part**	**The Shell**	**The Nest**	**The Hive**
Environmental	Atmospheric control	197.19	398.99	398.99
	N2 & O2	135.19	218.17	218.17
	Whipple Shield	632.45	-	-
	MTB	-	3761.47	3761.47
	Active radiation protection	2579	-	-
	Passive thermal control	380.66	-	-
	Active thermal control	519.1	524	543
Structure	Load bearing	2317	207.34	207.34
	Atmospheric layer	-	468	468
	Support (Pins, tubes)	17.44	-	-
	Floor	200	375	375
Bioastronautics	Airlocks	1262.2	631.1	631.1
	Space Suit	75	-	-
	Water & Waste Management	797	797	-
	Tank Mass	109.9	193.4	75.3
	H2O	615.4	1436.0	357
	Food supply	1237.9	-	-
	Tool Supply	170	-	-
Communication	LLST incl. backup	61.4	-	-
	Laptops and data bus	25	-	-
	Wifi router	0.2	0.2	0.2
	Cables	20	5	5
Interior	Walls	21.93	85.47	36.95
	Furniture	152.71	386.1	596
Power	Contingency Power System	492.36	315.55	315.55
Total		12019	9803	7989


Giant Leap 2 (2034)
Completing the habitat
Giant Leap 1 (2034)
Sending The Shell
Small Step 2 (2032)
Creating the infrastructure
Small Step 1 (2029)
Scouting for a construction site
LEAP
Lunar Exploration Access Point
Settling on the Moon by 2035
Small Step 0 (2020)
Lunar Reconnaisance

3
Environmental Protection Subsystem Design

To survive on the Moon, astronauts have to be protected against the hostile environment. As explained in chapter 2, the Moon has a fundamentally different environment compared to Earth. These differences are counteracted by a subsystem of the habitat. This subsystem is the environmental protection subsystem which is the subject of this design chapter. The environmental protection consists of meteorite protection, thermal control, radiation protection and atmospheric control. First some general information on the thermal control and radiation is provided. Secondly, in section 3.1, the detailed design of the passive environmental protection for The Shell is given. Next, in section 3.2, the design of the passive environmental protection of the inflatables is explained. Finally, in section 3.3 the active environmental control is designed.

Thermal Control

At first, this chapter will deal with the large differences in temperature. For the astronauts, it is comfortable if the temperature inside the habitat is between 18 °C and 26 °C. Therefore, an inside temperature of around 295 K needs to be maintained. The thermal control of the inflatables and The Shell are treated separately, because of their autonomous and modular design.
The thermal control consists of a passive and an active thermal control. The passive thermal control system will be insulation to prevent heat from entering the habitat during the lunar day and leaving during the lunar night. The active thermal control will then transport the excess heat to the outside or perform extra heating. This means that if the insulation is optimised, less active control is needed. Active thermal control is relatively heavy. Therefore, passive thermal control is favourable to use.

Besides the incoming heat flux, heat is generated in the habitat by the systems and by the astronauts. The heat generated by the systems will differ based on their performance. Most of the systems will be connected to the active thermal control. Active thermal control is designed to cool when all systems are turned on. Systems which are nominally switched on will be taken into account when deciding on the passive and active thermal control configuration. For example, when a system continuously dissipates heat inside the habitat, less heating and insulation will be needed.

The heat generated by the astronauts is rather stable, although it depends on where all the astronauts are situated. At maximum, the active thermal control should be able to cool if the astronauts are all together in one module. For the design of the thermal heating system, on the other hand, the astronauts are not taken into account. This is because the astronauts can be out or in another module. It is then not possible to rely on the heat coming from the astronauts. Thus, for the maximum heating requirement the astronauts are assumed not there.
Assuming the astronauts use 2975kcal on average [3] per day, the heat dissipating from their body can be calculated. One kcal is 4148 Joule. To get to heat in Watt the amount of Joules has to be divided by time. This results in a heat dissipation of 144W per astronaut. The thermal cooling system should thus be able to cool an additional 576W per module.
On the heat produce of systems inside the habitat will be elaborated later in section 3.3.

Radiation Protection

On the Moon, astronauts are constantly subjected to unhealthy levels of solar cosmic radiation (SCR) and galactic cosmic radiation (GCR). Both consist mainly of subatomic particles travelling close to the speed of light. While GCR originates from sources outside the solar system, SCR is radiation caused by the Sun. SCR has relatively low energy compared to the GCR, yet SCR flux is higher, especially during solar particle event (SPEs). Shielding against primary particle radiation can be done by designing thick walls. However, the main issue with radiation shielding originates from secondary radiation. Secondary radiation is caused by the particles interacting with the atoms of the shielding material. Thus, the more shielding material, the more secondary radiation will occur. The challenge is to find a balance between

the primary and secondary radiation such that the total ionising dose is below the allowable yearly limit. Radiation protection is split in passive and active protection. Whereas passive protection features shielding and consumes no power, active protection employs sensors and a magnetic field generator and require large amounts of power if in full operation. The annual maximum dose equivalent for the astronauts is 0.5Sv. A conversion factor of 6 is estimated from ionising dose to dose equivalent [4].
Applying this conversion factor results in a maximum ionising dose of 8.3rad per year. To keep track of the absorbed dose, the astronauts will carry dosimeters which will record the accumulated dose. Furthermore, solar particle events can be predicted some hours before reaching the Moon. In case of such an event, the astronauts will be instructed to seek shelter in The Shell.

3.1. The Shell - Passive Environmental Protection

The Shell is designed as the safe room of the habitat. It provides protection during solar particle events. As The Shell will be the first module to be sent to the Moon, it needs to function completely autonomously and automated. This section shows the design process of its passive environmental protection system. The section is structured from the outside layer to the inside layer of The Shell. First the shielding against meteorites is explained, then the passive thermal control is discussed and finally, a means of passive radiation protection is proposed.

Meteorite Protection

As The Shell is acting as the safe room of the habitat, it is also designed to withstand meteorite impacts without the help of any regolith. NASA's handbook for designing MicroMeteorite and Orbital Debris (MMOD) protection proved to be particularly useful for the entire design process [5].

A design concept had to be selected to initiate the optimisation process. Considering the Design Options Tree, a few options have remained for The Shell; the monolithic wall, the Whipple shield and the stuffed Whipple shield [2]. After consulting [5], another interesting option arose, that being the multi-shock shield. A multi-shock shield consists of an aluminium or Kevlar rear wall, with a multitude of ceramic fibre layers spaced out in front of it.

- A monolithic wall needs to be very thick and heavy to withstand hypersonic impacts.
- All remaining shields are effective by inducing a shock inside the projectile, which causes it to shatter and spread out. The most critical impact condition for these shields is actually for ballistic velocities. The occurrence of primary impacts with ballistic velocities is very low, as can be seen in Figure 3.1. However, secondary impacts can be at ballistic velocities due to the Moon's escape velocity of 2.38km/s [1].
- The stuffed Whipple shield is significantly superior to the original Whipple shield while increasing the structural complexity relatively little.
- The multi-shock shield is more effective than the stuffed Whipple shield at stopping hypersonic particles, yet its structure is relatively complex.

Because of these findings, it has been decided to design a stuffed Whipple shield for The Shell. Two sets of equations are given in [5]. The first set contains design equations, which provide an initial thickness for every layer; a starting point for the design. A key input for these equations is the size of the projectile. Afterwards, the so-called performance equations can be used. These calculate the projectile size for which the shield would fail for different velocity regimes, given the shield's dimensions. This allows for an iterative process that stops when the design projectile size is equal to the most critical projectile size.

The design equations are based on empirical data, and are given in 3.1, 3.2 and 3.3. They are used to calculate the thickness of the aluminium plates in cm and the areal density of the intermediate bumper in g/cm^2. It is also given that the mass ratio between Nextel and Kevlar should be 3:1, which can be used to calculate the thickness of the intermediate layers.

[1]https://nssdc.gsfc.nasa.gov/planetary/factsheet/moonfact.html [Cited: 14-6-2017]

$$t_b = c_b d_p \frac{\rho_p}{\rho_b} \quad (3.1) \qquad m_{Nextel-Kevlar} = c_{N-K} d_p \rho_p \quad (3.2)$$

$$t_w = c_w \left[\frac{c_o d_p \rho_p}{t_b \rho_p + m_{Nextel-Kevlar}}\right]^{1.1} M_p^{1/3} V_n (cos^{0.5}\theta)\rho_w^{-1} S^{-2} (\sigma_w/40)^{-0.5} \quad (3.3)$$

For the first iteration, the same design projectile was used as the one for the ISS, namely an aluminium sphere of 1cm in diameter (i.e. d_p = 1cm, ρ_p = 2.7g/cm^3, M_p = 1.41g). Also, [5] suggests the use of Al6061-T6 for the outer bumper, Nextel AF10 and Kevlar 29 for the intermediate bumper and Al2219-T87 for the rear wall (i.e. ρ_b = 2.7g/cm^3, ρ_w = 2.84g/cm^3, σ_w = 57.0ksi). The constants are given as c_b = 0.15, c_{N-K} = 0.23, c_w = 8.84s/km, c_0 = 0.38. Based on the research from the Midterm Report, the internal offset between the front bumper and rear wall has been set to S_{b-w} = 30cm.

To determine a proper design velocity, meteorite statistics have been consulted [6]. Figure 3.1 shows a velocity distribution of meteorite impacts on the lunar surface. Based on this, a design velocity of V = 45km/s has been selected. Additionally, a worst-case impact angle is assumed for sizing the meteorite protection (i.e. normal impact angle, so θ = 0rad, $V_n = V$).

The performance equations for high, low and intermediate velocity regime are given in Equations 3.4, 3.5 and 3.6 respectively.
For V≤2.6cos$^{-0.5}$ θ:

$$d_c = K_{L-SW} V^{-2/3} (cos(\theta))^{-4/3} \rho_p^{-0.5} \left[t_w (\sigma_w/40)^{0.5} + C_L m_{b-total}\right] \quad (3.4)$$

For V≥6.5cos$^{-0.75}$ θ:

$$d_c = K_{H-SW} (t_w \rho_w)^{1/3} \rho_p^{-1/3} (\sigma_w/40)^{1/6} V^{-1/3} (cos(\theta))^{-0.5} S_{b-w}^{2/3} \quad (3.5)$$

For 2.6cos$^{-0.5}\theta$ < V < 6.5cos$^{-0.75}\theta$:

$$\begin{aligned} d_c = & \left[\frac{K_{Li-SW}\left[t_w (\sigma_w/40)^{0.5} + C_L m_{b-total}\right]}{cos(\theta)\rho_p^{0.5}}\right] \cdot \left[\frac{6.5(cos(\theta))^{-0.75} - V}{6.5(cos(\theta))^{-0.75} - 2.6(cos(\theta))^{-0.5}}\right] \\ & + \left[K_{Hi-SW} (t_w \rho_w)^{1/3} \rho_p^{-1/3} (\sigma_w/40)^{1/6} (cos(\theta))^{-0.25} S_{b-w}^{2/3}\right] \cdot \left[\frac{V - 2.6(cos(\theta))^{-0.5}}{6.5(cos(\theta))^{-0.75} - 2.6(cos(\theta))^{-0.5}}\right] \end{aligned} \quad (3.6)$$

The constants used in these equations are K$_{H-SW}$ = 0.6 km$^{1/3}$s$^{-1/3}$, K$_{L-SW}$ = 2.35 g$^{0.5}$km$^{2/3}$cm$^{-3/2}$s$^{-2/3}$, C$_L$ = 0.37 cm^3/g, K$_{Hi-SW}$ = 0.321, K$_{Li-SW}$ = 1.243 g$^{0.5}$cm$^{-3/2}$.
Besides the Whipple shield itself, the Multiple Layer Insulation (MLI) also contributes to the ballistic shielding performance. This contribution is shown in Equation 3.7 and applies when the MLI is attached directly on top of the rear wall. For the high-velocity regime, the MLI can contribute to the shield's performance if it were to be positioned more towards the front bumper. There is no information about whether the MLI's ballistic contribution remains the same when this is done, therefore the conservative approach is taken by disregarding it for that situation. Equation 3.8 shows this contribution, where k_{MLI} = 1.4cm^3/g and S_{MLI} is the offset between the MLI and the rear wall in centimetres.

$$\Delta d_c = 2.2 m_{MLI} \rho_p^{-0.47} (V cos(\theta))^{-0.63} \text{ when } S_{MLI} = 0 \quad (3.7)$$

$$\text{For } V_n \geq V_{high},\ \Delta d_c = k_{MLI} m_{MLI} (S_{MLI}/S_{b-w})^{0.5} \text{ when } S_{MLI} \neq 0 \quad (3.8)$$

With an initial estimate of the MLI's density and thickness of ρ_{MLI} = 1.4g/cm^3 and t_{MLI} = 5.0cm, a significant portion of the shielding performance originated from MLI as opposed to the Whipple shield itself. The critical diameter was obtained in the low-velocity regime. Due to the MLI, it increased from 0.29cm to 1.17cm, which meant that the design now performed adequately. Whereas the Whipple shield weighed around 1100kg, the MLI weighed a tremendous 9100kg.

For the second iteration, a more refined method was implemented to determine a projectile size that is more representative for lunar impacts, as opposed to impacts in Earth-orbit. Literature findings of lunar meteorite flux from previous reports have been used to base the future iterations on.

1. The desired reliability factor X is inserted into the model. This factor is the shielding reliability of a single module over the course of the entire mission, i.e. 10 years.
2. A yearly reliability is calculated from the input, namely $X^{1/10}$.
3. A yearly chance of failure is derived from this by simply calculating $1 - X$.
4. This probability is inserted into a Poisson distribution where $k = 1$ (one critical impact means mission failure). This leads to a yearly meteorite impact rate of $e^{-X} * X$ per year.
5. The impact rate is then divided by the vulnerable surface area of the module, which leads to a yearly meteorite flux.
6. Data points from Figure 3.2 have been put into Excel and an exponential regression line has been constructed. Using the function of the regression line, the meteorite flux is finally converted to a projectile size for which needs to be designed. It can be noted that the sizes in Figure 3.2 have been computed using a projectile density of 1g/cm^3. For safety reasons and due to a lack of other information, it has been decided to design for the projectile sizes given in Figure 3.2, even though a projectile density of 2.7g/cm^3 is used for this design. The projectile density of 2.7g/cm^3 was selected after inspecting Figure 5.2 from the Midterm Report, which shows a histogram of the meteorite densities that occur.

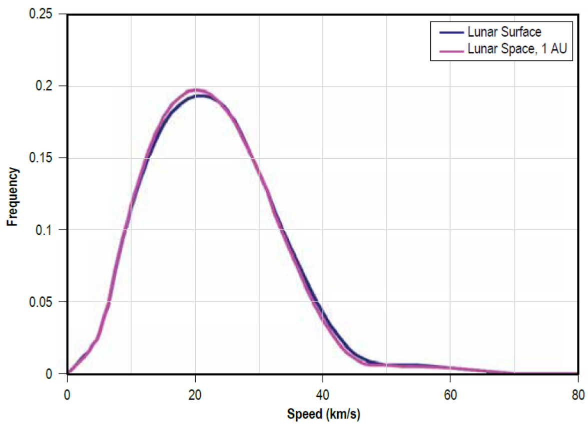


Figure 3.1: Velocity distribution of meteorites on the lunar surface and within the lunar space environment [6].

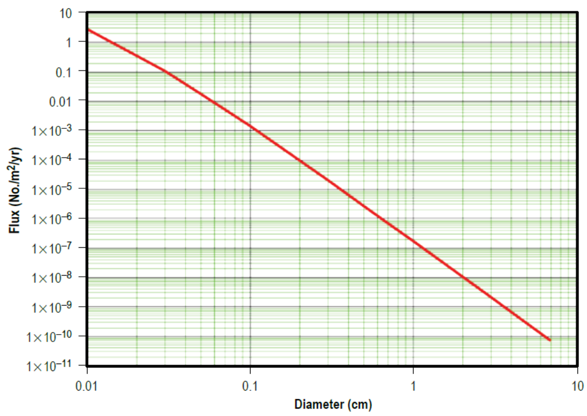


Figure 3.2: Meteorite impact flux near the Moon as a function of particle size (sizes were computed assuming a density of $1g/cm^3$) [6].

For the intermediate iteration, new information from the thermal protection subsystem led to the reduction of the MLI thickness to 2.0cm, which reduced the weight from 9100kg to around 4000kg. By adhering to the provided design equations, a reliability factor of 0.994 could be achieved, as this led to $d_{des} = d_{crit} \approx$ 0.536cm. In order to triple the shielding reliability (i.e. R = 0.998) to meet requirements, some of the layers ought to be thickened manually.

Before arriving at a solution for the next iteration, new and final information about the MLI came in: the MLI now had a density of 0.7941 g/cm^3 and a thickness of 0.442cm. As the MLI properties have changed drastically over the design process, the significant contribution it had initially had faded by now. Because of this, an abysmal reliability of 0.263 was obtained by adhering to the design equations, with d_{crit} = 0.093cm. Just like the previous iterations, the low-velocity regime proved to be the most problematic by far. For that reason, the MLI stays where it was located at the start of the design process, namely right on top of the rear wall.

Evidently, the most recent iteration called for manual adjustments. To start off, the reliability was constrained to 0.998 (to meet requirement SYS-EP-02-02), which led to a design projectile of 0.747cm in diameter and a mass of 0.589g. In order to get the most critical projectile above this size, the aluminium rear wall was considered first: it had a thickness of 0.109cm. Input from the Structures department led to a required minimum thickness of 0.30cm for this material layer. Besides acting as the rear wall of the meteorite shield, it also serves as the pressurisation shell. This is explained in detail in section 4.3.

Despite the increased rear wall thickness, the shielding system was still not strong enough for the ballistic velocity regime. Now, two options were explored: increasing the thickness of the rear wall even further, or increasing the thickness of the intermediate layer, such that d_{crit} reached 0.747cm. The latter option led to increasing the areal density of the Kevlar-Nextel layer from 0.796g/cm^3 to 2.51g/cm^3, or a thickness increase from 0.440cm to 1.133cm. This increased the final weight of the Whipple shield to 3025kg. The option of only increasing the rear wall thickness to 0.64cm proved to be vastly superior, with a final Whipple shield weight of 2133.1kg. An overview of the iterations can be found in Table 3.1. For the mass budget, it is important to note that the rear wall of the Whipple shield is incorporated in the Structures department, as it also serves to provide structural integrity. For that reason, the Whipple shield's weight contribution to Environmental Protection is formally 632.4kg.

Table 3.1: An overview of the iterations performed for the meteorite shielding system.

	ISS design	Variable reliability	Constrained reliability [SYS-EP-02-02] (d_{crit} = 0.747cm)			
Iteration > Design parameter	**Initial result**	**Inter-mediate result**	**Final result**	**Applied structural constraint**	**Increased inter-mediate layer**	**Increased rear wall**
Internal wall offset (S_{b-w}) [cm]	30	30	30	30	30	30
Total thickness [cm]	30.309	30.159	30.221	30.412	31.105	30.752
Shielding reliability [-]	N/A	0.994	0.998	0.998	0.998	0.998
Critical projectile diameter [cm]	1.0	0.536	0.2612	0.437	0.748	0.750
MLI thickness [cm]	5.0	2.0	0.442	0.442	0.442	0.442
MLI areal density [g/cm^2]	7.0	2.8	0.351	0.351	0.351	0.351
MLI mass [kg]	9095	3329	176.9	176.9	176.9	176.9
WS areal density [g/cm^2]	1.479	0.773	1.077	1.618	3.664	2.584
WS mass (excl. MLI) [kg]	1111	685.3	889.0	1336	3025	2133

RAMS Characteristics and Sensitivity Analysis
The RAMS-characteristics (Reliability, Availability, Maintainance and Safety) have been prevalent during the design process of the meteorite shield. An attempt to quantify the reliability of the system using meteorite statistics has been made. To keep Safety in mind, a conservative approach has been applied during the entire process, but especially with the use of the meteorite statistics (i.e. the high design velocity derived from Figure 3.1 and the density conversion from Figure 3.2). Regarding availability, all materials that are used have already been tested and applied extensively on Earth. While metallurgy has already been fully developed, there might be some room for improvement for the intermediate fibre layers in the future. Finally, maintainability still needs to be addressed. After an extensive period of exposure, some areas of The Shell might require replacement. For that reason, a panel construction of the shielding system appears to be the most convenient option, as this allows for localised replacement of damaged areas.

The meteorite shielding system contains two major parameters, namely the shield's total thickness and the shield's total areal density. These parameters are mainly affected by the size of the design projectile, which in turn depends on meteorite statistics and the desired shielding reliability factor. If the frequency of hypersonic particle impacts increases, the most efficient way to address this is by increasing the internal wall offset. On the other hand, an increased occurrence of ballistic projectiles can only be addressed by increasing the wall's areal density, for which increasing the rear wall thickness proved to be the most effective.

Recommendations

As it stands, the meteorite shielding system has the dimensions shown in the most rightward column of Table 3.1. The critical diameters related to this layout are $d_{crit-low}$ = 0.750cm, $d_{crit-medium}$ = 2.925cm, $d_{crit-Vdes}$ = 1.558cm, which means it is overdesigned for hypersonic particles. Equation 3.4 shows that $d_{crit-low}$ is the only critical diameter that is not affected by the internal offset S_{b-w} (as the projectile does not shatter at this velocity). For this reason, smaller internal offsets were tried (with t_{rear} constrained at 0.640cm) until one of the other critical diameters also reached around 0.747cm, such that the design was optimal. After a few trials, it turned out that an internal offset of only 14.7cm led to $d_{crit-low}$ = 0.750cm, $d_{crit-medium}$ = 1.389cm, $d_{crit-Vdes}$ = 0.750cm, which could have been a significant space reduction. Unfortunately, there was not enough time to implement this change, as it would require too much time to perform a substantially new iteration of the other subsystems, and it is therefore a recommendation for future design iterations.

Passive Thermal Control

The detailed design of the thermal control starts with the passive thermal control for The Shell. To decrease the conductivity, it is more efficient to use thin layers on top of each other than using one kind of material with a greater thickness. This kind of insulation is called MLI [7]. The chosen MLI is made of a layer of aluminised Mylar which is known for its very low conductivity and two types of Dacron: one net layer for spacing between the layers and one fabric layer for reinforcement [8]. This MLI is chosen over other MLI structures since it is light, it needs little volume and has proven to have a very low conductivity. In addition, a lot of research has already been done for this kind of MLI, therefore it is a reliable choice. In total, the MLI has a conductivity of 0.00005 W/(mK) per layer [8]. As explained in the previous section, the MLI is situated on the rear wall of the Whipple shield. The Whipple shield itself contains Nextel and Kevlar, which also contribute to the insulation capacity of The Shell. Especially the vacuum gaps efficiently reduce the heat flow, since it has no conductivity, so the heat is only transmitted by radiation from one plate to another. To calculate the number of layers of MLI needed for The Shell the heat influx has to be calculated. It is known that the heat flux of the Sun has a maximum value of 1361.00 W/m^2 for the Apollo 11 site [2]. The heat flow per plate can be calculated using Equation 3.9 [9].

$$Q = \frac{A_{ex} \cdot \Delta T_{pl}}{\frac{t_{pl}}{k_{pl}}} \qquad (3.9) \qquad\qquad \frac{t_{tot}}{k_{tot}} = \sum_{i=1}^{n} \frac{t_{l_i}}{k_{l_i}} \qquad (3.10)$$

For this equation, $t_p l$ is the thickness and k_{pl} is the thermal conductivity of the plate. If a number of different materials (n) is used on top on each other, the thicknesses and conductivities can be added as calculated in Equation 3.10.

ΔT_{pl} is the temperature difference between the outer side of the plate and the inner side of the plate. A_{ex} is the surface area of the plate of which the temperature is calculated

Since there is a vacuum in between the different layers of the Whipple shield the ΔT_{pl} need to be calculated for every layer. The temperature of the outer layer of the Whipple shield can be calculated by Equation 3.11 [9]. Since this outer layer is made from aluminium, which has a very high conductivity (166W/(mK)) and is very thin, it is assumed that there is no temperature difference between the outer and inner side of the both the layers of aluminium.

$$T = \sqrt[4]{S_R \frac{\frac{\alpha_a}{\epsilon} \frac{A_p}{A_r}}{\sigma_{S-B}}} \qquad (3.11)$$

S_R is the solar irradiance. σ_{S-B} is the Boltzmann constant. A_p, short for projected area, is the area the Sun projects on, whereas A_r (the radiating area) is the area that radiates outside. α_a is the absorptance of the material, whereas ϵ is the emittance of the material. The values of these two can be changed, depending on which coating is going to be used.

The structure of The Shell consists of panels, which will be further elaborated on in chapter 4. Since the outer layer is made of aluminium it is assumed that it has a perfect conductivity. The effect of that is that the reflective area is the area of the complete panel when it is shined upon. This is a valid assumption since the conductivity of aluminium is 166W/(mK) [10] and the maximum area of one of the plates is only around 1.2m^2.
Since the Apollo 11 site is situated almost on the equator [2], the Sun will turn straight over the habitat. With this knowledge, the incident angles on the panels of the habitat structure can be calculated and the ratio of A_p/A_r with that as well.

After having found the right approach, these steps are written in a code. The code calculates the temperature of every single outer plate of The Shell for every angle of inclination of the Sun. From the temperature of the outer plates, and temperatures of the middle and inner plate were calculated using an initial heat flow for every panel at every angle. In a loop in the code, the initial heat flow approaches the actual one by checking the temperature of the most inner plate. When this temperature reaches the desired room temperature, the heat flow from one outer panel is known. Summing up the heat flows of the cylindrical and spherical part of the hard-shell structure at a given solar inclination angle gives the total heat flow of the hard-shell module at that specific angle of inclination. Finally, the heat flows are plotted for every inclination angle.

After iterating the coating, the layers of the Whipple shield and the MLI layers, it is chosen to use 13 layers of MLI and have an outside coating made from White-Epoxy paint for aluminium [2]. This coating has an emissivity of 0.924 and and absorptance of 0.248. The result is given in Figure 3.3.

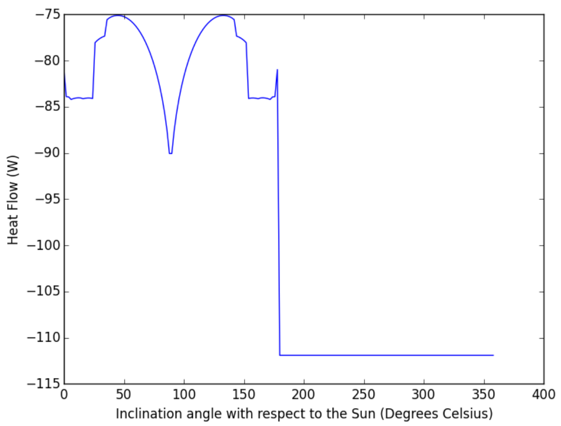


Figure 3.3: Heat flow for The Shell.

The graph shows that the heat flow is both negative during the day and during the night. This is caused by the chosen coating which has a very low absorptivity/emissivity ratio. Other things can be noticed about the graph is that there are two tops when the Sun just rises and when the Sun just goes down. In addition, the maximum tops, at which the least heat flow is going outside the habitat, are at an inclination angle of the Sun of 45 degrees Celsius and the most heat flow happens when the Sun is straight above the hard-shell. This can be explained by the geometry of The Shell. If the Sun is at the top the whole cylindrical part (which is the biggest part) does not take in heat, but it does dissipate heat coming from the inside. At 45 degrees, most plates are shined upon, which gives the least heat flow. A drawing of the situation is shown in Figure 3.4. The yellow arrows are the radiation from the Sun and the blue arrows are heat rejected from the habitat. The size of the arrows shows the amount of heat rejected from the habitat. It depicts that when the surface is facing the solar heat radiation perpendicularly, the outgoing heat is minimum.

The choice for the coating and the number of MLI layers is based on the assumption that even if systems fail at the least one computer inside will dissipate heat continuously, which equals to 90W. At the moment the Sun is directly above the habitat, which means The Shell does not generate any shadows and it is difficult to release heat, the heat flow to the outside is counteracted completely by the heat generated by the continuous computer. At that moment no heating or cooling is needed, only for extra

[2]http://www.alternatewars.com/BBOW/Space/Spacecraft_Ext_Temps.htm [Cited: 22-06-2017]

systems and humans.

The pins situated in the shell of the habitat also conduct heat. The maximum heat flow, from inside to outside, through one strut would be 0.73W, this happens at night. There are 125 struts. This would mean an extra heat loss of 91.25W for the hard-shell at night. This is taken into account in the heating capabilities.

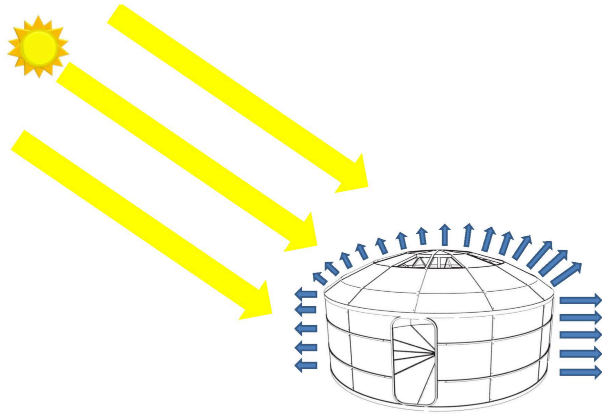


Figure 3.4: Heat rejection from the habitat (not to scale).

Another big influence on the heat flow in- and outside the habitat is the floor. During night and day the floor will radiate heat to the lunar regolith [11]. More investigation should be done on how this lunar regolith will react and what the equivalent temperature of the air in between the habitat and the floor will be. For this design, the heat flow out through the floor is only taken into account during heating. This way the thermal control is designed for the maximum possible. The MLI for the floor is decided to be 25 layers, using more layers will increase mass, but will not decrease the heat flow much more. The outside temperature is 2.7K, which is the background temperature of space. With this a constant heat flow outside can be calculated to be 51W. If the equivalent temperature in between the habitat and the regolith is higher than 2.7K the heat flow will be lower. Because of this for the cooling it is assumed no heat is lost to get to the maximum cooling possibly needed.
The contribution of the passive thermal control, of the walls and the floor, to the mass budget of the environmental control is 380 kg. This is based on a density of the MLI Dacron, Mylar, Dacron of 747.12kg/m^3 [3] The RAMS and senstivity of this design will be performed in with the active thermal control section.

Passive Radiation Protection

The possibilities of having a passive radiation protection are researched. An outline of the outer protective layers is given in Figure 3.5. The Al 6061-T6, which acts as a front bumper is part of the meteorite protection, together with the ceramic and aramid fibre layers. The MLI layer is part of the thermal protection while the rear wall, Al 2219-T87, carries the structural loads such as the pressurisation of the habitat. Finally, the layer which lies most inward is the radiation protection layer. All the layers are taken into account during the radiation analysis. The layer of air is needed to simulate the inside atmosphere of the habitat.
The effect of creating secondary particle radiation occurs in a random manner. Thus, a statistical model has to be run to make a proper analysis. The chosen model is GEANT4, which is a toolkit for the simulation of particles through matter, developed by CERN. [4] This model uses a Monte Carlo simulation to predict the amount of ionising dose for each layer configuration. For the simulation of solar particles, the PSYCHIC (Prediction of Solar particle Yields for Characterising Integrated Circuits) model is used. The considered materials for the specific radiation protection layer are water, boron-nitride nanotubes with hydrogen storage (BNNT) and polyethylene. All the mentioned materials have a high concentration of low Z atoms, which are more effective in blocking secondary radiation than high Z atoms.

[3]http://www.eiccompany.com/en/assets/download/data-sheet-dmd.pdf
[4]https://www.spenvis.oma.be/ [Cited:21-06-2017]

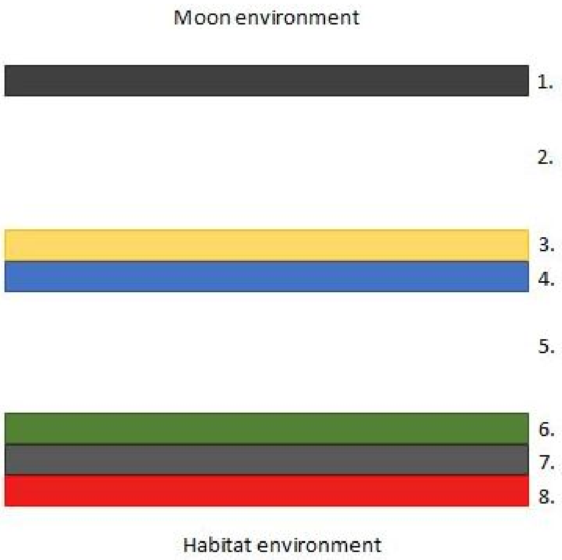


Figure 3.5: Overview of protective layers.

Table 3.2: Layer characteristics for The Shell

Layer number	Material	Thickness [cm]
1	Al 6061-T6	0.112
2	Vacuum	14.55
3	Nextel AF10	0.129
4	Kevlar 29	0.081
5	Vacuum	14.55
6	MLI	0.442
7	Al 2219-T87	0.64
8	Polyethylene	23.5
9	Air	270

	Total ionising dose during SPE [rad]		
Layer thickness [cm]	Water	BNNT	Polyethylene
1	233.2	245.6	233.8
5	81.2	111.0	76.6
10	29.5	50.3	28.6
15	16.5	31.3	15.9
20	10.3	17.6	10.2

Table 3.3: Ionising dose for different material layers.

A comparison between the total ionising dose within the habitat during SPEs is given in Table 3.3. From Table 3.3 it can be concluded that polyethylene is the preferred material for passive radiation shielding, as it provides the most shielding per thickness. The effectiveness of radiation shielding with polyethylene during SPEs is given in Figure 3.6. The ionising dose is plotted on a log scale and an exponential trendline is plotted with a coefficient of determination of 0.9. From Figure 3.6 it can be concluded that 23.5cm of polyethylene is enough to shield against SPEs for a year. The layers are defined according to Figure 3.2.

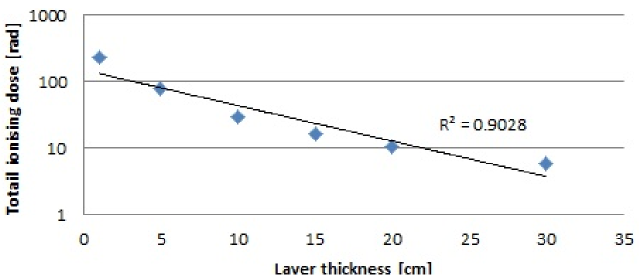


Figure 3.6: Dose analysis for different thicknesses of polyethylene

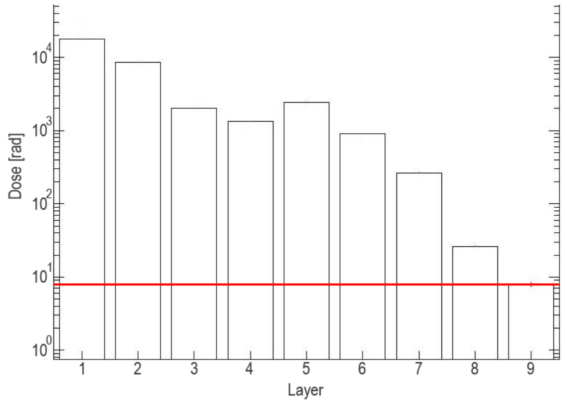


Figure 3.7: Analysis average dose in wall layers during SPEs with a layer of 23.5cm of polyethylene (8).

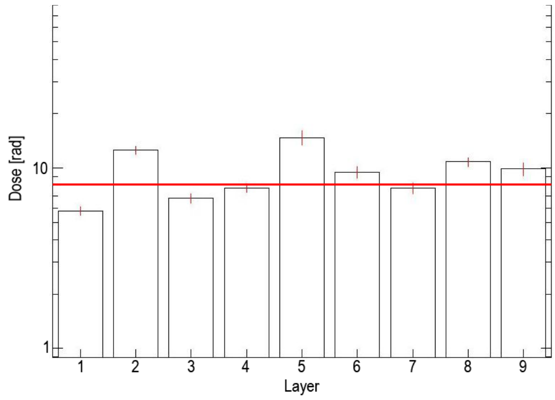


Figure 3.8: Analysis average dose in wall layers during GCR with a layer of 23.5cm of polyethylene (8).

The dose analysis of a radiation shield with a 23.5cm polyethylene layer is given in Figure 3.7. The red line is the maximum allowable dose.
The amount of ionising dose during SPEs over a year is 7.8rad with a margin of 0.5rad. The weight of this shield alone is 18t.
Another analysis is run to predict the ionising dose caused by GCR. The results are shown in Figure 3.8. A 23.5cm polyethylene layer is sufficient to shield against SPEs and partially against GCR. However, the mass of the shield is too heavy to put in a launch together with the hard-shell module and installing the shield later would include more complexity. Therefore, the option of active radiation protection is explored and considered in section 3.3.

3.2. Inflatable Modules - Passive Environmental Protection

Since the inflatables have a different structure than The Shell, another way of special shielding needs to be designed. In this chapter the design of the shielding and thermal control will be explained.

Meteorite protection

The environmental protection structure of the inflatable modules is a very particular case in space engineering as only flexible materials are used. The interest of inflatable concepts is high due to the high volume to mass ratio. Therefore, major space players are already conducting research regarding its applicability. In the following, the design process of the inflatable structure is presented. It has to be noted that most of the calculations are based on theoretical equations which are adapted to the needs of a lunar habitat. Tested or even proven values and procedures are limited or not available at all. Thus, the values presented need to be used with care.

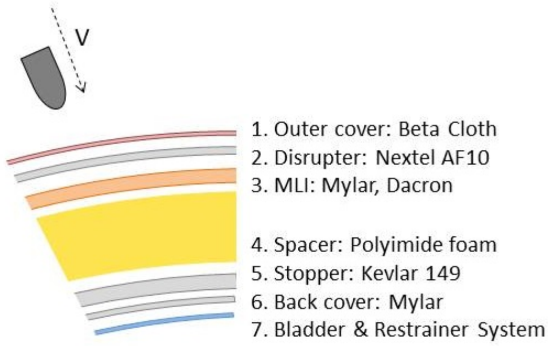


Figure 3.9: Cross section of the inflatable structure: multishock thermal blanket (MTB). The spacing in between the layers are only for visualisation purposes.

Generally, inflatable structures are built up of a bladder which is airtight and a restrainer. The bladder is pressurised from the inside resulting in outward expansion. The further it expands, the more the material stretches which leads to a loss in strength. A restrainer layer is employed in order to limit the stretching capability of the bladder. Additionally, it protects the bladder from any external harm.

A regular inflatable structure is not strong enough to sustain the hostile lunar environment. Therefore, next to the basic inflatable structure, several strengthening layers and methods are applied. In short, the shell of the inflatable structure of LEAP is featuring a fusion of meteorite protection, thermal

control and sealing system. Thus, the basic inflatable shell is covered by a multilayer blanket called a multi-shock thermal blanket (MTB). An MTB contains all typical thermal layers as discussed in REF while the MMOD composition philosophy follows the same approach as the Whipple shield of the hard-shell module which can be found in REF.
The general cross section of such a shield is shown in Figure 3.9. The MTB is a so-called toughened thermal blanket. Layers 1, 3 and 6 are the basic layers of a classic thermal blanket while the layers 2,4 and 5 are added to offer MMOD protection. The shell structure is divided into two parts: the lower half of the structure is in contact with the lunar surface, the other is facing the environment. As the lower half is protected from MMOD, the bladder & restrainer system needs to be strengthened to sustain the surface roughness only. For this, the addition of a beta cloth suffices to protect the restrainer against abrasion. Therefore, only the upper part of the structure will be covered with a thickened MTB leading to a desirable reduction in mass. Thus, in the following section, the design of the MTB of the upper half only will be considered. The design of the bladder & restrainer system can be found in 4.4. Further, it has to be noted that adherents and other kinds of connections are not integral to the calculations. During the test phase it ha to be determined whether these connections lower the performance. In this case, the design needs to be adjusted accordingly.

MMOD Protection

The MMOD protection layers are made of high performance materials and serve the same purpose as the traditional Whipple shield layers. The disrupter (layer 2) breaks up the projectile to slow it down and to reduce its energy. The spacer (layer 4) allows the fragments to spread while the stopper (layer 6) is employed to finally stop the fragments.

To ensure sufficient MMOD protection, a suitable calculation method has to be found. The NASA handbook and further research regarding the inflatable shielding technology calculations is considered [12]. The MTB shielding employs a comparable configuration of layers as the stuffed Whipple shield. However, the equations used are altered slightly due to the flexibility of the material. Firstly, the area density of the bumper m_b (layers 1-4) and rear wall m_w(layers 5-6) need to be calculated with Equation 3.12.

$$\begin{aligned} m_b &= m_1 + m_2 + f_{MLI}(m_{MLI+m_{foam}}) \\ m_w &= m_5 + m_6 \end{aligned} \tag{3.12}$$

Here, the areal densities m_{layer} are numbered as found in Figure 3.9. Furthermore, MTB tests concluded that an 'equivalent' areal density of the MLI-foam combination approximates the protection capabilities of these materials best using a factor f_{MLI} of 0.25 [13].
For low velocity projectiles ($V_{slow} \leq 2.4cos^{-0.5}\theta$) Equation 3.13 holds. This equation assumes the projectile perforating the first bumper while deforming and losing velocity. Equation 3.14 shows the calculation for hypersonic velocity ($V_{fast} \geq 6.4cos^{-0.25}\theta$) projectiles. These projectiles burst into smaller particles once it perforates the first bumper.

$$d_{crit,slow} = 2.7\frac{0.5m_w + 0.37m_b}{\rho_p^{0.5} \cdot (V_{slow}cos\theta)^{2/3}} \tag{3.13}$$

$$d_{crit,fast} = 1.24\left[\frac{m_w \cdot S_{b-w}^2}{c_w \cdot \rho_p \cdot (V_{fast}cos\theta)}\right]^{\frac{1}{3}} \tag{3.14}$$

Here, ρ_p is the density of the projectile, cw is the coefficient using Kevlar as a rear wall material and S_{b-w} is the total thickness of the MTB. Furthermore, θ is the angle of incidence. For intermediate velocities ($2.4cos^{-0.5}\theta$ $V_{med\ med} < 6.4cos^{-0.25}\theta$), a the projectile will not break up, neither is it deforming. Thus, by interpolating between the equations Equation 3.13 and Equation 3.14 as seen in Equation 3.15 a linear prediction of the critical projectile diameter $d_{crit,med}$ can be found.

$$d_{crit,med} = d_{lo} + (d_{hi} - d_{lo})\frac{(V_{med} - 2.4cos^{-0.5}\theta)}{6.4cos^{-0.25}\theta - 2.4cos^{-0.5}\theta} \tag{3.15}$$

In order to analyse the behaviour of toughened thermal blankets several configurations are presented in Table 3.4. The thin and medium sized blankets are predefined options which have been tested already

Table 3.4: Available and proposed blanket configuration and their layer set-up. * Configurations thick and medium are tested options taken from [13].

Blanket layout (Thickness in cm)		**Outer cover**	**Dis-rupter**	**MLI**	**Spacer**	**Stopper**	**Back cover**	**Total**
Fabric		Beta Cloth	Nextel AF10	Alumin. Kapton	Polyimide AC550	Kevlar KM2	Mylar	-
Thickness per layer		0.025	0.011	0.011	2.535	0.017	0.007	-
Density [g/cm^3]		1.0	2.7	0.794	0.0071	1.44	1.38	-
Thin*	Layers	1	2	1	1	3	1	8
	Thickness	0.025	0.022	0.154	2.535	0.051	0.007	2.67
Medium*	Layers	1	12	1	6	12	1	28
	Thickness	0.025	0.132	0.154	15.21	0.204	0.007	15.67
Thick	Layers	1	24	1	12	24	1	53
	Thickness	0.025	0.264	0.154	30.42	0.408	0.007	31.51
Super thick	Layers	1	48	1	24	48	1	103
	Thickness	0.025	0.528	0.154	60.84	0.816	0.007	62.77

[13] while the thick and super thick blankets are proposed configurations. These proposed blankets are designed by doubling number of layers of the medium sized which has been adjusted with respect to the thin blanket. It has to be noted that the other layers, namely the classic thermal blanket layers were tested of performance, however, they did not alter the MMOD performance significantly. Thus, these basic layers are kept to be single layers which may be altered during the thermal blanket design. However, the number of MLI layers may increase during the thermal insulation design.
Table Table 3.5 shows more properties of the thermal blanket options. Here it becomes evident, that the increasing thickness also increases d$_{crit}$: a doubling in thickness almost doubles d$_{crit}$. It has to be noted that the material selection is made based on the outcome of a MMOD shielding test. There it is found that the selected materials perform best in terms of hypervelocity impact tests. Thus, other materials are not considered at this point as no test data is available. However, further testing with other shielding material may result in the finding of more suitable materials and thus may enhance the performance of the MTB blanket.

Table 3.5: Comparison of blanket configurations and lunar regolith properties.

Thermal blanket configuration	**Total density in [g/cm^3]**	**Thickness in [cm]**	**d$_{crit}$ in [cm]**, (d$_{crit,min}$: 0.85cm)
Thin blanket*	6.86	2.67	0.07
Medium blanket*	1.17	15.67	0.29
Thick blanket	0.6	31.51	0.6
Super thick blanket	0.29	62.77	1.16

Figure 3.10 shows the performance of the 4 thermal blankets to determine their behaviour with respect to the critical diameter of the meteorites. Furthermore, the horizontal line depicts d$_{crit,min}$. This parameter is determined following the same approach as for the hard-shell module. The exposed surface of the inflatable is 130m^2 assuming that the cylinder is buried halfway. With a reliability of 0.998 the size of d$_{crit,min}$ is found to be 0.87cm. As can be seen, the critical diameter is too low for the tested and also the thick blanket to sufficiently shield from the critical meteorite size. From the graph, it is evident, that only the super thick blanket is able to sustain an impact of a meteorite of the size d$_{crit,min}$.
It is obvious that the super thick blanket is overdesigned, thus, to ensure the design meeting the minimal requirements, the LEAP blanket is created, which meets d$crit, min$ and features a low density. In Figure 3.10 it can be observed that the low-speed velocities of the projectiles impose the highest hazard. The number of layers of the disrupter, spacer and stopper is reassessed and, if possible, reduced. From 3.13 and 3.14 it is evident that the real wall areal density is the leading factor regarding the MMOD strength. Furthermore, the disrupter layer possesses the highest density, thus with respect to mass

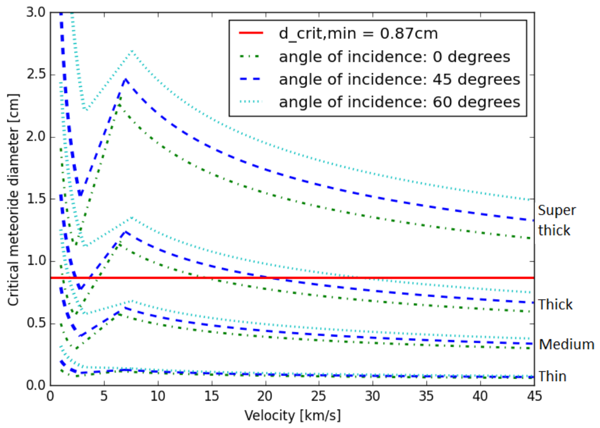


Figure 3.10: Meteorite shielding properties of several toughened several blankets. *Thin and medium thermal blanket configuration taken from [13], Thick and super thick blankets: proposed configurations.

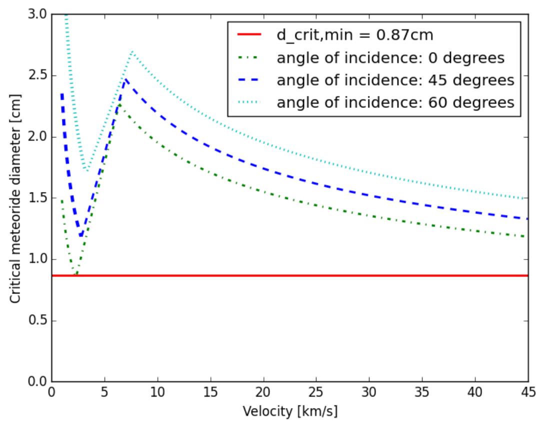


Figure 3.11: LEAP Blanket meeting minimum requirements: $d_{crit,min}$=0.87cm. Amount of layers: Disrupter (24), Spacer (24), Stopper (48).

concerns, it is desirable to reduce its thickness the most. An iterative design process is started to meet the minimal requirements by trying different layer configurations. It becomes evident that the stopper has the largest impact on the performance for both, low and high velocity. It features, however, a medium range density of 1.44g/cm^3. The disrupter features a very high density (2.7 g/cm^3). However, it has a lower impact on the performance than the stopper layers as can be seen in Equation 3.13 and Equation 3.14. Thus, a decrease in the number of disrupter layers results in a less drastic decrease in performance but a higher decrease in weight when compared to the stopper layer. Further, as the spacer material is a low-density foam, its impact on the strength is only significant with a high layer number. As its impact on the mass is only marginal, the number of layers of the super thick blanket is deemed sufficient. Finally, Figure 3.11 shows the final LEAP MMOD protection performance. In this particular design, the numbers of disrupter, spacer and stopper are 21, 24 and 48 respectively. The other layers are kept constant as they influence the performance only slightly. Once the number of MLI layers is determined, a final iteration can be carried out.

MMOD using Lunar Regolith Structure

To avoid an excessive shell thickness and for thermal regulation reasons, the creation of a protective regolith shell is considered. The advantage of using a lunar regolith shield is, that part of the material is already present on site. To reduce the loads on the inflatable structure the regolith shells shall be constructed to be a self-standing tunnel. This tunnel shall be long enough to cover the entire shell. Again, calculations are based on the projectile design diameter which is calculated in the sae fashion as for the Whipple shield. The regolith structure is assumed to have an offset of 0.5m in order to provide sufficient space for the inflatable to inflate without touching the structure. Furthermore, as the inflatable features an exit on each end, the tunnel will not be closed off. Therefore, the inflatable still needs the protection of the MMOD at its end caps, however, with a significantly reduced vulnerable area. The surface area (110m^2) of the tunnel is derived by the dimensions of the inflatable structure. Here, 0.5m is added towards the semi-major and semi-manor axis to account for the surface increase due to the offset. Furthermore, the length of the tunnel is equal to the length of the total structure.
Calculations of the design projectile diameter with a reliability of 0.998 results in a $d_{crit,min}$ of 0.87 which is the same diameter as used for the MMOD design of the inflatable structure. Now, the minimum thickness of the lunar regolith has to be found using Equation 3.16 [5]. In order to offer sufficient protection, the regolith shield is required to stop all particles. By setting $t \geq 3P_\infty$ incipient spall is prevented [5], ensuring the shield not being perforated.

The equation is used to determine the penetration depth of a aluminium shield and thus, has to be used with care and under application of appropriate safety factors. The Brinell hardness (BHN) factor of 670

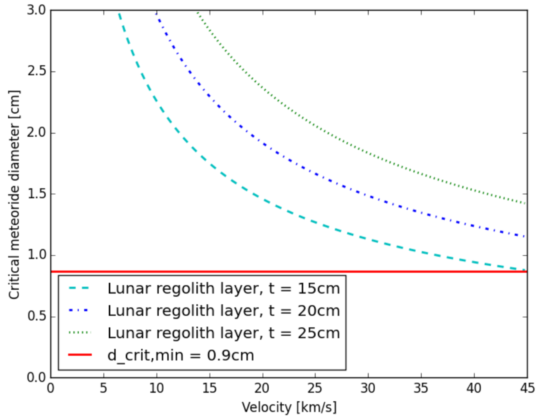


Figure 3.12: Meteorite shielding properties of lunar regolith: critical meteorite diameter of the projectile in centimetre as a function of projectile velocity in km/s.

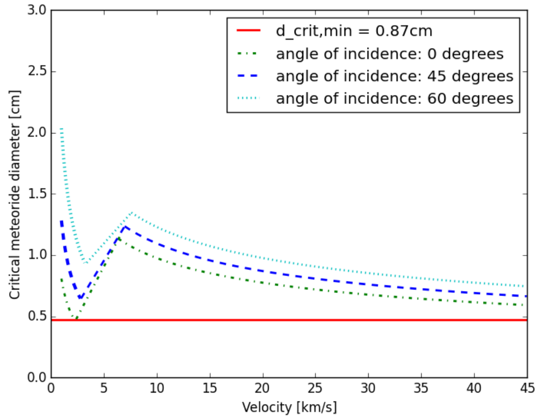


Figure 3.13: LEAP Blanket using lunar regolith meeting minimum requirements: $d_{crit,min}$=0.47cm. Amount of layers: Disrupter (15), Spacer (12), Stopper (24).

is derived of the Knoop hardness which is found to be 850 for lunar dust [14]. Furthermore, C_t is the speed of sound within the material as is assumed to be 3.5km/s which is approximating the speed of sound in concrete or bricks [5].

$$P_\infty = 5.24 \cdot (d_{crit})^{19/18} \cdot BHN^{-0.25} \left(\frac{\rho_p}{\rho_{reg}}\right)^{0.5} \left(\frac{V}{C_t}\right)^{2/3} \tag{3.16}$$

Figure 3.12 shows the shielding capability of lunar regolith with 15cm, 20cm and 25cm thickness. It can be seen that a 15cm thick regolith shield meets the minimum requirement while the others are overdesigned with respect to MMOD protection. In order to save construction time, t = 15cm is chosen.

Applying a lunar regolith shield the exposed inflatable shell reduces drastically. It follows that the $d_{crit,min}$ in the calculations regarding the inflatable shell reduces, as the only the end caps are exposed. Following the calculations for finding the projectile design diameter as done for the hard-shell module and with an exposed surface area of 19.9m^2 the resulting d_{crit} is 0.47cm. Thus, the MTB can be decreased as well. Although the cylindrical part on the inflatable is not exposed to MMOD during its operational lifetime, it needs to be shielded from it during the deployment time. Thus, the same blanket will be used on the entire upper half. For mass reduction, it can be assessed, however, whether is it sufficient to use a single layer MTB along the shielded area and use a protective cover during the deployment phase. Then, the designed MTB will only be applied at the exposed end caps. starting off from the thick blanket, which is slightly over designed, the same iterative process and reasoning as used for the inflatable shell are followed. Figure 3.13 the selected thermal blanket where it can be seen that it meets the minimum requirements.

RAMS Characteristics and Sensitivity Analysis

Generally, the calculation methods NASA uses regarding MTB is based on test data coefficients. These coefficients are generated with expertise. For inflatable structures, however, the expertise is very low, thus, the calculated values are only an indicator of the actual protection needed. Furthermore, the use of fabrics needs to be researched with respect to their performance in hazardous environments. This also holds for the proposed rigidifying process as the rigid structure will have a very different load case than the flexible one. Thus, up to this point, the toughened thermal blanket needs to be researched further to prove its reliability. Thorough testing and improvements of the equations at hand need to occur in order to deem the structure reliable and safe enough for astronauts to live in.
Further, the materials chosen are derived from test materials and can be subject to change. However, the presently chosen materials are available and commonly used. With proceeding time, there may be more suitable materials available which can enhance the performance of the structure.

[5]http://www.engineeringtoolbox.com/sound-speed-solids-d_713.html [Cited: 14-07-2017]

Several competitors such as Bigelowaerospace are researching inflatable structures and are currently testing models such as the BEAM. A collaboration helps with exchanging expertise and knowledge and thus may result in an improvement of the structure and its operations.
Generally, the MTB structure is very insensitive with respect to the environmental conditions. Due to the variable layers, an increase or decrease in hazardous conditions can simply be addressed with an adjustment of the layer thicknesses. This, of course, will induce a weight change which needs to be accounted for during launch preparations.

Passive Thermal Control

For the thermal control, both of the inflatable both the options of using regolith and not using regolith are investigated.

The self-sustaining shell: is only with MLI and without regolith. As inside The Shell, the insulation layer is put on the meteorite protection. In contrast to that, the inflatables are made of fabrics which touch each other. This increases the conductivity inside the inflatable shell. The conductivity of the outer layer of beta cloth is very high, as the resistivity is negligibly small [15]. Because of this, it can be assumed that the heat coming from the Sun transfers fast through the material and that the upper material has the same temperature.

As explained before, the idea is to lay the inflatable in a regolith bed. This means that half of the inflatable will be surrounded by regolith[16]. The material of the downside of the inflatable will also be thinner than the upper side. This change in thickness of the wall will prevent heat to be conducted from the upper side to the lower side of the inflatable. Since thick layers of regolith barely conduct [11]. the bed can be assumed to not conduct. Now the layers of MLI can be calculated in the same way as done in the hard-shell.

First of all the outside temperature is calculated with Equation 3.11. The radiating area is half of the cylinder and the projected area can be calculated using the inclination angles the Sun makes and the geometry of the inflatable. The conductivity of the complete soft-shell can be calculated using Equation 3.10. Finally, the heat flow will be calculated using Equation 3.9. The heat flow can then be plotted against the number of layers of MLI, see Figure 3.14.
It can be noted that the effectiveness of one MLI layer will decrease with an increase in the number of MLI layers. It is chosen to use 29 layers of MLI on the upper side of the inflatable, since at that point the derivative of the graph has an angle of 45 degrees and the MLI can be considered effective. After that point it adds weight, but not that much of heat resistivity.
On the lower side, the regolith is already very insulating [11]. It is then also chosen not to have any MLI insulation. The equatorial temperature of the regolith at 30cm deep is 250K [17]. Comparing this to the 295K inside the habitat it can be assumed that the regolith just around the habitat is going to have an equilibrium temperature, but not release any more heat after that. The heat flow of the inflatable with this amount of layers is shown in Figure 3.15. The difference in heat flows of the two inflatables is caused by the angle the inflatables make with the equatorial plane of the moon. Even if this is changed the maximum and minimum heat flow stay the same. With these values the active thermal control is designed.

The regolith protection shield: has a positive effect on the thermal control. Since the regolith has a low conductivity up until approximately 400 K it is difficult for the heat to pass through it [11]. The total conductivity at a certain temperature can be calculated through Equation 3.17. With c_1= 1.281e-2 and c_2=4.431*e-10.

$$k = c_1 + c_2 * T^3 \tag{3.17}$$

If the regolith wall was not coated and the Sun is at its highest, the outside layer of the regolith wall would be 453K hot. This is calculated using Equation 3.10 and the fact that emissitivity of regolith is 0.32 and an absorbitivity of 0.88[18]. The conductivity of the regolith at that point will be then 0.054W/(mK). Since this will be the hottest part of the regolith this will also be the maximum conductivity for this structure.

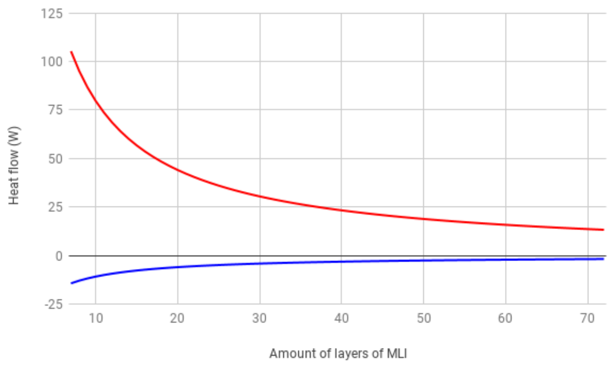


Figure 3.14: Amount of layers MLI against heat flow at certain inclination. Blue line is at night, red line is during the illumination time.

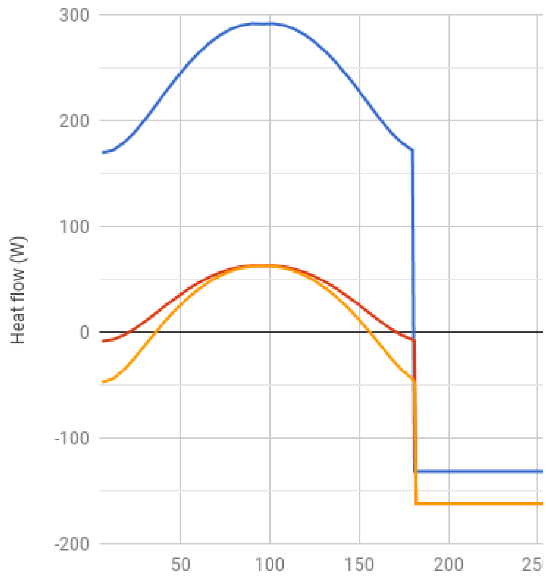


Figure 3.15: Heat flow coming inside or going outside the inflatable. From 180 degrees onwards the heat flow is constant. The blue line is the heat flow with a regolith tunnel of 20 cm thick and without MLI. The yellow and red lines are the heat flow for the inflatables with 29 layers of MLI.

In between the regolith structure and the inflatable there would be a vacuum layer. This means there is no conductivity between the regolith structure and the fabric of the inflatable.

The regolith acts different in conductivity than other materials. Because of this, for the trade-off between a self sustaining shell or a regolith shell, an estimation is made of the regolith shell in which the conductivity is constant and equals 0.015W/(mK) , this number is chosen since it is the average conductivity of regolith [11].

As with the hard-shell the heat flow is calculated over the regolith wall and the layers in the inflatable. With this the layers of MLI needed, will be calculated. It is found that the thickness needed for the meteorite protection, which is 15 cm, does not have enough insulation properties as desired. To use only regolith as insulation the tunnel needs to have a thickness of at least 3m. This is because the absorptance/ emissitivity ratio of beta cloth is much smaller than that of regolith. Because of this, the regolith wall takes up a lot of heat, and although the conductivity is low, if the structure is too thin, it will radiate to the other side as well. With 3m thickness the maximum heat flow inwards is 100W.

Figure 3.16: Layer characteristics inflatable.

Layer	Material	Thickness [cm]
1	BetaCloth	0.025
2	Nextel AF10	0.528
3	MLI	0.011
4	Polyamide foam	60.84
5	Kevlar 29	0.816
6	Mylar	0.007
7	Air	340

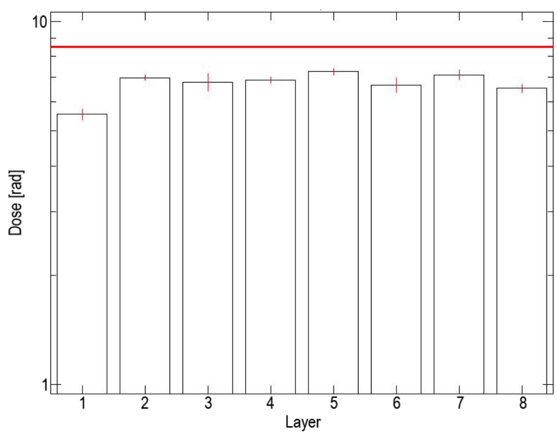


Figure 3.17: Dose analysis during GCR for the soft-shell.

Radiation Protection

The passive radiation shielding for the inflateble module is analysed using the GEANT4 simulation. For the inflatable modules only the effects of the GCR are to be considered, since during SPEs the astronauts will remain in the hard-shell. The layer layout of the inflatables differs from The Shell. For each layer, the used material and thickness is given in Figure 3.16.
The dose analysis for the inflatables module during GCR is given in Figure 3.17. From the analysis, it can be concluded that the soft-shell does not require additional material for radiation shielding, as the ionising dose for a year is 6.6 rad.

Final Design: Trade-Off

The thermal insulation layers are assessed and the number of layers is known, thus the MTB configuration can be optimised with the new values. Again, following the layer reducing process the final LEAP blanket can be found in Table 3.6. For this configuration, a new radiation prognosis is made, resulting in a yearly ionising dose of 6.7rad which is below the maximum radiation dose.

Table 3.6: Final MTB blanket configuration.

Material	Amount of Layers	Thickness [mm]	Mass [kg]
1. Cover: Beta Cloth	1	0.25	32.5
2. Disrupter: Nextel AF10	18	1.98	694.98
3. MLI	29	9.86	1,017.75
4. Spacer: Polyamide foam	18	456.3	421.16
5. Stopper: Kevlar 149	48	8.11	1550.2
6. Back cover: Mylar	1	0.07	12.38
Total	115	477.66	3,728.98

Table 3.7: Trade-off of the inflatable structure MTB set-up: self-protecting versus using lunar regolith protection. Here, the thermal insulation is said to be one layer only, this may change during thermal design. Green: Excellent; exceeds requirements. Blue: Good; meets requirements. Yellow: Correctable deficiencies. Red: Unacceptable.

Design Type	Launches	Mass of MTB	Assembly Cost	Assembly Complexity
Self-stustaining inflatable	1 *green*	3728.98 *yellow*	Low *green*	Low *green*
Inflatable & lunar regolith	2-3 *yellow*	2171.13 *blue*	High (regolith rovers) *yellow*	High *yellow*

Finally, a trade-off between the self-standing inflatable and regolith protected inflatable structure has to be made. It has to be noted that the thickness of the MTB and lunar regolith is adjusted to meet the minimum requirements. Further, the safety requirement is not part of the trade-off as it is said to be integral of both designs. Furthermore, a higher d_{crit} does not automatically render the design to be more protected but rather indicates a higher possibility of being struck by larger projectiles as the exposed area of the structure is larger. Thus, the size of d_{crit} is also not part of the trade-off.

The first advantage of using regolith that comes to mind is mass reduction of the structure to be transported. Consequently, this may result in a reduction of launches. In Table 3.7 it can be seen that the mass of the self- protecting inflatable is almost double.
However, using a protective regolith tunnel, its construction is required prior deployment and assembly. This requires special rovers which have to be transported to the building site giving rise to the need of additional launches. Furthermore, the rovers are very costly and the constructing time of the tunnel is high (Further information in section 11.2). Additionally, the assembly complexity using lunar regolith is extremely high; it requires great accuracy to navigate the inflatable towards and through the tunnel. This may lead to the need of astronauts guiding and positioning the inflatables. The positioning is of high importance as the structure shall not touch the regolith when inflating as it can impose high loads causing damage. Summing up, the lunar regolith tunnel is not a necessity with respect to MMOD and thermal insulation although it leads to a reduction in structural mass.

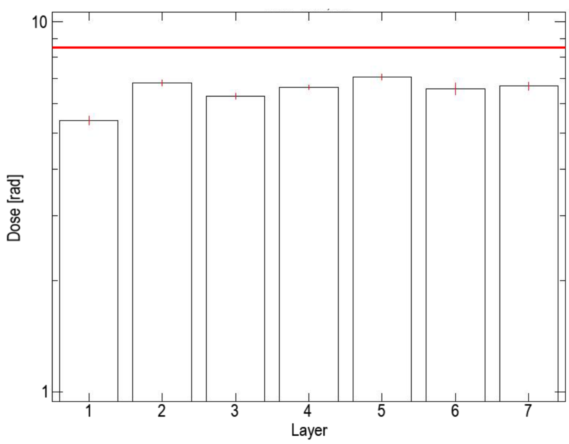


Figure 3.18: Analysis of average dose during GCR for the final soft-shell design.

Further, it was found that, in order to aid thermal insulation, a layer of at least 3m is necessary. Due to aforementioned disadvantages regarding the lunar regolith construction and the fact that a sufficient insulation can be achieved without regolith, the proposed lunar regolith tunnel is not necessary.

Finally, with respect to the radiation protection, no regolith tunnel is necessary as the inflatable structure is already sufficient to block the radiation in such an extend that a yearly dose of 8.3rad/yr is present within the habitat. This lays 19% below the maximum annual dose. The final distribution of ionising dose is given in Figure 3.18. The seventh layer indicated in the graph corresponds to the pressurised habitat interior.

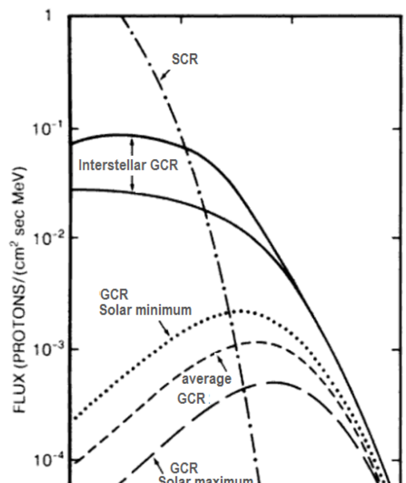
The regolith tunnel was considered as there were concerns that inflatable structure cannot provide sufficient protection against the lunar environmental hazards. However, during the design phase it became evident that all subsystems regarding environmental protection are sufficient in their shielding without the employment of a lunar tunnel. Therefore, the self-standing structure is chosen to be more suitable due to less complexity and a lower cost.

3.3. Habitat Active Environmental Protection

The passive environmental protection is not able protect against all hazards of the Moon. For these the active environmental protection is used. This section starts of with active radiation protection, followed by active thermal control and atmospheric control.

Active Radiation Protection

The GCR entering the habitat through the walls designed for the pressurisation and the thermal and micro-meteorite protection, will not be harmful to the astronauts, as the structure takes 6.1rad and the inside 6.4rad (0.384Sv). These values follow from an



analysis run in SPENVIS without adding any layers for radiation protection. Therefore the radiation shielding is only designed for SPE circumstances. SPE radiation mainly consists of protons. Protons have a positive charge and can, therefore, be deflected by magnetic fields. In this section, the possibility of using electromagnets for protection against SPE's is explored.
The challenge in this is to design the electromagnets strong enough to deflect the protons, and not too strong so that the magnetic field strength is too high inside the habitat. A magnetic field can cause systems to malfunction if the magnetic field strength is above 1mT[6]. Therefore it is desirable to keep the magnetic field strength below this number. This means that the electromagnets used in the radiation shielding will have to be outside or in the habitat walls. The latter is preferred because that reduces the amount of shielding area that needs to be covered by the electromagnets.

To determine the magnetic field strength required to deflect the particles, their momentum has to be determined using Equation 3.18 [19]. This equation uses the particle energy (E), the speed of light (c) and the mass of the particle (m_0). As seen in Figure 3.19 the maximum energy of the protons in an SPE is below 10^3 MeV, which is equal to $1.609*10^{-10}$ J per particle. The speed of light is 299792458 m/s and the mass of a proton is 1.6726219e-27 kg.

$$p_1 = \sqrt{\frac{E^2}{c^2} - m_0^2 * c^2} \tag{3.18}$$

This results in an initial momentum (p_1) of 3.66e-38kgm/s. For the particle deflection, it is also important to determine the angle of deflection required. For this design, it was chosen to have the electromagnets mounted directly on the pressurised shell of the hard-shell module. This again for the amount of shielding area and to make sure the electromagnets are not damaged by any micrometeorites. In this case, the protons will have to be deflected in such a way that they will not enter the living space of the astronauts, as shown in Figure 3.20.

With the simple trigonometric equation stated in Equation 3.19 this angle (α) can be computed to be 1.266rad. Subsequently, α can be used to calculate the momentum increment (dp) required to obtain this deflection angle with the use of Equation 3.20.

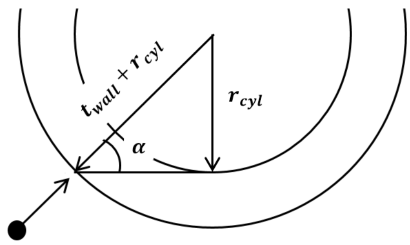


Figure 3.20: Minimal deflection angle (α) proton moving through habitat wall.

$$\alpha = \arcsin\left(\frac{r_{cyl}}{r_{cyl} + t_{wall}}\right) \tag{3.19}$$

$$dp = p_1 * tan(\alpha) \tag{3.20}$$

This momentum increment is equal to the force over time that needs to be applied to the proton to cause the desired deflection (F_L), as is demonstrated by Equation 3.21. This F_L is related to the magnetic field strength (B), the initial velocity of the particle (v_1) and the charge (q), which in the case of a proton is

[6]https://www.supermagnete.de/eng/faq/What-is-the-safe-distance-that-I-need-to-keep-to-my-devices [Cited: 19-06-2017]

1.60218e-19 C. This relation is shown in Equation 3.22. This v_1 can be determined using Equation 3.23 [19].

$$dp = \int_0^t F_L dt \qquad (3.21) \qquad F_L = qv_1B \qquad (3.22) \qquad v_1 = \sqrt{p_1^2/(m_0^2 + p_1^2/c^2)} \qquad (3.23)$$

Then finally to calculate the magnetic field strength necessary to deflect the protons in the SPE can be computed with Equation 3.25. This is with use of the time it takes the proton to move straight through the wall, which was calculated using Equation 3.24.

$$t = t_{wall}/v_1 \qquad (3.24) \qquad B = \frac{dp}{qv_1t} \qquad (3.25)$$

All these computations resulted in a required magnetic field strength of 25.3T. This number will not be entirely accurate, because for this computation it was assumed that the magnetic field the electromagnets will produce is constant, which is not the case. However, the error resulting from this assumption is countered by the fact that the proton will meet the magnetic field of the shield before it passes the electromagnets and that the magnetic fields of adjacent electromagnets will strengthen total magnetic field.

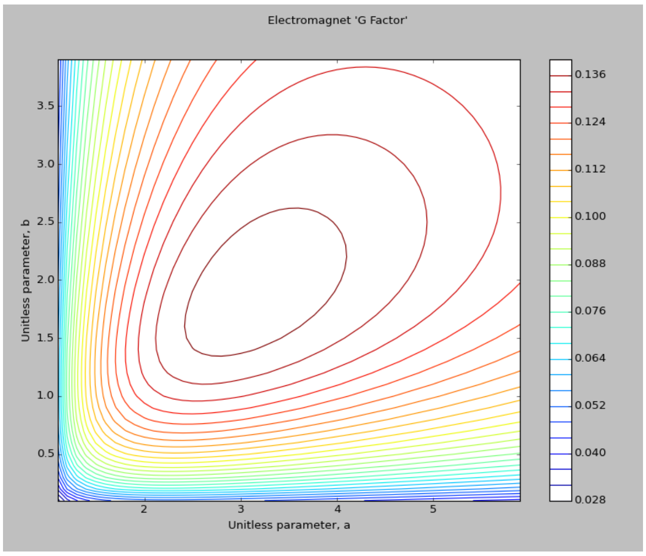


Figure 3.21: Color plot of the G-factor w.r.t. unitless coil dimensions (a,b).

The electromagnets in the shield will have be sized for the 25.3T by optimising their dimensions, power supply and the materials used. To determine the optimal size of the coils for the highest B, the G-factor (G) was maximised. This G-factor is calculated with Equation 3.26[7], which uses the unitless parameters a and b, which are the outer radius over the inner radius and the coil length over the outer diameter, respectively: $a = r_2/r_1$, $b = l/(2r_1)$.

$$G = \sqrt{\frac{1}{8\pi b(a^2-1)}} * 2b * \log\left(\frac{a+\sqrt{a^2+b^2}}{1+\sqrt{1+b^2}}\right) \qquad (3.26)$$

Figure 3.21 shows a plot of the optimisation process, which resulted in $a = 3.096$ and $b = 1.8623$. With a minimal inner radius of 1mm that will mean an outer radius of 3.1mm and a coil length of 3.75mm. With these dimensions the produced B can be calculated with Equation 3.27[7]. This equation utilises the power in the coil (P), the packing factor (F), which is 0.75[7] for the best achievable case, r_1 and two material properties. Of these properties the μ_0 is the permeability of the core of the coil and ρ_s is the resistivity of the material used for the coil windings.

[7]http://nbviewer.jupyter.org/github/tiggerntatie/emagnet-py/blob/master/solenoids/solenoid.ipynb [Cited: 20-06-2017]

$$B = \mu_0 * G * \sqrt{\frac{P * F}{r_1 * \rho_s}} \tag{3.27}$$

For the core nanoperm is used, because it has a relatively high permeability and a low density, but the permeability is not that high, that it will cause problems inside the habitat, due to the high magnetic field strength entering. Nanoperm has a permeability of 1.0e-1H/m and density of 7350kg/m^3[8]. For the windings copper is used, because it has a very low resistivity of 1.68e-8Ωm[9] and is relatively inexpensive. These material decision were made after several iterations with different materials. This will result in a power of 7.416e-05W per coil. This number will have to be multiplied by the amount of coils in the wall to determine the total power required for SPE shielding.

Equation 3.28 and Equation 3.29 are used to determine whether the magnetic field strength ($\sqrt{B_x^2 + B_r^2}$) produced outside these coils at a certain radial position (r) and an axial position (x) will not harm the systems inside the habitat. These equations both include two integrals, which are represented in Equation 3.30 and Equation 3.31. The m in these equations is equations is equal to: $m = (\frac{(4*r/r_2)}{(1+r/r_2)^2+(x/r_2)^2})$[10].

$$B_x = \frac{I * \mu_0}{2r_2} * \frac{E(m) * (1 - (r/r_2)^2 - (x/r_2)^2)/((1 + (r/r_2))^2 + (x/r_2)^2 - 4 * (r/r_2)) + K(m)}{\pi * \sqrt{1 + (r/r_2)^2 + (x/r_2)^2}} \tag{3.28}$$

$$B_r = \frac{I * \mu_0}{2r_2} * (x/r) * \frac{E(m) * (1 + (r/r_2)^2 + (x/r_2)^2)/((1 + (r/r_2))^2 - 4 * (r/r_2))) - K(m)}{\pi * \sqrt{1 + (r/r_2)^2 + (x/r_2)^2}} \tag{3.29}$$

$$K(m) = \int_0^{\pi/2} (1 - m * \sin(t)^2)^{-\frac{1}{2}} dt \tag{3.30}$$

$$E(m) = \int_0^{\frac{\pi}{2}} (1 - m * sin(t)^2)^{\frac{1}{2}} dt \tag{3.31}$$

It turns out that with the coils described the magnetic field strength is already below the critical value of 1mT at a distance of 4.08cm from the coil. This means that no harmful magnetic field strength will enter the inside of the habitat.

Now that it is clear that the single coils have the desired properties, the next step is to determine the amount of coils required to build the shield. As stated earlier, the influence on adjacent coils on the magnetic field strength at the deflecting coil is neglected. This was mainly due to time constraints. This means that the entire surface area of the hard-shell will have to be covered in coils.

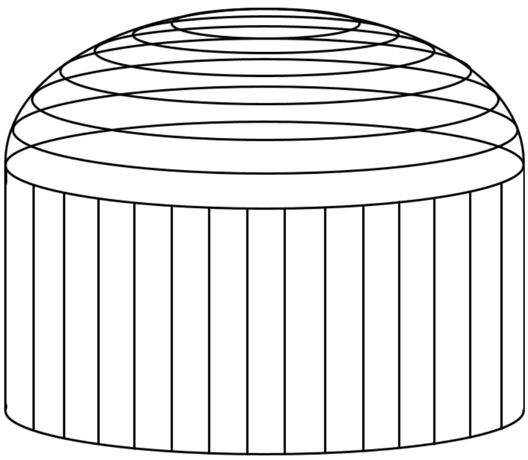


Figure 3.22: Magnetic field lines on the hard-shell module (schematic).

[8]http://www.magnetec.de/en/nanopermr-products/technical-data-nanopermr/ [Cited: 20-06-2017]
[9]https://en.wikipedia.org/wiki/Electrical_resistivity_and_conductivity [Cited: 20-06-2017]
[10]http://nbviewer.jupyter.org/github/tiggerntatie/emagnet-py/blob/master/offaxis/off_axis_loop.ipynb [Cited: 21-06-2017]

Figure 3.22 shows a schematic of the lines the axes of the coils follow, which coïncide with the magnetic field lines. This layout of the shield will result in a deflection to the side if a proton hits the cylindrical part of the hard-shell and up or down (depending on the direction of the magnetic field) if it hits the elliptical cap of the hard-shell.

The amount of coils in the shield in the cylindrical part of the hard-shell are easily calculated by dividing the circumference of the cylinder by the outer diameter of the coils and multiplying that by the height of the cylinder divided by the length of each coil. This came to 1.771e6 coils in the cylinder.

For the elliptical cap this is slightly more complicated. To calculate half the circumference of the ellipse, which is the length of the cross section of the cap (C_{cap}), Equation 3.32 is used.

$$C_{cap} = \frac{1}{2}\pi(3(r_{cyl} + h_{cap}) - \sqrt{10 * r_{cyl}h_{cap} + 3(r_{cyl}^2 + h_{cap}^2)} \tag{3.32}$$

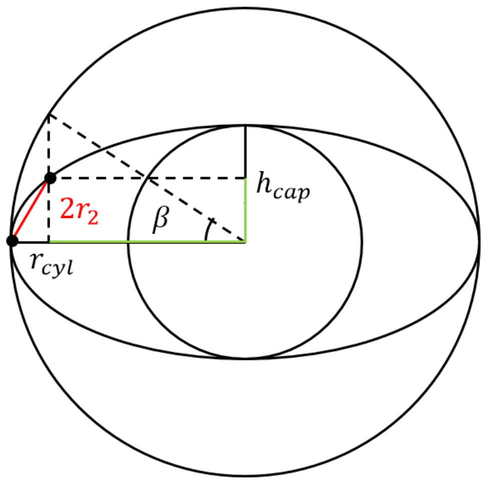


Figure 3.23: Angle, distance relations for equally distanced points on a ellipses.

With the use of the trigonomy relations shown in Figure 3.23, a relation can be set up that allows to determine the radius of each circular row of coils in the cap. This relation is based on the fact that the radius of the circle (horizontal green line) is equal to $r_{cyl}\cos(\beta)$, that the elevation of the circle is equal to $h_{cap}\sin\beta$ and that the distance to the previous point is equal to $2r_2$.

In Figure 3.23 the first and second circle of coils are represented. For the first coil $\beta = 0$. This knowledge can be used to determine the next angle with Equation 3.33. Then, using all radii obtained form these angles till the top of the ellipse, the total coil length required in the cap is calculated, represented in Equation 3.34

$$(2 * r_2)^2 = (h_{cap}\sin(\beta_{i+1}) - h_{cap}\sin(\beta_i))^2 + (r_{cyl}\cos(\beta_i) - r_{cyl}\cos(\beta_{i+1}))^2 \tag{3.33}$$

$$l_{cap} = \sum_{i=0}^{n} 2\pi r_{cyl}\cos(\beta_i), \qquad n = \frac{C_{cap}}{r_2} \tag{3.34}$$

These calculations resulted in a total of 3.389e6 coils in both the cap and the cylinder. This results in an electromagnetic shield mass of 2.579e3kg and a total power of 251.4 W (P_{tot}) required while the shield is active. The final consideration for the design of the shield is the amount of heat generated by the coils. This heat (Q) is obtained with Equation 3.35, which is a multiplication of the emissivity of copper (ϵ_{Cu}), the Boltzmann constant ($/sigma_B$), the radiating surface area of the shield (A_{shield}) and the temperature (T_{shield}).

$$Q = \epsilon_{Cu} * \sigma_B * A_{shield} * T_{shield}^4 \tag{3.35}$$

$$T_{shield} = P_{tot}/(C_{Cu} * m_{Cu}) * t_{SPE} \tag{3.36}$$

The temperature (T_{shield}) depends on the P_{tot}, the specific heat of copper (C_{Cu}), the total mass of copper in the shield (m_{Cu}) and the duration of the SPE (t_{SPE}). It can be seen from this that for short durations the shield is not going to add that much heat. However, any time increase will result in a heat increase to the power four. Therefore it is vital that the shield is constantly cooled when it is active so that the amount of cooling required is minimised.

RAMS Characteristics and Sensitivity Analysis
Active radiation shielding has a TRL of 2. This means this field still requires long extensive research before it could be used in our habitat. Therefore, comparable systems will be used to estimate the RAMS-characteristics.
The reliability of particle accelerator magnets is very much depended on the reliability of their water piping (cooling), the type of power connections used and the manufacturing reliability [20]. For the cooling, the piping reliability is higher if suitable materials are used [20]. Also, our shielding will only be active during SPE's, this decreases the amount of cooling required. This will also decrease the chance of the pipe leakage. It is adviced to use louvred or bayonet joints to increase the power connection reliability. The connections should also be inspected regularly and before activating the shield to avoid failure [20]. The manufacturing reliability can be increased by performing test in different stages of the manufacturing process [20].
The availability of solenoids is high. However, the coils for this design are at the minimum bound of a radius. This will lower the availability. Furthermore, nanoperm is included in the design, which is a relatively new material. This rapidly quenched iron based alloy with a fine crystalline microstructure could decrease the availability of the shield. As mentioned before special attention should be paid to the power connections routinely and before every activation of the shield. The coils should be accessible when the shield is not active, to enable replacements in case of defects. Furthermore, there should be sensors in the wall to detect any leakage of coolant to avoid failure due to overheating and enable quick repairs.
For the safety, it is vital for this shielding to be researched more. In this design, a number of phenomena are neglected. Therefore, this design is more of a conceptual feasibility study than an actual detailed design. For this final design, models should be made to quantify the amount of secondary radiation created while the particle moves through the wall and the impact this radiation has on the radiation dose inside. Furthermore, the interaction between the magnetic field of the coils and the irregularities in the coils field should be included in the model. This design also assumes that the coils are everywhere, so also in the doors. This would mean that due to the magnetic field in the wall the airlocks can't be operated while the shielding is active. This should be made clear in the safety protocols or the airlock design would have to be altered, such that it is not harmed by the magnetic field.

The shielding is dependent on the energy of the particles and the deflection angle required. An increase in the particle energy or deflection angle will increase the B required to deflect the particle. This increase can be met by the design in four ways: increase the wall thickness, increase the power, increase the permeability by changing the core material, or decreasing the size of the coil. The last is not an option because the coil is already a minimum dimension. There are materials that have a better permeability than nanoperm, however, this will increase the distance from the coil that still contains a harmful magnetic field strength. This will decrease the area for systems inside the habitat. The same holds for increasing the wall thickness. That leaves increasing the power. This will increase the amount of cooling required and in extreme cases the number of power trucks that have to be sent up.

Recommendations
The Shell has to be protected from SPEs. This can be done by passive radiation protection. However, it turns out that this would increase the mass to an infeasible level. The other option is to use active radiation protection which is estimated to be much lighter. The problem with this solution is that the technology for this type of shielding is not ready yet and even though it seems like a feasible option it is possible that the technology won't develop enough to be able to use it in the habitat. If neither of these problems can be solved in the near future, requirement [MR-MT] regarding the 1-year stay will become a killer requirement and the duration would have to be reduced to ensure the astronauts' long-term health. For this report, it is assumed that the active radiation shielding can be developed in time to meet the requirements set.

Active Thermal Control System
The excessive heat needs to be transported outside through the active thermal control. The design of this thermal control is based on the design of the ISS [21]. The active thermal control consists of an Internal Active Thermal Control System (IATCS) and an External Active Thermal Control System (EATCS).

In the ISS both an IATCS and an EATCS is present. The IATCS use water as a coolant and cools the systems and especially experiments inside the habitat that have a certain operational temperature. The EATCS cools the complete habitat of the remaining heat flow going inside the habitat and the heat coming from the systems and the astronauts. It consists of cold plates in the structure walls, outside wall of the living area, that is connected to ammonia loops. The choice of using ammonia is made because it has a very low freezing point, namely $-77\,°C$, so when the temperature drops at night the coolant will not freeze. The loops cannot be inside the living area because this is dangerous for the astronauts. Because the hard-shell is the first period of time alone, it needs to be able to at least cool the heat coming from all astronauts, from the communication systems and the personal computers.
For the habitat of LEAP, it can be assumed that no research is being done in any of these modules. Besides, the systems that are present in the modules have their own cooling system and will operate at room temperature. Because of this, the IATCS how it is designed for the ISS is not applicable to the habitat of LEAP.

However, for the inflatables, it will not be possible to have cold plates inside the structure, because it is a fabric type of structure. In section 3.2 the inflatable structure will be explained. Thus, for the inflatables, cold plates will need to be inside the pressurised area. This also means that the coolant used can only be water. These cold plates can be attached to existing systems.

To size the active thermal control, it is needed to know what the maximum heat generated from the inside is. The heat coming from the astronauts will be 576W and for the communication system and for the laptops the heat dissipation is estimated to be 900W (equals 10 laptops). For lamps and other miscellaneous systems, an estimation has been done of 100W per module, since this is a little bit more than the heat coming from one laptop[11]. This miscellaneous could entail systems for entertainment (beamer, screens) or heat from other unexpected systems. It should be noted that the values of heat generation are taken from existing systems, like laptops. In the future, special systems for space missions will be developed, that will have less heat production.

In Table 3.8 it can be seen how much heat needs to be dissipated, more explanation on the heat coming in the inflatables can be found in Equation 3.2.

Table 3.8: Heat rejection.

Module	**Max income [W]**	**Exerted by humans [W]**	**Exerted by systems [W]**	**Miscala-neous [W]**	**Max heat to be rejected [W]**	**Max to be heated [W]**
The Nest	63	576	900	100	1639	162
The Hive	63	576	900	100	1639	162
The Shell	-75	576	900	100	1501	165

To ensure the safety of the astronauts and the habitat, it is important to have redundant systems. Because the loops will be located partly outside the meteorite protection, there will be three loops configured in total, each on every side. Each loop is connected to two modules. The EATCS will work optimal if all three the loops are operational, but if one of the loops is damaged, the habitat is still able to sustain.

The maximum heat the loop for the hard-shell should be able to reject is that all the astronauts are in that module and all the systems there as well. This means that the maximum heat rejected for the hard-shell is 1501W. Afterwards the inflatables and their active control system will be added. The other two loops should cool at maximum 1639W each. With these budgets, the sizing of the different components of the active thermal control can be done based on the sizing and performance of the thermal control ratios of the ISS [21].
The outcome of the masses is given Table 3.9. The different components belonging to the active thermal control are also given in this table. The Interface Heat Exchanger (IHE) connects the internal loops

[11]https://support.lenovo.com/nl/nl/solutions/pd008989

Table 3.9: Masses of Active Thermal Control.

System Part	Mass loop The Shell [kg]	Mass loop The Nest [kg]	Mass loop The Hive [kg]
Interface Heat Exchanger	82.6	82.6	82.6
Cold Plates	65.9	57.6	76.6
Radiator	72	78.9	78.9
Radiator Beam Valve module	22.7	22.7	22.7
Thermal radiator rotary joint	54.1	59.1	59.1
Pumps	154	154	154
NH_3 tank (included NH_3)	10.9	11.9	11.9
N_2 tank (included N_2)	6.7	7	7
Total	519.1	524.0	543.0

with the external loops, two per loop are needed, one for redundancy. This heat exchanger also consists of a heater which is possible to heat the water. The total heating power of the heat exchanger is 1.8kW. This will be enough for the heat losses through panels, floor and pins together. The maximum energy needed for heating is given in Table 3.8. During cooling the heat exchanger transports the heat from the internal loops to the external loops. It also cools the water so it is able to take up the heat again. The temperature control inside the heat exchanger is also able to regulate the heat coming from and going away from the heat exchanger. This way a desired temperature in the habitat can be obtained.

The DDCU (Direct Current-to-Direct Current Converter Units) and MBSU (Main Bus Switching Units) are cold plates (CP) attached to the structure of the pressurised wall of the habitat to cool the habitat. In total five Cold Plates are used per loop. Since in The Hive sporting facilities are present it is decided to have 3 Cold Plates of every loop in there and two in the other connecting modules. For the third loop (red) it is assumed that 3 cold plates are situated in the shell and two in the nest, see Figure 3.24. This is also where the difference of mass comes from in Table 3.9. The Radiator Valve Module (RVM) is able to close of the ammonia from one loop going into the radiator. This can happen for example when the radiator has a defect or a leakage. Again two of these valves are used for redundancy.

The radiator is the area from which the heat is released. The area of the radiators of the inflatables is $5.6m^2$ and the area of the hard-shell is $5.1m^2$. Very fine tubes run through the surface of the radiator filled with the warmed up ammonia from inside the habitat. The thermal radiator rotary joint makes sure the radiator is able to turn to or away from the Sun. To release heat the radiator has to be pointed towards the Sun. The top of the radiator will be coated with highly emissive and low absorbing paint, like as done with the habitat. This way the plates do not heat up and the heat from the ammonia is able to be expelled. If no heat needs to be expelled from the habitat the ammonia will get very cold from the outside. To avoid the ammonia from freezing the radiators are able to retract and if needed the Heat Exchanger can warm the ammonia up a little. The placement of the radiators can still change. Depending on attitude accuracy while landing the radiators can be placed on the equatorial line. Doing it in this way, half of the time the radiator is in the shadow of the habitat and the other half of the time it is able to turn with the sun. The length will then be shorter than the elongated part of the hard-shell or for the inflatables, shorter than half of the height above ground of the inflatable. The width will be adapted according to that.

The loops will also be able to close of one module with the use of valves. For example, in the first stage, when only the hard-shell is on the Moon, the loop will only be cooling/heating the hard-shell. The inflatable can be attached to it afterwards. If one module has depressurisation that module can also be closed from the loops. All three modules will have their own tanks for ammonia and nitrogen. Nitrogen is needed to keep the pressure inside the tubes. The mass of the tank includes the meteorite protection needed.

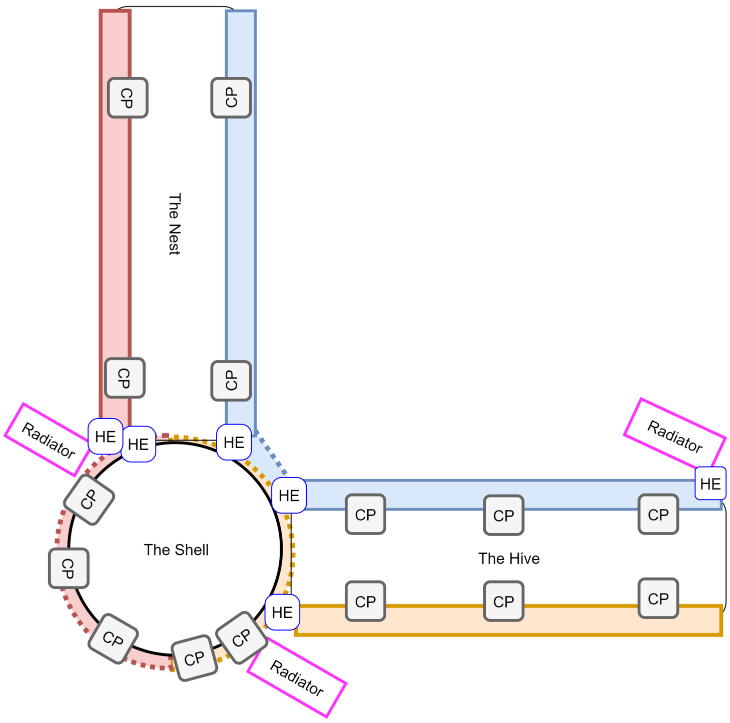


Figure 3.24: Layout Active thermal control. Dotted line means ammonia loop and continues line is water loop.

In Figure 3.24 the layout is given of the active thermal control showing the cold plates, the ammonia loops (dotted line) and the internal loops. The three different colours show the three different loops all connected to their own radiator. This radiator is able to rotate completely. When the Sun is on the other side of the habitat it stands in the shadow of the habitat. Especially for The Shell the peaks of heat rejection needed is at 45 degrees. The radiator should then thus be in the shadow.The valves and tanks are not given in this layout. The tanks will be attached to the outside of the hard-shell. The valves are placed as explained before.

RAMS Characteristics and Sensitivity Analysis
During the design of the thermal control subsystem, the reliability has played a big role. If the thermal control fails, the situation for the astronauts can get dangerous very fast. The chosen MLI has had a lot of research and is proven in space flight. The active thermal control is based on the thermal control used in the ISS, but then downscaled. This is also proven in manned space flight and thus a reliable option. For the calculations, all maximum possible values are taken into account and redundant system is implemented for the Active Thermal Control. This also contributes to the reliability of the system.
Since the materials have been used a lot, the availability on Earth is sufficient. Though the insulation can not be 3D printed yet, so the panels need to be brought from Earth. So the availability on the Moon is limited.
Maintenance, on the other hand, is more difficult, because the insulation is situated inside the Whipple shield and it is not easy to reach. However, when the Whipple shield will be maintained or replaced the MLI can also be replaced if needed. For example, if a large meteorite hits the shell and the MLI is damaged, this will also mean the Nextel and Kevlar are damaged. When those are replaced the MLI can also be replaced. The Active Thermal Control will need regular checks, but these are more reachable.
The materials used inside the habitat for thermal control are safe for the astronauts. It is deliberately chosen to put the ammonia only outside the habitat so the safety of the astronauts is not endangered. The use of three loops in the thermal control makes sure that two modules can be kept operational if one of them fails.

The design of the thermal control is mainly based on the heat coming from the Sun. If the habitat would move from the equator, this would decrease the energy coming from the Sun. When this happens, MLI

layers have to be removed or added, so more heat of the Sun will get in. Another aspect that will differ the design of the thermal control is the angle of the inflatables with respect to the equatorial plane of the Moon. This will depend on the way the hard-shell lands. It will change the A_p/A_r. At lunar night and when the Sun is right above the habitat, this ratio does not change with the angle. This means that the maximum heat flow in and the maximum heat flow out will not change.
If the heat coming from the astronauts and from the systems increases the active cooling system will become heavier. Nevertheless, during the design process of this system the maximum cooling, that would be needed if all systems are working and one of the loops is broken, is used. that this amount of cooling would be needed for one loop is very unlikely so there is a margin.

Recommendations
As mentioned before, more research has to be performed on how the regolith takes up and emits heat, this will influence the heat that will be rejected or taken up through the floor of the habitat.
In addition to this, research should be performed to the heat storage capabilities of lunar regolith. There have been done several studies on how heat could be stored in lunar regolith. By storing heat in lunar regolith the natural environment of the habitat can be used and less power needs to be generated. This sounds like a money and mass saving idea. Nevertheless, the studies did not find an efficient way of realising the idea yet [11][22].
For the active thermal protection, the detailed design of the valves and the radiator still needs to be performed and optimised to make it as light as possible.

Atmospheric Control System

The habitat shall have an atmospheric pressure of 101235Pa, which is equal to the atmospheric pressure on Earth. This pressure was selected to lower the strain on the astronauts' health during the one-year stay on the Moon. The air composition in the habitat shall be 20% O_2 and 80% N_2, which is also similar to that on Earth and a humidity between 40% and 60% [23]. The atmospheric control system shall maintain this pressure and air composition inside the habitat as well as filter the air of contaminants, control the humidity and airflow and detect and suppress fires in case these occur.

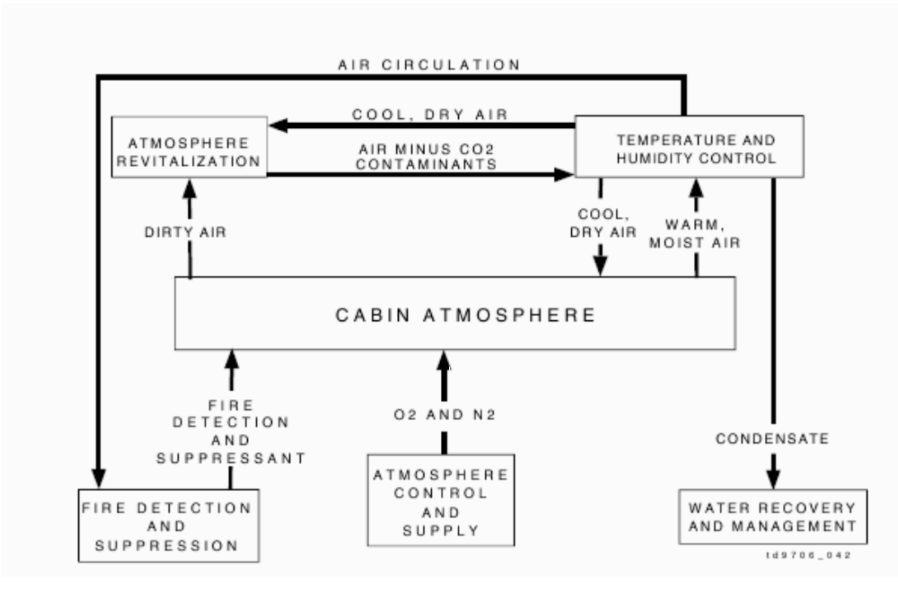


Figure 3.25: Atmospheric control system interfaces [24].

In Figure 3.25 the four systems that form the atmospheric control system are displayed with their interrelations and how they influence the bioastronautic system that was defined in chapter 5.

The atmosphere control and supply system block in Figure 3.25 contains the Pressure Control Assembly (PCA) and the Oxygen Generation Assembly (OGA) [24]. The PCA consists of pumps, valves and tanks. These tanks are filled with N_2, which is used to counteract any pressure losses due to leakage. This

leakage consists of airlock losses, which is about 10% of the airlock air mass for each operation and general module leakage, which is about 0.18% of the habitat air mass [3].
The oxygen generation by the OGA can be done by electrolysis or photosynthesis. Experiments in microgravity have shown that plants cannot ensure a reliable source of O_2 for long term use [25]. Further research would be required to see how the plants react to the lunar environment and to develop a reliable system that can utilise their photosynthesis in the lunar base [25]. In section 15.1 this will be further elaborated on. The OGA, therefore, consists of fuel cells that will perform electrolysis to turn water into O_2 and H_2. The astronauts each consume 0.84kg O_2 per day. This consumption together with a loss of 0.55kg per day will put a minimum constraint of creating 3.91kg per day on the OGA [3]. The water to be used for this generation is not included in the atmospheric control, but in the bioastronautic system of water management and described in chapter 5.

The atmosphere revitalisation system block in Figure 3.25 contains the Trace Contaminant Control Subsystem (TCCS), the Carbon Dioxide Removal Assembly (CDRA) and the Major Constituent Analyser (MCA) [24]. The TCCS filters volatile gasses, which are produced by systems or the astronauts metabolism, except for CO_2. The CO_2 is taken care of by the CDRA. Although not shown in Figure 3.25 the CDRA feeds the collected CO_2 to the Sabatier of the bioastronautics system. The MCA finally monitors the composition of the air, which is will be used to manage the OGA, TCCS and CDRA.

The temperature and humidity control system block in Figure 3.25 consists of the Common Cabin Air Assembly (CCAA), the Avionics Air Assembly (AAA), the Intermodule Ventilation (IMV), the High Efficiency Particle Atmosphere (HEPA) filter and the lunar dust filters [24]. The CCAA handles intramodular ventilation and removes moisture from the airflow to obtain a humidity between 40 to 60% [24]. The condensed moisture will then be fed into the water management of the bioastronautics system. The AAA is the electronics system that serves as an interface for the CCAA. The intramodular ventilation is aided by the IMV, which joins all the ventilation systems to create a pleasant atmosphere in the entire habitat. Before the air is blown through the ventilation it is filtered by the HEPA filters and the lunar dust filters.
The lunar dust is filtered using the Lunar Air Filtration with Permanent Magnet System (LAF-PMS). The permanent magnets in this filter will collect the lunar dust, which is known to be of high in Fe-content [26]. The LAF-PMS will have to be cleaned once a week by placing a magnet between the magnets in the filter that will release the dust so that it can be disposed of [26].

Finally, the fire detection and suppression system block contains smoke detectors and Portable Fire Extinguishers in each module [24]. If a fire is detected, the CCAA and IMV will temporarily be shut down and the astronauts will be notified. This all to ensure the fire can be extinguished fast and effectively.

All these systems were sized based on systems of the ISS and the systems envisioned for the Transhab, which were linearly interpolated with respect to the habitat volume [12]. The final dry masses are given in Table 3.10.

Table 3.10: Masses of atmospheric control per module.

	The Shell	**The Nest**	**The Hive**
Mass (kg)	197.19	398.99	398.99

RAMS Characteristics and Sensitivity Analysis
The majority of this atmospheric control design has worked reliably on the ISS for many years. There only two subsystems that are not used on ISS. These are the lunar dust filter (LAF-PMS) and the PCA[12]. The LAF-PMS has been tested on Earth and it is expected that it could work on the Moon without the HEPA filters [26]. However, to increase the reliability the atmospheric control will still be equipped with both. The pressurisation tanks of the PCA are STS based and have operated reliably for years on Earth now and are expected to do the same on the Moon[12]. The pressurisation valves of the PCA are the same as designed for the Atmospheric Revitalising and Pressurisation Control System (ARPCS) of the X-38 of NASA[12], which was in the drop test phase when it was aborted due to lack of funding.

[12]http://salotti.pagesperso-orange.fr/lifesupport3.pdf [Cite: 09-06-2017]

All these systems again except for the LAF-PMS are expected to be readily available since they are operational either on Earth or on ISS. The LAF-PMS is a fairly simple system and therefore it is also expected to be available.
There are five subsystems in the atmospheric control system that include filters. These are the TCCS, CDRA, CCAA, HEPA and the LAF-PMS. The maintenance of these subsystems will mostly consist of cleaning or replacing those filters when they are nearly quenched [24] [26]. This means that each the filters will either be equipped with a sensor to alert maintenance is required or they will be very easily accessible for regular checks.
For safety precaution, each module will have its own atmospheric control system that can operate autonomously from the other modules. In case the wall of a module is punctured and the atmospheric conditions of the module are lost, the system can be shut down and the airtight connection doors can be locked without influencing the atmospheric conditions in the other modules.

The atmospheric control system scales with respect to the pressurised volume of the habitat and the number of astronauts; If the volume is increased the system size will increase. It is assumed that this sizing is linear. This will lead to a slight overestimation if sized to be in a larger volume than the reference system and a slight underestimation if sized to be in a smaller volume. This is due to the fact that each system requires certain components and these cannot be reduced in size infinitely. Further, if there is an increase in the number of astronauts that will stay in the habitat the OGA and CDRA system will have to be increased to account for the raised O_2 consumption and CO_2 production.

4
Structural Subsystem Design

The explanation of the design of the habitat and its three modules starts off with the discussion of the structural subsystem. The structural subsystem has to house all elements of the other subsystems, give them their adequate spacing and create enough volume and area for the astronauts to live in. Next, it is an integral part of the environmental protection system. The structure acts as a pressure shell, preventing the atmosphere from escaping, and serving as part of the radiation and meteorite protection system (chapter 3). Finally, the structural system shall perform all of these actions, for the period of ten years, as required in [MR-LT] (chapter 14). Ideally, the structure sustains all loads without permanent damage or deformation. Therefore, this chapter starts-off with an analysis of the load cases which it will encounter, in section 4.1. In order to facilitate four astronauts and provide safety ([MR-AS] and [SYS-ST-01]), this chapter continues with a discussion in section 4.2 about structural set-ups and a brief recap of the choices made in the Midterm Report [2]. Thereafter, in section 4.3 the structure of The Shell will be discussed. After a brief recap of all design constraints, a baseline structural analysis, as well as an analysis of its functions is presented. Similarly, the inflatable structures are discussed in section 4.4.

4.1. Expected Load Cases

First, all load cases have to be identified, to ensure that the structure of the habitat does not fail. There are two different kinds of loading: static loading and dynamic loading.

During the transportation to the Moon, the payload inside the launch vehicle will endure launch loads, which change with time. The longitudinal accelerations, however, vary slowly with time, and can thus be interpreted as causing quasi-static loads. These quasi-static loads are analysed as static loads in section 4.3. The vibrations which occur due to engine functioning and aerodynamic turbulence are dynamic loads. They have a specific frequency and force, and the payload has to be designed for these vibrations. Since the SLS has not flown yet, no measurements have been performed on the acceleration of the launch vehicle during ascend. From historical data[1], a longitudinal launch load of approximately 4g is expected. During the descent, a load of approximately 1.75g is expected, as can be seen in chapter 10. This means that the descend loads are considered to be insignificant with respect to the ascend loads, and thus the module should be designed for the ascend loads.

The next load to be considered is the static load of internal pressure. This load is caused by the fact that the habitat will be kept at one standard atmosphere of pressure, which is 101325Pa. The difference in pressure with the outside of the habitat, which is in ultra-high vacuum, is therefore also 101325Pa. This is a constant load acting distributed on the inside of the lunar habitat. Since this pressure is present during the whole operational time of the habitat, it means the load will continuously last for at least 10 years. This long-term static loading will cause fatigue in the metal structure[27], which is dependent on the material used. This means that during the design, this fatigue has to be taken into account, in the form of applied safety factors, which will be done in section 4.3. After identifying this internal pressure load, it can already be predicted that the launch loads will be small with respect to the internal pressure load.

Furthermore, the gravitational force of the Moon causes a static loading on the structure. Firstly originating from the structure itself, secondly from everything inside, including the astronauts, being pulled onto the floor of the habitat. This force is considerably lower than it would be on Earth since the gravitational acceleration on the lunar surface is around $1.62 m/s^2$, or 0.16g. As with the launch loads, it can

[1] https://space.meta.stackexchange.com/questions/249/resources-and-references-on-the-topic-of-space-exploration/319#319 [Cited: 12-06-2017]

be predicted that this load are negligible with respect to the internal pressure in terms of designing the outer walls of the hard-shell module. However, when the feet of the hard-shell module are designed, this gravitational force is the leading load case. This will all be elaborated further in section 4.3.

Additionally, stresses can be induced due to the difference in temperature within the material. These thermal stresses are caused by the different expansions of the materials in higher temperatures. Dependent thermal expansion of the materials, this effect can cause large stress concentrations on the borders between illuminated and shaded structure. These stresses, however, are not active on the load-bearing structures of the habitat, since sunlight does not reach these. The outermost layer of the habitat is designed for these thermal stresses, since that layer experiences the high fluctuations due to incoming solar heat flow, and experiences the highest difference in Sun exposed and shaded parts.

4.2. Structural Set-up

The driving factor for the set-up of the structural system is the living area recommendation, which is dominated by bioastronautics (chapter 5) and interior demands (chapter 7). Finding the set-up is then finding the ideal combination between living space generation, and mass and launcher constraints. To meet interior requirements, it was found that a floor area of around 120m^2 is needed at least to ensure that all requirements can be met. Alongside with the conceptual decisions taken in the Midterm Report [2] a minimum of launches is tried to be achieved. From early mass estimates, it was found that one singular hard-shell module can be carried by an SLS and mass-wise, multiple inflatables can be carried by another SLS, if sized carefully.

To fully preserve the advantages of inflatable and hard-shell structures, a combination of the two was chosen. The hard-shell module acts as an anchor point, safety zone and system node for the inflatables, which are docked to the hard-shell module. The aim of this detailed design then is not only to use the full potential of the chosen launch vehicle, but also to find an optimal combination of modules to minimise cost and the number of launches.

Finally, the final design is iterated over configurations and sizes of inflatables. It is found that for a living area of 130m^2 the optimal packing factor for two SLS missions is found. This entails one launch of a hard-shell module and one launch of two inflatables (see chapter 11).

4.3. Hard-Shell Structure

The structure of The Shell is rigid and mainly made of one material, as opposed to the inflatable structures. In this section, it is shown how this module is designed, including the accompanying calculations.

Material Selection

At first, the material for the structure of The Shell has to be chosen. The material selection has to be made carefully, as there are human lives at stake. To withstand such an internal pressure the material has to be strong. Material degradation properties have to be taken into account for a mission duration of ten years. In a space environment a structural material has to further withstand radiation and possibly the impact of micrometeorites. Lastly, for weight preservation, it should be checked if a certain material in a structural subsystem can integrate functions of other subsystems. For aerospace applications generally two material classes can be considered to act as structural components which keep an internal pressure. These are aluminium alloys and composites (such as carbon or glass fibre). Therefore, the following trade-off compares high strength carbon fibre with aluminium 2219-T87 (the rear wall alloy of the meteorite protection shield as described in chapter 3).

From Table 4.1 it becomes evident, that the aluminium is the better option for the structural components, despite its relatively high weight and low strength. The advantages of designing a structure to take over functions of other subsystems results, most likely, in an overall weight benefit too. Hence, from here on all structural considerations are made about aluminium 2219-T87.

Table 4.1: Material selection of The Shell's structural subsystem [10]. The values for carbon fibre are average values.

Material	**Yield Strength [MPa]**	**Density** [kg/m^3]	**Radiation resistance**	**Temp. range [°C]**	**Integrative design**
Carbon Fibre	3875 ^{green}	1820^{green}	Bad^{red}	-273 - 550 ^{green}	Atmospheric Control^{yellow}
Aluminium 2219-T87 [2]	393 ^{yellow}	2840^{yellow}	Good ^{green}	<413 ^{green}	Atmospheric Control, Radiation Protection, Meteorite Protection^{green}

Pressure Shell

To obtain some baseline parameters, the dimensions of an SLS payload bay are analysed. Together with a Whipple shield thickness of 300mm (see chapter 3), a maximum height for docking of 200mm (over a length of 1800mm (see chapter 11), which is a guesstimate of twice the width of an airlock door plus 40mm to each sides), a 100mm wall offset for internal systems, as well as a maximum payload bay radius of 3750mm, an interior radius of 3000mm has been found, creating a little less than 28.3m^2 (see Figure 4.1.

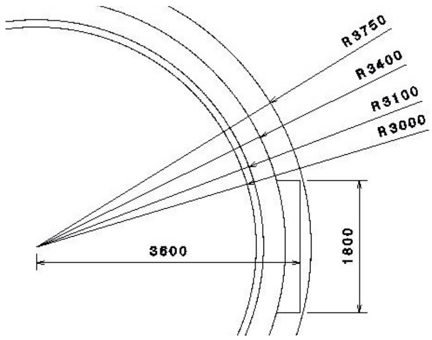


Figure 4.1: Schematic drawing of the sizing of the wall spacing for the hard-shell module. From inner to outer the layers are inner wall, pressure shell and rear wall of Whipple shield, bumper plate of Whipple shield and inner wall of SLS payload bay. The rectangular part represents the airlock and docking system.

To contain the pressure, it is generally advisable to use round shapes rather than shapes with edges, to minimise local stress concentrations and the overall stresses. For simplicity the habitat is modelled to be a cylinder with a spherical endcap on the top and a flat bottom, which can be assumed to be clamped and hence is neglected for this first calculation. The formula for hoop stress [3] can be found in Equation 4.1.

$$\sigma_h = \frac{P_{internal} r_{cyl}}{t_{cyl}} (cylinder) \sigma_h = \frac{P_{internal} r_{sp}}{2 t_{sp}} (sphere) \tag{4.1}$$

It can be seen, that the cylinder is the more critical point, hence this thickness value is used for the first analysis. It can be found that an aluminium shell would need to be ca. 0.8mm thick for an internal pressure of 1atm. It is advisable to use a safety factor to account for material degeneration over the ten year period.

Nonetheless, it can easily be observed that this wall thickness is far below the wall thickness required for meteorite protection which is 6.4mm. As the rear wall of the meteorite protection shield is designed to not be penetrated, it is advisable at this stage to create a shell of the required 6.4mm of structural aluminium, rather tan two separate shells. Therefore the overall wall thickness is found to be 6.4mm.

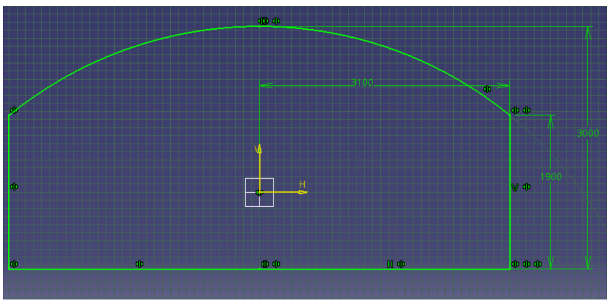


Figure 4.2: Schematic drawing of front cross section of first iteration of hard-shell module.

A first iteration of the structure can be made at this stage. For the modelling of the pressure shell, CATIA is used. A schematic drawing is shown in Figure 4.2. The heights were chosen in a way that the airlocks fit the module and that every astronaut could easily stand upright at the very edges of the inner sides of The Shell. Therefore the structure as

[3]http://www.engineersedge.com/material_science/hoop-stress.htm [cited:19.06.2017]

seen in Figure 4.3 is created initially.

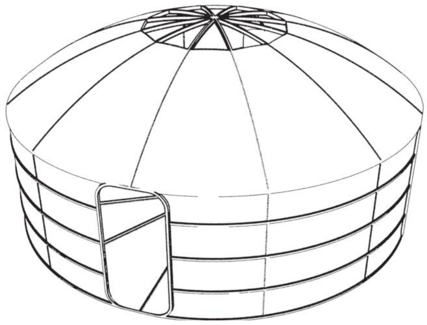


Figure 4.3: Render of first iteration of hard-shell module.

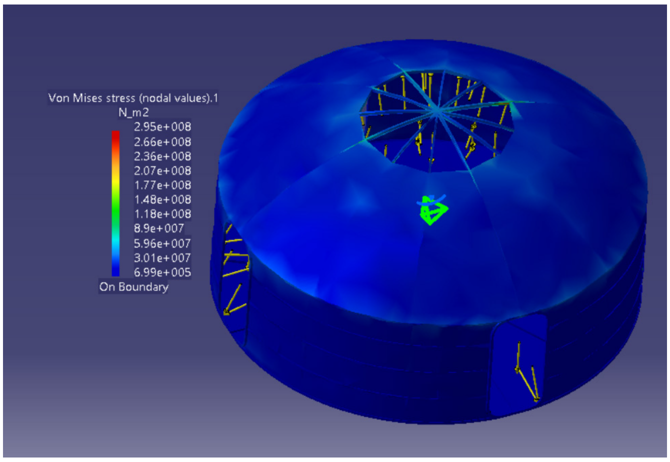


Figure 4.4: Structural analysis results for first iteration of hard-shell module generated in CATIA.

As can be seen, the model needs some cut outs in the structure, for the doors and windows. These places are expected to induce load concentrations and are therefore reinforced with ribs and spars. Whereas the radial ribs are local increments of the thickness to 20mm with a height of 40mm, the spars, on the other hand are modelled as I-beams and their dimensions are guesstimates. The detailed design of these beams has to be performed at a later stage of the design, as the induced vibrations and the mounting of the interior gravely affect their dimensions. In the following models, and also this first iteration, their dimensions are kept constant to give an idea of the weight they induce on the structure, this can be seen in Figure 4.5.

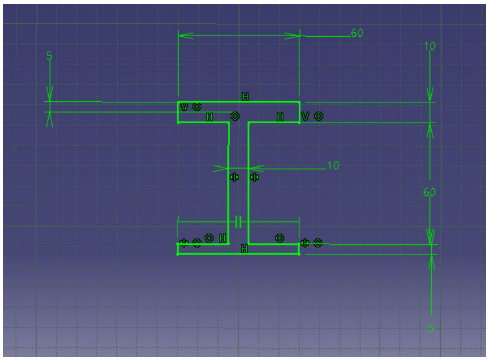


Figure 4.5: Sketch of I-beam (spar) cross section

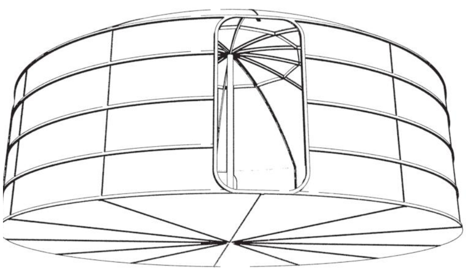


Figure 4.6: Render of second iteration of hard-shell module.

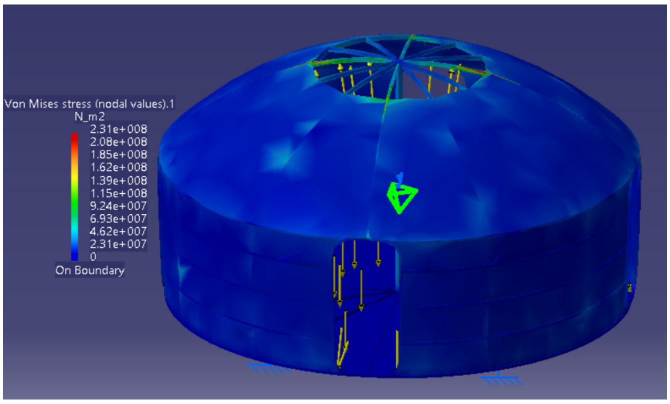


Figure 4.7: Structural analysis results for second iteration of hard-shell module generated in CATIA.

At this stage, a structural analysis can be performed, for the special case of the internal pressure acting on the inner walls. For this analysis, the bottom plate of the habitat is assumed to be clamped, as there is an underlying structure to mount the feet and the deceleration stage for the transportation to the

Moon. The results of this structural analysis can be seen in Figure 4.4.

The analysis shows that there are local stress concentrations at the cut-outs for the windows. These concentrations reach up to 200MPa. This can lead to an issue, looking at the safety factor for this material. Literature suggests that aluminium can lose up to 50% of its yield strength when permanently loaded for ten years [27], implying that a safety factor of two has to be applied. Therefore the stress should not exceed a value of 197MPa anywhere in the structure. For the mass of this structure, a value of 1877kg has been found utilising the CATIA model.
In a second iteration, these problems have been tackled by introducing a central solid pole of 60mm in diameter. This is to take up some of the stresses acting in the upward direction, as can be seen in Figure 4.6. This iteration weighs 1990kg. Even though the overall stress levels drop, the analysis (Figure 4.7) still suggests that there are local concentrations exceeding the limit of 200MPa. Further, the second iteration is the first iteration where the airlocks are integrated into the model.
In a third iteration, the extension of the central pole through struts are investigated. To further reduce the stress in the roof, ribs on the roof are introduced. A schematic drawing of the third iteration is given in Figure 4.8, while Figure 4.9 shows a render of the third iteration. It is found, that the floor needs to be lowered to give space, not only for systems but also to accommodate the pressure cabin of the airlock. Figure 4.10 shows the analysis of the third iteration. It can be observed that the overall stress levels decreased a lot. This module, however, weighs 2265kg.

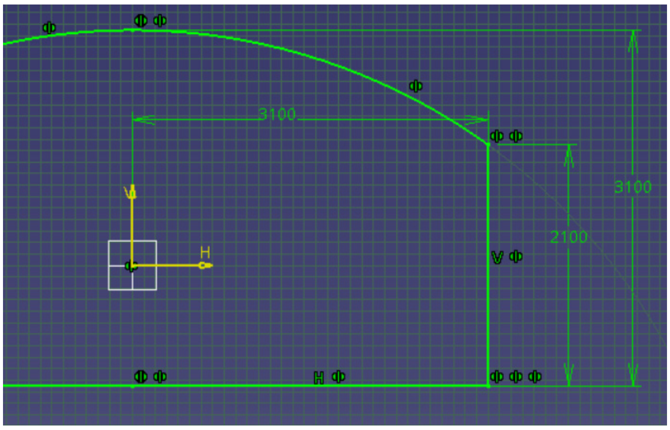


Figure 4.8: Schematic drawing of front cross section of third iteration of hard-shell module.

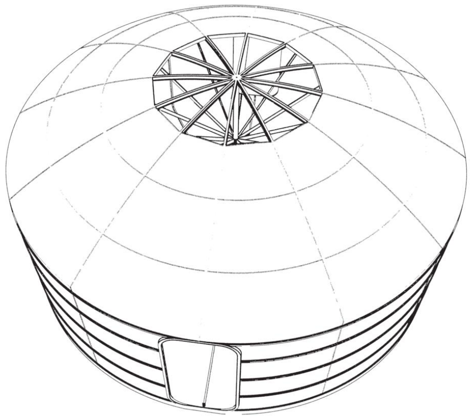


Figure 4.9: Render of third iteration of hard-shell module.

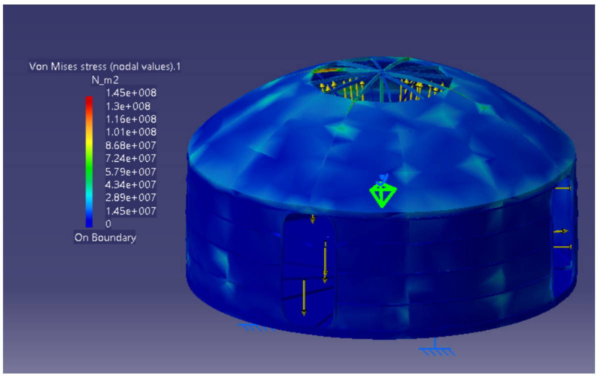


Figure 4.10: Structural analysis results for third iteration of hard-shell module generated in CATIA.

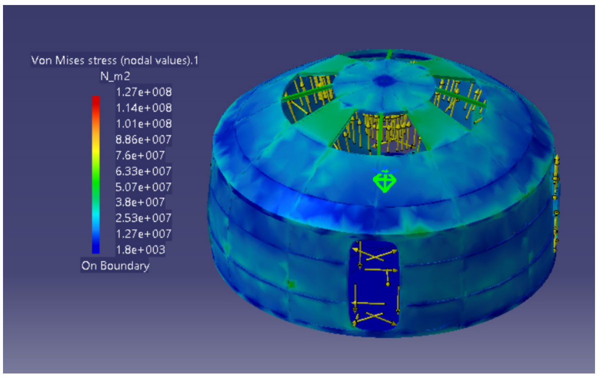


Figure 4.11: Structural analysis results for fourth iteration of hard-shell module generated in CATIA.

In a fourth iteration, also the integration of the meteorite shield is considered. As it is almost impossible to manufacture a continuous bumper plate, it was decided to split the bumper plates up into panels. The ribs and spars offer an ideal mounting possibility for these panels. Figure 4.12 shows how this is implemented. The two most significant changes are the alteration of the window location and the change in geometry for the roof. Figure 4.13 shows a schematic drawing of the geometry. This change is made to ease the assembly and maintenance of the bumper plates of the Whipple shield by making the sections straight, panels can simply be mounted onto pins, which can be seen in Figure 4.14. The window location was changed, to lower stress concentrations on the top, due to the many sharp edges. Additionally, the central pole was removed, as it proved to weigh too much compared to the structural benefits it gives (compare the analysis of iteration 1 with iteration 2).

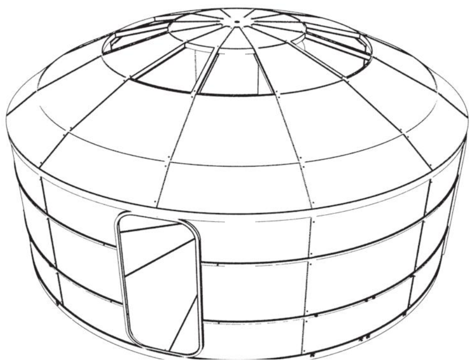


Figure 4.12: Render of fourth iteration of hard-shell module.

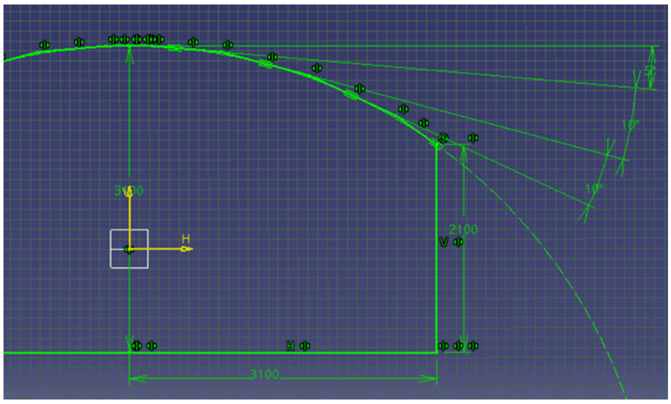


Figure 4.13: Schematic drawing of front cross section of fourth iteration of hard-shell module.

It can be seen that the overall stress levels dropped again (see the scale on Figure 4.10 and Figure 4.11). Clearly, moving the location of the window cut-outs has a beneficial effect on the stress concentrations and it can also be seen that this structure can withstand all loads, also incorporating the required safety factor of 2. Therefore the pressure shell iterations are concluded at this stage. It is recommended to perform a thorough design optimisation campaign in the future, to determine the optimal dimensions of the ribs and spars, and to perform vibration tests. Especially the latter cannot be performed at this stage due to missing vibration properties of the launch vehicles. The ribs and spars depend highly on the exact layout of the interior of The Shell, and the results of the vibration tests. Nonetheless, most of the structural mass comes from the relatively thick walls which, as explained previously, must not be made thinner. With a mass of 2317kg, this structure is heavier than the initially estimated 12% of the initial mass estimation [28], yet this can be explained by the fact that the shell takes over vital parts of the environmental protection, i.e. atmospheric control and meteorite protection.

Whipple Shield Mounting Pins

To accommodate the Whipple shield and to give it its required spacing, it is suggested to use pins. Pins act as spacers and allow for the maintenance of any panel on the Whipple shield. Maintainability of both, the pressure shell and the Whipple shield, are key to ensure nothing goes awry in the ten years of operations. Generally, it is expected that the Whipple shield bumpers and intermediate layers will require frequent replacement due to the constant impact of meteorites and possible due to the intermediate layer's reaction to the radiation environment. Each panel is held in place by four pins, one on each corner of the panel.

The reason these pins are designed in this part of the structure is that they most likely add some weight to the overall structural system. The pins are made of the same aluminium as the pressure shell, thus their yield strength is 393MPa. To reduce the weight, the most optimal diameter has to be found. The pin itself is a forked pin, allowing for the mounting of two neighbouring panels (see Figure 4.14).

The pins take two main loads. Firstly, the weight of the intermediate layer and the aluminium bumpers of the Whipple shield act in one direction perpendicular to the length of the pin. The first weight (F_2) is applied half way through its length, i.e. 150mm from the base of the pin and the second weight (F_1) is applied at 300mm of its length. Next to shearing, these loads also introduce normal stresses due to bending moments. Another source of loading is a forced displacement due to the thermal expansion of the panels. At the largest panel (see Figure 4.15) this displacement is assumed to act perpendicularly to the length of the panel and perpendicular to the applied loads. The bottom section of the pins is

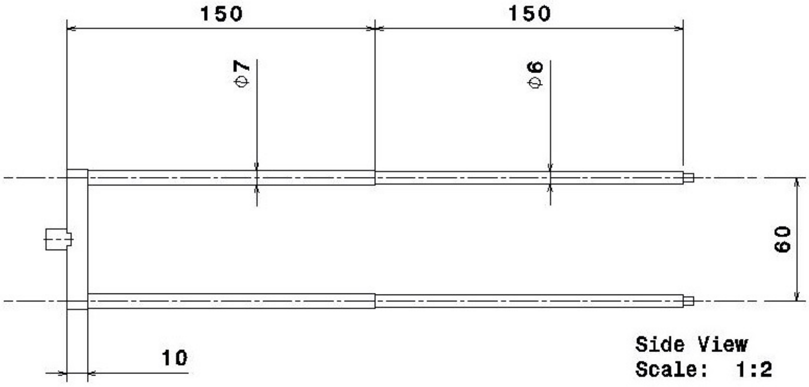


Figure 4.14: Schematic drawing of pin designed to mount the meteorite shield onto the pressure shell. The thicknesses and diameters are guesstimates and will be iterated further after the analysis of the load case (dimensions are given in millimetres).

assumed to be clamped into the wall of the pressure shield. A depiction of this load case can be found in Figure 4.16.
The exact values of forces and displacements can also be found by analysing the most critical cases. Firstly, the force induced by weights are most critical under launching conditions, with an acceleration of 4g. Therefore, with the area of the largest panel ($1.344m^2$ see Figure 4.15) and the areal density of the intermediate layer ($4.64kg/m^2$ see chapter 3) and the areal density of the bumper plate ($3.02kg/m^2$ see chapter 3), the forces can be computed per pin.

$$\begin{aligned} F_1 &= 1.344m^2 \cdot 3.02kg/m^2 \cdot 4 \cdot 9.81m/s^2/4 = 159.3N/4 = 39.8N \\ F_2 &= 1.344m^2 \cdot 4.64kg/m^2 \cdot 4 \cdot 9.81m/s^2/4 = 244.7N/4 = 61.2N \end{aligned} \tag{4.2}$$

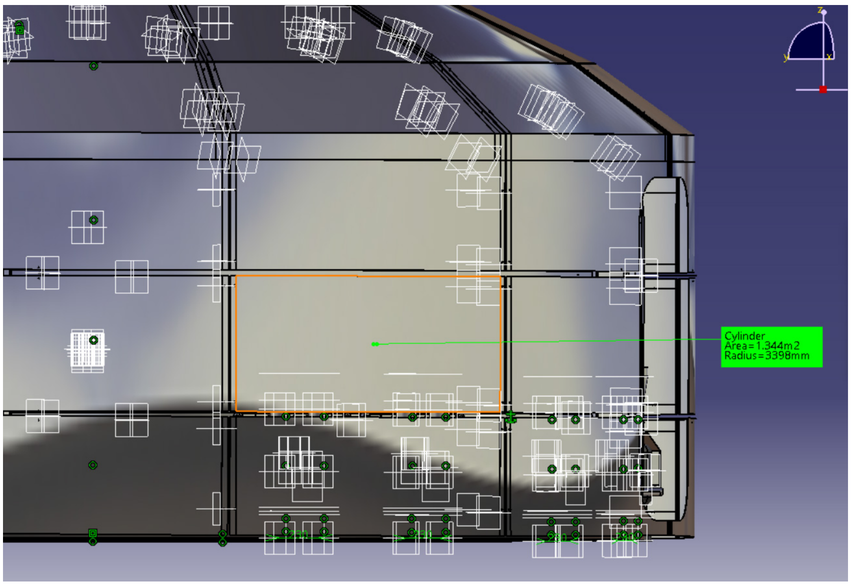


Figure 4.15: CATIA screenshot of shard-shell module with integrated whipple shield panel. The largest shield is highlighted and its data can be found in the green box.

Finally, the forced displacement can be found by looking at the maximum temperature difference the plate has to endure. From chapter 3 the maximum temperature difference can be found to be $\Delta T = 269K - 69K = 200K$. This special type of aluminium has a coefficient of linear thermal expansion of $24.1\mu m/mC°$, meaning that it increases in length by 0.00241% per degree Celsius. With a maximum temperature range of 200K, this means the expansion of the panel is 0.48%. The panel has a length

of $l = 2 * r * \pi * 1/12$ as the panel spans 30 degrees of the circular arc. Figure 4.15 further shows the radius of curvature to be approximately 3.4m. This yields a length of 1.78m. The maximum expansion hence is 0.74mm leaving to a maximum forced displacement of 0.37mm for one pin.
The design of the pins is iterated in a similar fashion as the design of the pressure shell. Even though an analytic solution to this problem can be found, the calculation is tedious for the presented load case and rather than finding an exact analytic solution an iterative process with the aid of a FEM (Finite Element Method) analysis in CATIA is performed. Unlike, however, the design of the pressure shell, no safe life philosophy is applied here. It is to be expected that meteorite impact and the exposure to lunar environment will call for the replacement of a pin, every once in a while, therefore no safety factors are applied, to safe weight, however, spare pins will be brought along.

The first pin iteration (see Figure 4.14) shows a rather spot-on stress concentration after performing an FEM analysis (Figure 4.16). The highest concentration of about 330MPa is found at the end of the pin's shaft, which is only slightly under the yield stress of the material. The thickness of the bottom plate and the front section of the pin, however can be reduced, as the stress is still pretty low. A proposal for a second iteration can be found in Figure 4.17. Figure 4.18 shows that with these slight changes the pin is now under designed. Therefore the thicknesses, especially in the bottom plate are increased again for a third iteration as can be seen in Figure 4.19. The third iteration comes sufficiently close to the yield limit, without exceeding it (see Figure 4.20). Therefore, the iteration process is stopped at this point. A singular pin weighs 55g and the structure employs 117 pins, introducing a total weight of 6.44kg.

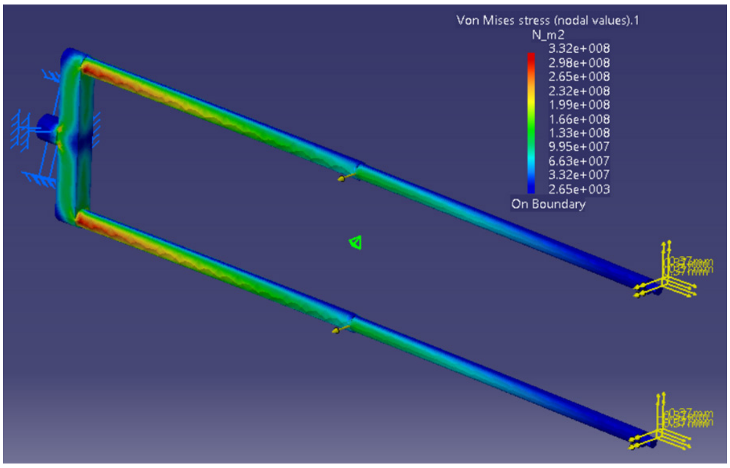


Figure 4.16: Structural analysis results for first iteration of mounting pin generated in CATIA. It can be observed that the pin is still over-designed and that the thicknesses can be reduced.

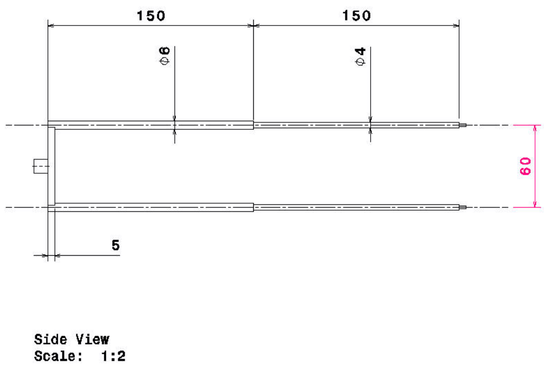


Figure 4.17: Schematic drawing of second iteration of mounting pin (dimensions are given in millimetres).

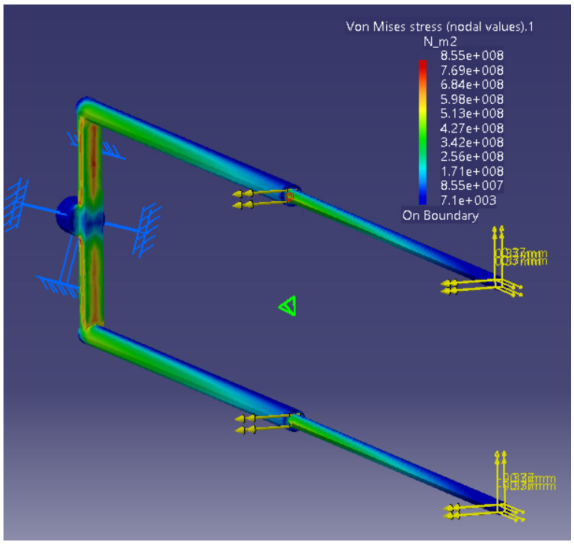


Figure 4.18: Structural analysis results for second iteration of mounting pin generated in CATIA. It can be observed that the pin is now under-designed.

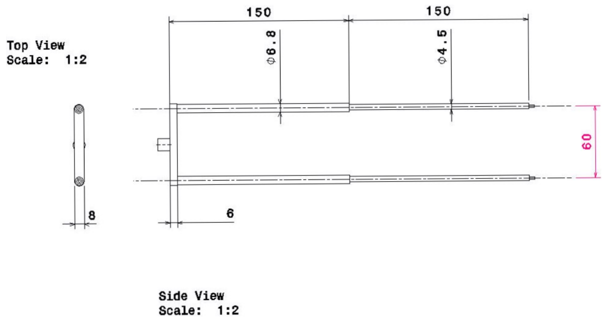


Figure 4.19: Schematic drawing of third iteration of mounting pin (dimensions are given in millimetres).

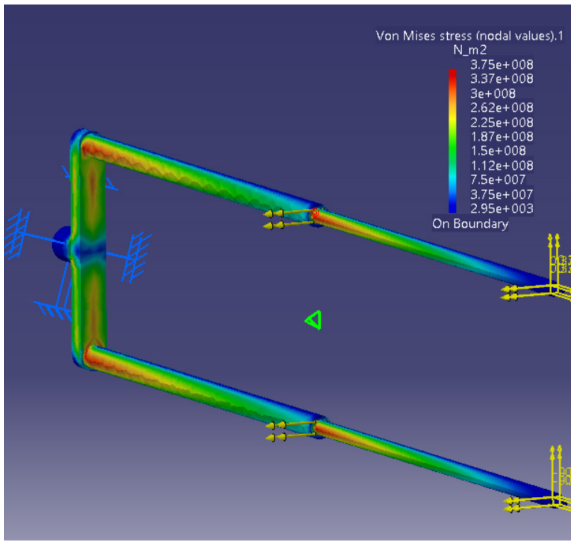


Figure 4.20: Structural analysis results for third iteration of mounting pin generated in CATIA. It can be observed that the pin is now well designed and that the maximum stress comes close to the yield of 393MPa.

Truss Structure and Feet

For the docking of the individual modules, it was decided to use an adjustable support system underneath The Shell. It is found that the most optimal system shape, to give all axis adjustability, is to arrange the bars (legs) of a truss-like structure in a tetrahedron. The ribs of the pressure shell offer an excellent mounting possibility for these legs and hence the following sizing is focused on precisely that. A depiction of the principle can be seen in Figure 4.21.
The attachments of the frontal legs are 3050mm away from the centre of the pressure shell's bottom. Therefore the rear leg is 1579mm from the centre (see Figure 4.22). Therefore an element is approximately 1579mm in length. As mentioned before, the structure is truss structure, meaning that the legs do not prohibit any rotation around their attachments, resulting in a loading in pure compression. As mentioned before, the habitat will have the ability to adjust its overall attitude, and to some extent its elevation, hence the angle of incident varies per leg. Nonetheless it is assumed that for this study every leg takes an equal amount of compressive load.

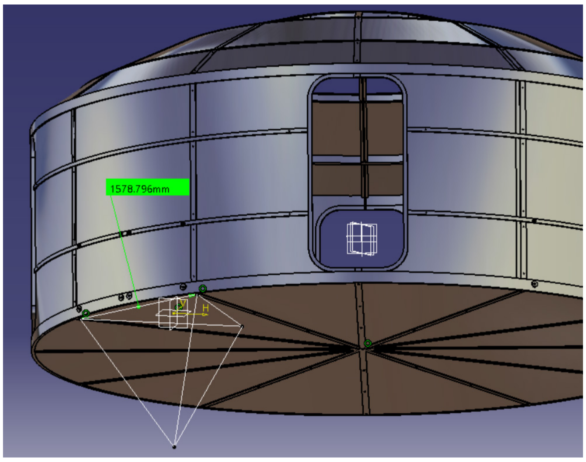


Figure 4.21: Schematic demonstration of the construction of tetrahedral support system for the hard-shell module. Eventually all sketch lines will be beams with adjustable length, to alter the attitude of the hard-shell structure to ease the docking of the inflatable modules.

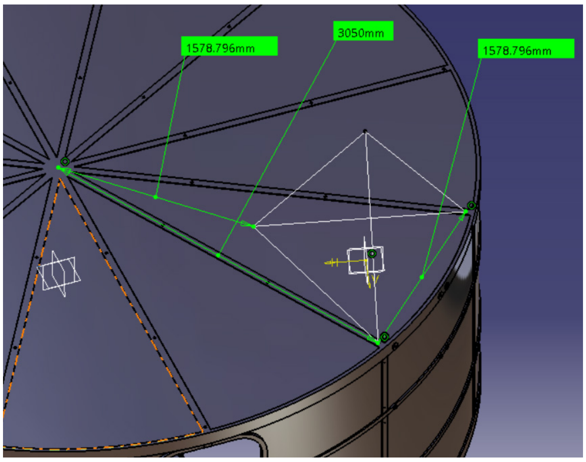


Figure 4.22: Different viewing angle of schematic demonstration of the construction of tetrahedral support system for the hard-shell module. Dimensions were measured in CATIA.

Assuming a habitat mass of 12.4 tonnes under launch conditions, with an acceleration of 4g the load results in $12.4t_l \cdot 4 \cdot 9.81m/s^2 = 486kN$. The structure will employ 6 tetrahedrons, to provide full attitude control and redundancy, where each tetrahedron features three legs. Therefore, the total load is split amongst the 18 legs, resulting in a force of 27kN per leg. With this, the stresses can be analysed. The normal stress in a circular beam can be found by:

$$\sigma_n = \frac{F_n \cdot SF}{A_l} = \frac{F_n \cdot SF}{\pi * r_l^2} = \frac{4F_n \cdot SF}{\pi d_l^2} \tag{4.3}$$

Rearranging for the diameter:

$$d_l = \sqrt{\frac{4F_n \cdot SF}{\pi \sigma_n}} \rightarrow d_l = \sqrt{\frac{4 \cdot 25,000N \cdot 2}{\pi \cdot 393MPa}} \tag{4.4}$$

By employing, again, a safety factor (SF) of 2, accounting for the degeneration of the yield strength of the aluminium the required diameter can be found to be approximately 13mm. Such a thin bar would, however, have the risk of buckling. The formula for Euler column buckling is:

$$F_{crit} = \frac{\pi^2 E_y I_l}{(KL)^2} \tag{4.5}$$

$$I_l = \frac{\pi}{4}(r_l)^4 \tag{4.6}$$

F is, again, the applied load to the structure, E is the Young's modulus (73.1GPa), K is a coefficient depending on the clamping of the bar (in this case it is 1) and L is the length of the bar. From solving these two equations a required diameter of 37mm can be found. With a material density of $2840kg/m^3$ this yields a mass for a singular leg of about 4.85kg and a total structural mass of about 87.4kg. This mass can further be reduced by hollowing the bar and making it a tube. This will increase the moment of inertia, preventing column buckling, while possibly holding the cross-sectional area constant and with that, reducing the mass. The area and moment of inertia of a hollow tube can be found with:

$$A_l = \pi(r_l^2 - (r_l - t_l)^2) \tag{4.7}$$

$$I_l = \frac{\pi}{4}(r_l^4 - (r_l - t_l)^4) \tag{4.8}$$

With a required area of 1.37e-4m^2 and a required moment of inertia of 9.33e-8m^4 this system of equations yields 6 solutions, of which just one is real and positive for both values, and the thickness is smaller than the radius. Therefore it can be found that a tube with a radius of 37.2mm and a thickness of 0.6mm satisfies both conditions. Such a tube weighs 610g and all 18 tubes thus weigh 11.1kg.

4.4. Inflatable Structure

The inflatable structures are hosting very different areas for living when comparing to the hard-shell module. While the hard-shell module serves as a safe room and contains a means of communication, survival and system integration, the inflatable structure shall host the astronauts and provide storage space. For the astronauts to be comfortable and healthy, a spacious living area needs to be created. For this, three modules are considered as a first estimate, featuring a living area of about 45m^2 each. This would result in a living area of 135m^2 excluding the hard-shell module. After reassessing the needed floor area, however, it is found that a total living area of 130m^2 is sufficient to host the astronauts comfortably. Thus, subtracting the area of the hard-shell leaves 100m^2 floor area provided by the inflatables.

Rather than dividing the living area into three relatively small modules, it is assessed whether it is desirable to decrease the amount of inflatables. The main thought behind it is to reduce the duplicate systems and airlocks needed to make all modules self-sustainable. Of course, one module would feature the least duplicate systems. However, in case of this module failing, there would be a drastic decrease in living area for the astronauts. Thus, the risk of mission abortion increases drastically as no further storage or living area despite the safe room is available anymore. Furthermore, one module featuring such a large living area may give rise to inflation issues.

Thus, choosing a number of two inflatables containing an area of 50m^2 allows for keeping redundancy and reducing the duplicate systems. In case of one inflatable module failing, there is still enough living space and food to continue the mission and repair the malfunctioning structure, if possible.

Structural Set-Up & General Dimensions

Using two inflatable module results in a floor area of 50m^2 per module. This floor area needs to be incorporated in such a way that the living area can be divided efficiently. The cross section of the inflatable structure is integral to a create a good living area to structural mass ratio. As the inflatable structure consists of flexible structure subjected to pressure, shapes which include corners are discarded as the impose stress concentrations. Thus, round shapes are in consideration, particularly circular and elliptic cross sections.
Circular cross sections have been considered as the shape does not impose any stress concentrations and it is a typical shape use for inflatable structures. However, in order to create an area of 50m^2, a large radius needs to be employed which results in a very high structure and in a large surface area. Furthermore, at the location where floor area and wall meet, the ceiling is very low and thus, not counted towards living area as the astronauts cannot walk there. Therefore, a larger floor area than the actual living area needs to be created in order to meet the 50m^2. The use of two floors in one module is considered, however, this would give rise to the need of increasing the radius even more. In order to host two stories, with a minimum height of one story of about 2.30m as on Earth the radius would need to increase drastically, making the maintenance of the outside shell very difficult. This again increases the radius or length of the structure, resulting in excess structural weight. Thus, circular cross sections are discarded as a optimal option. Elliptic cross sections can reduce the circumference length of the cross section resulting in a lower surface area and, therefore, in a desirable mass reduction. Further, it decreases the height of the structure which is favourable for maintenance.

In order to maintain the advantages of an elliptic cross section, the structure is cylindrical. However, to reduce stress concentrations, the caps are rounded off. The width of the living area is chosen to be 5m and a length of the cylinder of 10m in order to prevent an excess length of the modules which may lead to issues during inflation as discussed below. In order to obtain the minimum surface area, the ellipse needs to be designed in such a way that the entire floor area to living area (min. ceiling height 1.80) is maximised. Thus, a box of the dimensions 1.80m by 5.0m is created. The minimum surface area is found with an ellipse possessing a major axis of 6246mm and a minor axis of 3003mm. In Figure 4.23 one can see that the floor is located 900mm below the axis of symmetry, this leads to the living area being equal to the total floor area, with is desirable. Further, Figure 4.24 shows the top view of the inflatable, with a total length of 17.5m. This includes the length of the cylinder (10m) and the rounded caps, containing airlocks, connections and bus with a length of 3.75m on both ends.

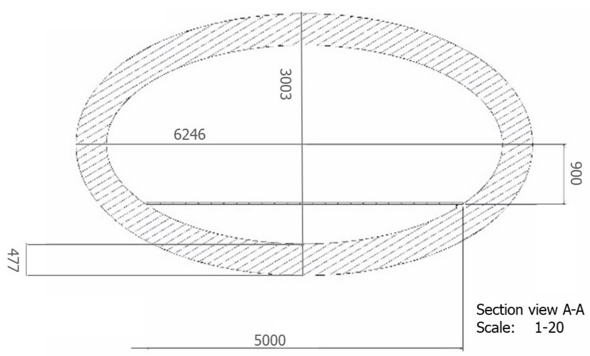


Figure 4.23: Front view of the inflatable structure.

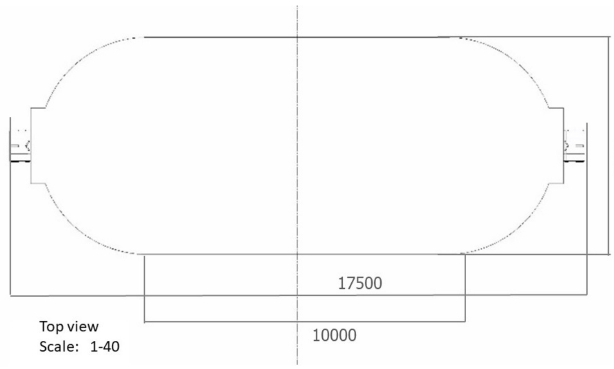


Figure 4.24: Top view of the inflatable structure.

Packed Structure

The inflatable structure consists of mostly flexible materials, however, the integrated airlocks which are located in busses are rigid. The busses also contain the tanks, floor and other items needed to set up the interior. It follows, that the packing capabilities of the structure is constrained by the size of these busses which are located at the cylinder caps. Generally, the folded fabric shall expands longitudinally and radially. Here, the folding pattern plays an important role regarding the deployment ratio (deployed/stowed boom length). In Figure 4.25 one can see a visualisation of the packing and its size. All of them enable a rapid inflation and due to the hole in the middle, a predictable inflation is possible

as the pressure increases evenly throughout the structure during the inflation process. The inflation process needs to be controlled thoroughly to prevent warping or other deformations of the structure which can lead to additional stresses of the structures [29].
Symmetric shapes, generally, can be folded smaller than asymmetric ones. As the inflatable structure of LEAP is has an elliptic cross-section and not a circular one as proposed in the figure, it has to be investigated whether the symmetric folding methods can be applied and which method is suitable with respect to the materials selected.

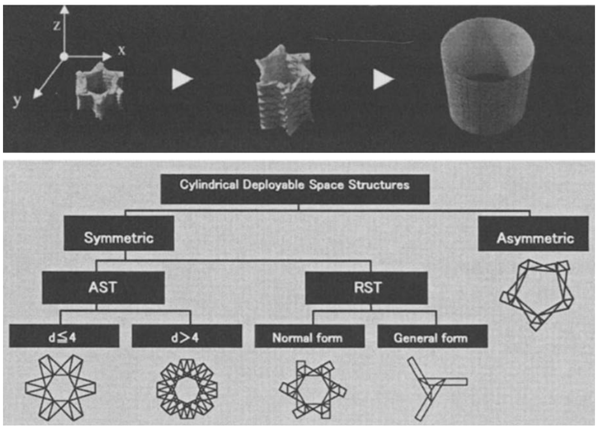


Figure 4.25: Deployment sequence and several folding patterns regarding cylindrical inflatable space structures[29].

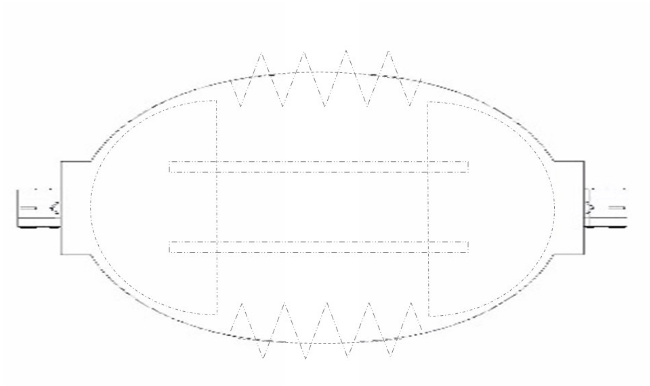


Figure 4.26: Impression of the folded structure. The sized are dependent on folding capabilities of the material and the folding pattern.

Figure 4.26 shows an impression of the packed inflatable structure. Here, the airlock busses are designed to approximate the rounded structure of the shell to reduce the introduction of edges as the fabric needs to be folded around it. Up to this point, the final dimensions cannot be determined as the folding capability of the materials needs to be investigated and the folding pattern needs to be selected. Furthermore, the busses need to be constrained in such a way that they to not damage themselves or other part of the structure. Therefore, they will be connected to each other with stiff hinges which will be released before inflation. The rigid structure of the busses and hinges need to be tested for the launch loads in order to assure their integrity. Furthermore, it needs to be assessed whether the rigid structure poses a thread towards the fabric structure during launch and descent.

Deployment and Technical Risks

The deployment of the structure is initiated after docking to the hard-shell system. The shipment and assembly method can be found in item 11.2. As the hard-shell contains the machinery for inflation, the pressurisation takes place from the connecting side. Generally, the structure is inflated evenly in length and width in order to prevent warping and distortions.

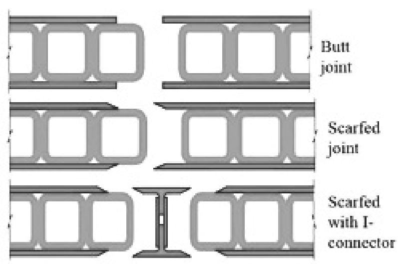


Figure 4.27: Floor panel connection possibilities[4].

The TRL of the mechanism regarding the deployment of inflatable structures is very low. Sogame and Furuya [29] note that is has to be investigated whether the materials can be bent into the stowage shape without creating damage. Furthermore, it needs to be checked whether residual dents remain after inflation as they may reduce the performance of the fabric. Thus, thorough testing is required to ensure the structural integrity of the inflatable after being folded and stowed. Further, the right packing technique has to be found to decrease the stowage volume as much as possible, leading to more space for auxiliary systems within the launcher.

Once the inflatable is pressurised, the floor needs to be integrated by the astronauts. The floor consists of panels (stored in the bus) which may be connected with a conventional panel stacking method. The

[4]http://www.sensorprod.com/news/white-papers/2008-04_slp/index.php [Cited: 20-06-2017]

connection of the panels have to be defined according to the loads it needs to withstand. Some proposed methods can be seen in Figure 4.27. For this, a similar means of fixation needs to be present to fix the flooring in place. Furthermore, struts may be needed to counteract downward bending resulting from the weight of the furniture and astronauts. The material of the floor can be derived from proven aerospace lightweight materials. To find the optimal floor composition and thickness, tests need to be performed to determine the loads acting on the structure. In order to account for the mass of the floor, a conventional aluminium honeycomb panel is chosen which is specifically designed for flooring. The panel features a total thickness of 25mm with a 1mm cover skins and an areal density of 7.5 kg/m^2 [5]. Thus, with a floor area of 50m^2 the weight of the floor per inflatable module is 375kg. The panels are chosen to be aluminium as they are relatively heavy, a down scaling during testing and design of the floor is likely. The excess weight of the structure is introduced to ensure that the inflatable structure fits into the launcher.

Structural Integrity: Design of the Bladder and Restrainer

The load bearing structure of the inflatable shell is the restrainer as shown in Figure 3.9. Table 4.2 shows a selection of commonly used materials which are considered as a restrainer material.
In order to judge the proposed materials, the major loads the structure is subjected to has to be defined. Equation 4.9 shows the hoop and Equation 4.10 longitudinal stress the structure experiences due to internal pressurisation.

$$\sigma_{hoop} = \frac{r_{in} P_{internal}}{t_{min}} \quad (4.9) \qquad \sigma_{long} = \frac{r_{in} P_{internal}}{2 t_{min}} \quad (4.10)$$

where $P_{internal}$ is the internal pressure of 101.325kPa and r_{in} is the inner radius (2.85m) of the inflatable shell. Furthermore, the contribution of the structures mass on the overall stress in the structure is investigated. Equation 4.11 shows the derivation of the stress σ_{iw} resulting of half the structural weight of the MTB: $F_{iw}=M_i \cdot g_M$. The weight is halved due to the fact that the weight assumed to be distributed evenly over both shell halves, as seen from the axis of symmetry.

$$\rightarrow \sigma_{iw} = \frac{F_{iw}}{L_{tot} \cdot t_{res}} \quad (4.11)$$

The trade-off found in Table 4.2 shows a selection of materials which are frequently used as restrainers. The minimum thickness is calculated by using the tensile strength as the stress experienced by the structure. Here it becomes evident, that the stress induced by the weight of the structure is negligible comparing to the pressurisation stress. Using Kevlar 149 as an example, $t_{min,hoop}$=0.085mm, $t_{min,long}$=0.042mm and $t_{min,iw}$=0.1μm resulting in Equation 4.12. Thus, the minimum thickness is driven by the hoop stress of the structure.

$$\sigma_{iw} << \sigma_{long} < \sigma_{hoop} \quad (4.12)$$

It can be seen that the tensile strength of Polyethylene naphthalate (PEN), Polyethylene terephthalate (PET) and Nylon as they have a low tensile strength which results in a high thickness and mass, thus these options are discarded. The trade-off table shows that two materials qualify as sufficient restrainers: Kevlar 149 (Aramid fibre) and Spectra 1000 (Polyethylene fibre). The density of both material differ significantly. Thus, a quick mass calculation is performed to determine if the order of mass difference has a significant impact on the overall structural mass. By calculating the minimum thickness t_{min} needed to sustain the internal pressure and using the overall surface area of the structure, a mass prediction can be obtained. It can be observed that the mass difference of 5kg is negligible in comparison to the total mass of the inflatable structure of several tonnes. The temperature, however, is a crucial parameter. During the day the inflatable may heat up extremely, thus, as Kevlar has a much higher temperature range it is chosen to be the restrainer layer. As seen from the example above, the minimum thickness is calculated to be 0.085mm. I had to be noted that up to this point, there are no safety margins introduced, furthermore, it needs to be investigated whether such a thin film can be produced with a high quality.

The material properties give a very general overview of the performance of a fabric. The final performance of the fabric, however, is a combination of material properties and manufacturing method.

[5]http://www.honeycombpanels.eu/33/honeycomb-panel-compocel-al-(fr), Cited: [20-07-2017]

Table 4.2: Material selection of Restrainer layer. Properties are averages of the respective material [10].

<table>
<tr><th>Restrainer Material</th><th>Tensile Strength [MPa]</th><th>Min thickness [mm]| Mass [kg]</th><th>Elonga-tion [% strain]</th><th>Fatigue over 10^7 cycles</th><th>UV resis-tance</th><th>Temp. range [°C]</th></tr>
<tr><td>PEN</td><td>48 red</td><td>6.02 | 1,227 red</td><td>60 red</td><td>19 red</td><td>Good blue</td><td>-40-170 green</td></tr>
<tr><td>Kevlar 149</td><td>3400 green</td><td>0.085 | 19 green</td><td>1.15 blue</td><td>2750 green</td><td>Fair yellow</td><td>-200-200 green</td></tr>
<tr><td>PET</td><td>54 red</td><td>5.35 | 978 red</td><td>100 red</td><td>21 red</td><td>Fair yellow</td><td>-43-87.5 red</td></tr>
<tr><td>Nylon</td><td>24.6 red</td><td>11.74 | 2,465 red</td><td>5.8 yellow</td><td>16.75 red</td><td>Good blue</td><td>-16-59 red</td></tr>
<tr><td>Spectra 1000</td><td>3100 green</td><td>0.093 | 13 green</td><td>3.2 blue</td><td>2750 green</td><td>Good blue</td><td>-185-100blue</td></tr>
</table>

Therefore, several strengthening methods are proposed. The actual properties of the fabric then are determined by testing.

As the restrainer layer covers a large area, it needs to be split into parts during production and then reattached to form the final layer. The seams introducing considered weak spots as the adherents have a significantly lower strength than the layer material itself. Therefore, the number of seams should be kept as low as possible. New technologies such as seamless weaving or circular weaving can be applied to increase the structures performance. Furthermore, high density weaves can be applied to increase the reliability of the structure as it increase its strength to punctures tears and abrasion [30].

Coatings can also increases the performance of a material with respect to abrasion, tears and corrosion. A commonly used material is polyutherane. Some coatings can also rigidify the structure once inflated resulting in a structure which does not require the atmospheric pressure anymore for structural integrity. Here, the coating is applied to the fabric before inflation. After the inflation procedure, the curing process may be initiated by a chemical reaction. Further research is to be made which materials can be used and how the chemical process can be triggered [30]. A comparable rigidifying concept is applied in the BEAM structure.

Bladder Selection
The bladder of the inflatable structure seals the structure to ensure that the atmospheric conditions are maintained. Typically, elastomers are used as they are classical sealing materials. This layer is attached to the restrainer layer and is not load bearing. Important parameter is its durability. A typical material which is used in the aerospace industry is a Fluroelastomer (FKM) which has high sealing properties. Furthermore, it is nonflammable and features an excellent resistance towards water, alkalis and acids.

To ensure airtightness it is suggested to employ several bladders providing redundancy. Furthermore, to protect the bladder from abrasion over the mission lifetime it may be useful to employ another layer or protection material beneath the bladder system. This layer then can serve as a connection interface for furniture, walls and flooring. Thus, further investigation is needed to determine whether the bladder is subjected to loads. Furthermore, a minimum thickness for these loads need to be set. To be able to give a mass estimation, the bladder material is chosen to be unreinforced FKM with a density of 1.8 g/cm^3 [10]. Again, to avoid an increase in weight during the testing and re-design of the bladder, the thickness is set to be 1mm and no extra protection layer towards the living space is employed.

4.5. Manufacturing & Material Sustainability

The habitat is composed of many different subsystems, where the specifications and standards on different components lead to the need of specialised producers. It is already a widely accepted approach in the aerospace branch that the different components originate from different companies and will be the case for manufacturing the lunar habitat systems. Implementing the subdivision in manufacturing ensures not only the delivery of a high-quality product due to the high levels of expertise, but also enables a faster manufacturing. However, making use of different companies complicates the process as they need to be controlled and deadlines need to be met. On one side, the subdivision enlarges the probability of meeting deadlines as smaller tasks are more likely to meet the set deadline; on the other side, this necessitates a clear communication between the different stakeholders.

Generally, for the manufacturing of the hard-shell structure, a similar approach can be undertaken as for the ISS modules. The expertise that was established for manufacturing these hard-shell modules can be used to make the manufacturing of the hard-shell structure more sustainable. However, since the hard-shell is shaped differently than current ISS modules, it will still require changes in the manufacturing method.
The manufacturing of the inflatable structure will mainly require layering and sewing[6]. This type of manufacturing does not require any pre-treatments and/or special conditions and therefore, does not impose any special location.

The main materials applied are addressed and partially characterised. Generally, all materials need to be applicable for space missions. Furthermore, lightweight structures were preferred in order to reduce the launch mass of the habitat.
The hard-shell module mainly consists of common aerospace materials: Aluminium, Nextel and Kevlar. Further, the inflatable shell consists of a combination of multiple materials starting off with the highest amount: Kevlar, Nextel, Mylar, Poliymide foam and beta cloth. The major materials used for the hard- and inflatable shell are shortly characterised with respect to sustainability, starting with the material which is featured the most within the habitat.

- **Aluminium:** is used for its low cost and high strength to weight ratio. Furthermore, it is isotropic and can be used in many areas. Aluminium is a very sustainable material, because it has been researched extensively in order to decrease required energy, the carbon footprint and other harmful emissions for aluminium production[7]. Generally, aluminium is a preferred material in space application as it is not reactive and therefore, can serve as an outer shell.
- **Kevlar (aramid fibre):** is an important component for the structural integrity of the Inflatable and serves as MMOD protection for the hard-shell and inflatable structure. Kevlar is widely used for its high impact resistance and low weight. Although the production of Kevlar actually can produce energy and no carbon emission takes place, the production is not very sustainable. The main problem with Kevlar production is that it requires sulphuric acid, which is very hazardous to the environment. Same as for carbon fibre, Kevlar reacts with the lunar environment and, therefore, contact with the exterior should be avoided. Furthermore, recyling is not possible and only a fraction of less than 0.06% of the produced material is downcycled [10].
- **Nextel (ceramic oxide fibre):** is known for its excellent durability and single fibre tensile strength. As it is able to sustain harsh environment and impact resistant, it is applied as MMOD protection material within the Whipple shield and Multishock blanket. It is not recyclable but can be downcycled. This means, that Nextel can be reprocessed to a material of lower quality or performance. As most ceramics, however, the sustainability needs to be improved as currently only 1/10th of the used manufactured material flows into back into manufacturing processes. The rest is discarded as waste [10].
- **Mylar (polyethylene):** has a very low conductivity and therefore, is used for thermal insulation in combination with Dacron as a spacer for all structures. Furthermore, Mylar is used as the back cover of the MTB protection. Compared to the other fibres used, Mylar is a fairly sustainable material as it can be recyled and downcycled. Currently, over 21% of the Mylar produced is recycled material. However, one kg of recycled Mylar creates around 1.5kg of CO_2 [10].
- **Polyimide AC550 Foam:** is an open cell foam and thus features a very low weight (7.1 kg/m^3). Used in the inflatable structure due to its thermal insulation properties and as spacer to aid the

[6]bigelowaerospace.com/bacareers/sewing_technicians/ [Cited: 19-05-2017]
[7]www.aluminum.org/major-sustainability-gains-north-american-aluminum-industry [Cited: 19-05-2017]

MMOD protection[8]. Polyimide foam has a good durability in extreme environments and requires no special handling[9].

- **Carbon Fibre:** is a commonly used material in the aerospace industry due to its high strength to weight ratio. Due to the limited space applicability and the availability of more suitable materials, Carbon fibres are not used for the LEAP structure.

RAMS and sensitivity of the LEAP structure

The RAMS and sensitivity analysis of the LEAP structure is, again, divided into two parts: hard and inflatable shell as the expertise of the structures differ significantly.

Hard-shell

Using a metal hard-shell structure is the classic approach for hosting live in a space environment, and is deemed as a reliable safe option. The behaviour of the materials and structures are well researched and can be modelled with available software such as CATIA. For the structure at hand, a safety factor of 2 is used in order to account for the material degradation due to continuous loading along the mission duration. All materials used for The Shell are commonly available. Further, the manufacturing processes are comparable to the ones of the ISS and thus, are available too. The Shell is built up with a panel structure, which facilitates the maintenance as the removal of single panels is easy and straightforward and does not require heavy machinery.
The hard-shell structure is rather insensitive to changes of the loads. Tests showed that an increase in the mass acting on the structure by 1tonne only lead to an increase in weight of about 0.5kg to account for these changes. Thus, if a mass or pressure increase is required, the structure may need to be adjusted slightly inducing extra mass. This needs to be taken into account for the launch.

Inflatables

The reliability of the inflatable structure is yet to be determined. The TRL of the inflation process and folding patterns are still low. Thus research needs to be performed in order to determine the reliability and the possible limitations and constraints regarding the folding mechanisms. Therefore, the outcome of the future research is decisive whether the use of inflatable structures is a valid solution. Further, all the materials, and the strengthening methods regarding fibres selected are common aerospace materials and, thus, available.
As the inflatable structure are a very new method to provide living space in the space environment, the maintenance of such a structure is not developed yet. Here, it needs to be researched whether it is possible to replace parts of the structure or if simple ”patching” can be used to repair minor damages.
The safety is closely related to the reliability for the inflatable shell. Up to present, there is only one inflatable structure in space, namely the BEAM module which is connected to the ISS. However, the space environment is different as the ISS is within Earth proximity, thus, the radiation and meteorite hazards are less severe than on the Moon. Thorough testing needs to be performed in order to ensure that the inflatable structures are able to sustain life in the hostile lunar environment. Further, The Shell needs to be researched on a long term base with respect to ageing and degradation due to radiation.
The inflatable structure, generally, is very insensitive to changes regarding the environmental constraints. A higher or lower pressure or other forces acting on the structure can be counteracted with an adjustment of the bladder and restrainer thickness. This may result in a mass increase which has to be taken into account when preparing the launch vehicle.

[8]http://www.tecnologiademateriais.com.br/mt/2009/cobertura_paineis/painelaero/apresentacoes/SOLIMIDEFoams.pdf [Cited: 26-06-2017]
[9]http://www.professionalplastics.com/POLYIMIDEFOAM [Cited: 26-06-2017]

5
Bioastronautic Subsystem Design

The bioastronautic subsystem involves the biological and psychological needs of the astronauts as well as utilities, and is divided into the water management, astronaut consumption, waste management and the airlock design. The water management system is closely related to the atmospheric control system as described in chapter 3, as it is desirable to have a closed loop system. The water management design is explained in section 5.1. Following that, the required amount of food and medical supplies is discussed in section 5.2. Finally, a design of the airlocks is given in section 5.3.

5.1. Water Management

The water management system entails the generation of water as well as the recycling of water, to make the system as efficient as possible. Different means of generating water are considered and a trade-off has been presented for the final selection. Furthermore, the water supply for a year is sized to meet the needs of the astronauts.

Trade-off Water Generation

The water for the astronauts and subsystems can either be generated by fuel cells or a Sabatier reactor. The fuel cell technology is based on the following chemical reaction:

$$2\,H_2 + O_2 \longrightarrow 2\,H_2O$$

The chemical reaction of the fuel cells will also generate electrical energy. Furthermore, from the chemical reaction, it can be seen that hydrogen and oxygen are needed to produce the water. Hydrogen and oxygen are both generated by the Oxygen Generation Assembly (OGA) through electrolysis.
The Sabatier reactor is based on a different chemical process:

$$4\,H_2 + CO_2 \longrightarrow 2\,H_2O + CH_4$$

Just like the fuel cells, the hydrogen can be extracted from the OGA. The carbon dioxide can be obtained from the habitat atmosphere. Methane is produced by the Sabatier process as a byproduct. The methane can possibly be used in fuel production.

Table 5.1: Trade-off of selected water generation systems. Green: Exceeds requirements. Blue: Meets requirements. Yellow: Correctable deficiencies. Red: Unacceptable.

Criterion > Option	**TRL**	**Needed resources**	**Power [kW]**	**Operating temperature [°C]**
Fuel cell	9, Space Shuttle *green*	O_2+H_2 *yellow*	None, generates power *green*	1-80 *blue*
Sabatier reactor	9, ISS *green*	CO_2 + H_2 *green*	±1.5 *blue*	250-400 *yellow*

A trade-off is performed on the fuel cell and Sabatier reactor. Each of the trade-off parameters presented in Table 5.1 will be briefly explained.

TRL: The technical readiness level of each system is considered. Both systems have been proven as an in-flight design, the fuel cells have been used on the Space Shuttle. On the ISS, a Sabatier reactor is used to produce potable water.

Needed Resources: The needed resources determine what is needed to produce water. For the fuel cells, oxygen and hydrogen is needed. Oxygen is preferably only used for the crew and storing oxygen and hydrogen brings complexity, since the storage tanks will require pressurisation and cooling.

Power: The consumed power by each system. The fuel cells do not consume power but rather generate power by consuming oxygen and hydrogen, resulting in water as a byproduct. The Sabatier reactor is part of the OGA, which consumes 1.5kW. A conservative estimation is done by assuming the Sabatier reactor will consume approximately 1.5kW[1].

Operating Temperature: The operating temperature indicates the temperature range in which the system needs to be to function properly. The fuel cell can operate in a much lower temperature than the Sabatier reactor.

Water Supply Sizing

The Sabatier reactor is chosen over the fuel cells because of its ability to use the carbon dioxide in the air to generate water. This results in a closed loop environment, which makes the habitat more sustainable as a system. Next to the carbon dioxide, hydrogen is also needed for the Sabatier reactor to produce water. This is obtained from the OGA and the amount of available hydrogen depends on the oxygen production. The electrolysis process of the OGA is based on the following reaction: $2\,H_2O \longrightarrow 2\,H_2 + O_2$ Every day, 3.9kg of oxygen needs to be generated, including oxygen losses due to leakage, according to Design Rules for Life Support Systems [3]. This results in 0.5kg of hydrogen produced by the OGA per day. A crew of four astronauts expels 4kg of carbon dioxide per day. When the masses are converted to moles and inserted in the chemical equation of the Sabatier process, it can be concluded that the amount of hydrogen is the limiting factor for the Sabatier reactor. Therefore, the amount of water the Sabatier can provide with this hydrogen supply is 2.2kg per day.

The amount of water that is needed from the water management for four astronauts (including water for consumption and hygiene) and the oxygen generation is 51kg per day. The amount of water that is extracted from the atmosphere and recycled from sanitary utilities is 46.6kg. Adding the 2.2 kg of water generated by the Sabatier reactor and applying an efficiency of 93%[2] results in a total water recovery of 45.4kg. This is less than the required water per day. A loss of 5.6kg of water is estimated and extra water storage is needed to compensate for this loss. Over one year, approximately 2 tonnes of water are needed to compensate for the water loss.

In case of a failure of the water management, which will halt all water recovery operations, a water supply for the astronauts for seven days needs to be provided, which is 357kg. A seven days period of emergency is based on the Apollo 13 mission, in which it took six days to return the astronauts to Earth after a mission abort[3] The tank size which will hold all the water will be approximately 2.5m^3. The system mass, volume and power of the water management are estimated to be equivalent to that of the ISS, which are 797kg, 11.47m^3 and 0.5kW respectively[4].

RAMS Characteristics and Sensitivity Analysis

The water management system is based on the one from the ISS. On the ISS, the water management system showed to have a reliability of five failures in two years [31]. Two of the five failures are suspected to be caused by an assembly error. It is assumed that the water management of the lunar habitat has the same or a better reliability, since the causes of the failures can be investigated and improved to reduce the chances of failure.
The water management system is a proven and working system, as it is operating on the ISS. Thus it will be available and possibly improved in terms of efficiency.
Regular checks should be conducted to make sure the system operates adequately. Inspection of failure

[1]salotti.pagesperso-orange.fr/lifesupport3.pdf [Cited:12-6-2017]
[2]https://www.youtube.com/watch?v=BCjH3k5gODI [Cited: 26-06-2017]
[3]http://spectrum.ieee.org/aerospace/space-flight/apollo-13-we-have-a-solution [Cited:26-6-2017]
[4]http://salotti.pagesperso-orange.fr/lifesupport3.pdf [Cited: 12-06-2017]

could be hard depending of the placement of the system, for example, if it is placed under the floor, inspection can be harder even with a removable floor. Therefore, it is preferred to design the water management to be safe life. The greatest risk of the water management is leakage. If water leaks into the habitat, not only will there be a loss of water reserves, but the water could also damage other systems.

The chosen design is not sensitive to changing parameters. Even if the water capacity was to be changed, either larger or smaller, the water management system would still be based on that of the ISS and make use of a Sabatier reactor. The main aspect that changes with changing capacity is the tank volume for the water.

5.2. Astronaut Consumption and Generation

This section describes the supplies the astronauts require for a one year stay and the waste and trash they will generate.

Food and Medical Supplies

The food consumption is estimated to be 2.5kg per day for four astronauts, which is based on a 2975kcal caloric intake per day [3]. For a whole year for a crew of four, this will result in 905kg of food and 1238kg including packaging. Most of the food is sent to the Moon in a dehydrated state and needs to be rehydrated for consumption. The amount of water that is needed for rehydration is taken into account in the water budget. The astronauts can select the menu according to their taste.
Next to food, medical supplies are provided to ensure the health of the astronauts. The medical kits for the lunar habitat are similar to those aboard the ISS. These medical kits include medication as well as tools. A total of nine medical kits are provided, including a special emergency treatment kit. Medication includes items such as syringes with sedatives and pills for mental health. Tools found in the medical kits include a defibrillator, oxygen supplementation and bandages[5].

Waste and Trash

In the lunar habitat, a clear distinction should be made between trash and waste. The waste is defined as all the organic matter expelled by the astronauts such as faeces and sweat solids. The trash is considered to be all material which is not usable anymore such as packaging material. The waste from the astronauts could be handled and process such, that it can function as fertiliser for a potential greenhouse. Until then, the faeces need to be stored inside the habitat. In a year, a total of 161kg of waste is produced, which leads to a storage volume of $0.16m^3$ [3] for a year, assuming that faeces have approximately the same density as water. The trash in the lunar habitat should be minimised as much as possible, since it is too expensive to burn the thrash in Earth's atmosphere as is done for trash on the ISS. Smart packaging is one way to reduce trash. Another way to reduce the trash is to make packaging material from polyether ether ketone (PEEK), such that it can be recycled using 3D-printing technologies. These 3D-printed components from PEEK can be used as repairing material or new items for example, furniture (see section 15.1).

5.3. Airlock Design

An airlock design is chosen by performing a trade-off on proposed lunar airlocks. Moreover, a dust mitigation plan is presented.

Airlock Trade-off

An important requirement of the habitat is that the astronauts can enter and exit the habitat (MR-EE). To be able to maintain a liveable atmosphere inside the habitat (SYS-EP-04), an airlock is required. There are four options for airlocks for the habitat: the regular airlock, the suitlock, the suitport and the Lightweight Inflatable Structural Airlock (LISA). The regular airlock has a pressure chamber with one door to the habitat and one door to the outside[6]. LISA is like a regular airlock, but the pressure chamber can

[5] https://ntrs.nasa.gov/archive/nasa/casi.ntrs.nasa.gov/20110010924.pdf [Cited:26-6-2017]
[6] https://ntrs.nasa.gov/archive/nasa/casi.ntrs.nasa.gov/20080018968.pdf [Cited: 07-06-2017]

be inflated and deflated[7]. The suitport is a small door to which a space suit can be connected in such a way that the astronaut can climb in. The suitlock is a combination of a suitport and an airlock. This means that the suitlock has two doors like the airlock, but also two suitports that transfer to the inside of the airlock. This enables equipment moveing into the habitat while the spacesuits stay in the airlock[6].

In Table 5.2 the trade-off of these different airlock options is displayed. The trade-off of these options is based on the following eight criteria: mass, power, volume, TRL, operation time, manoeuvres, dust measures and seal length. The weight of the criterion is represented by the width of its respective column in the trade-off table.
The mass is a relatively important criterion, because it influences the transportation, which is a big driver for the cost. The power required is equally important, because this will influence the size of the power system, which again influences both transportation and cost.
Still quite important, but less so are the TRL and the operation time. A lower current TRL puts a higher strain on the schedule of the habitat, because the development of these systems might take longer. Current TRLs below three are not acceptable for this phase in the mission. The operation time of the airlock creates time loss for the astronauts, which in case of an emergency could have critical consequences. Therefore, the astronauts should be able to operate the airlock within 30 minutes in case of an emergency[8].
The dust measures, which include the necessity to operate any additional dust removal systems each time the airlock is used are important, because these systems may increase the mass and power of the atmospheric control. However, because these systems are most likely already present in case dust enters the habitat in some other way, this increase is not very high and therefore not as significant. The seal length is of importance, because the amount of leakage increases with increasing seal length. This would create a minor increase in the size of the tanks to account for this.
Lastly, the volume transferable outside (number of astronauts or tools) and the amount of manoeuvres required for two astronauts are taken into account in this trade-off. The volume influences the transportation, although less so than the mass and power. The amount of manoeuvres are important for the operationality of the habitat. If the amount of manoeuvres is very high, the operation of the airlock will be complicated and become a tedious ordeal for the astronauts.

Table 5.2: Trade-off of selected airlock systems. Green: Exceeds requirements. Blue: Meets requirements. Yellow: Correctable deficiencies. Red: Unacceptable.

Criterion > Option	**Mass [kg]**	**Power [kW]**	**TRL**	**Operation time [min]**	**Dust measures***	**Seal length**	**Volume [m^3]**	**Manoeuvres****
Airlock	631.1 *blue*	nominal:5, crisis:15 *blue*	9 *green*	nominal:35, crisis:10 *blue*	always *yellow*	9.3 *blue*	4.25 *blue*	7 *blue*
Suitlock	715.1 *blue*	nominal:5, crisis:0 or 15(equip) *blue*	4 *yellow*	equip:35, exit:20, enter:10 [8] *blue*	only equipment *blue*	13.75 *yellow*	4.25 *blue*	7 *blue*
Suitport	50 *green*	0 *green*	4 *yellow*	exit:20, enter:10 [8] *blue*	never *green*	6 *green*	0.125 *green*	N/A *red*
LISA	unknown *red*	unknown *red*	2 *red*	unknown *red*	always *yellow*	9.3 *blue*	0.95 *green*	9 *yellow*

*additional measures required to avoid dust travelling into the habitat via astronauts and equipment.
**amount of manoeuvres required to move two astronauts and a toolbox outside.

[7]https://techport.nasa.gov/externalFactSheetExport?objectId=34150 [Cited: 06-06-2017]
[8]http://www.kta-gs.de/e/standards/3400/3409_engl_2009_11.pdf [Cited: 27-06-2017]
[8]http://astronautical.org/sites/default/files/astronauts-robots/2015/astronauts-robots_2015-05-12-1515_gernhardt.pdf [Cited: 07-06-2017]

As can be seen in Table 5.2, the LISA is unacceptable for multiple reasons. Most of this follows from the fact that the TRL is very low and therefore a lot of the other trade-off parameters are still unknown about the concept[7].

The suitport is with respect to many trade-off parameters an excellent option, because it is small and therefore light[6] and can be operated without using pumps and thus without power[9]. However, the suitport can not accommodate astronauts that have to transport items in and out of the habitat. The purpose of a Moon walk will often be to perform research or to repair and maintain the habitat. Since these actions both require tools, parts and/or samples to be moved in and out of the habitat, the suitport would not suffice as single type of airlock. It is discarded entirely, because it would create additional weak points in the structure and it can not be used to extend the base.

This leaves the suitlock and the regular airlock. The suitlock is better with respect to dust measures, because the dirty spacesuits do not enter the habitat as is the case with the regular airlock system[6]. However, this requires more doors, which increases the seal length which is equivalent to leakage[6]. In the end, the regular airlock is the better option, mainly due to its higher TRL.

Airlock Dust Mitigation

Lunar dust can damage the health of the astronauts, because the small particles can easily be inhaled and have toxic properties. The main cause for lunar dust migrating into the habitat is dust adhering to the astronauts' suits and tools during a Moon walk, which is then brought back into the habitat. To limit the amount of dust migrating into the habitat, the space suits should be cleaned, so that the majority of the dust does not enter the habitat.

There are six cleaning solutions: electrostatic, electromagnetic, gas, fluid, manual cleaning and the use of covering materials. The electrostatic and electromagnetic cleaning solution are based on the specific properties of lunar dust, while the gas, fluid and manual cleaning solutions generate a general motion to remove the dust. The cover, which is worn as a suit over the space suit, the lunar dust will adhere to the cover, after which it is removed before entering the habitat.

Since fluid cleaning solution can be discarded immediately, because lunar dust will agglomerate when exposed to small amounts of water [32]. Therefore to remove it a relatively large amount of fluid would be required, which is not desired because this will create significant losses in the water system. To solve this, the water would have to be recovered and filtered again to avoid contamination of the habitat water. This solution will only move the problem and not solve it directly and therefore is not considered.

Table 5.3: Trade-off of selected airlock dust mitigation systems. Green: Excellent; exceeds requirements. Blue: Good; meets requirements. Yellow: Correctable deficiencies. Red: Unacceptable.

Criterion > Option	**Efficiency [%]**	**Operation time [min]**	**Power [kW]**	**Mass [kg]**
Electrostatic	80 ^{blue}	5 to 15 ^{blue}	unknown $_{yellow}$	unknown $_{yellow}$
Electromagnetic	40 ^{yellow}	5 to 15 ^{blue}	unknown $_{yellow}$	unknown $_{yellow}$
Gas	40 ^{yellow}	5 ^{green}	unknown $_{yellow}$	unknown $_{yellow}$
Manual	60 ^{yellow}	10 to 15 ^{blue}	0 ^{green}	0.64 ^{green}
Cover	80 ^{blue}	5 to 10 (twice) ^{blue}	0 ^{green}	0.45 ^{green}

For the other five cleaning solutions the trade-off is presented in Table 5.3. As can be seen in Table 5.3, the electrostatic solution is efficient enough and the operation time is not too long. However, the system is still being developed and therefore it is only known that it will require power and not what this required

[9]http://spacearchitect.org/pubs/SAE-2000-01-2389.pdf [Cited: 07-06-2017]

power and mass of the system would be [33].
The same holds for the electromagnetic system which has the disadvantage of not being efficient enough to meet the requirements of the system[10] just like the gas solution. Nevertheless, the operation time is relatively short because the dust can be removed quickly by the expelled gas[11]. The manual option would not require any power and the mass is low. The criterion setting this option back is the fact that due to human errors the efficiency is lower than required[12]. This leaves the cover option which has the required efficiency, requires no power and has a relatively small mass. The fact that the operation time is a bit longer, because the suit cover will have to be donned and doffed[13].
Finally, based on the trade-off the cover solution is the best option.

RAMS Characteristics and Sensitivity Analysis

The conventional airlocks have proven to be reliable, as there were no critical incidents during the ten year use of the airlocks on the ISS. The concept of a conventional airlock system is widely used is space missions. Therefore, the technology and mechanisms required are well developed and ready to use. However, airlocks with a door shape with the desired dimensions has yet to be put in practice. Regarding the maintenance of the airlocks, regular inspection and checkups is advised, to spot possible damage or malfunctions. Especially for the airlock, safety is of utmost importance. The airlock acts as a gateway between the lunar habitat and the lunar environment. A double door system reduces the chance of leakage from the habitat to outer space.
The use of the conventional airlock design is insensitive to design changes. For example, if the crew size was to be increased, an extra airlock would be installed rather than choosing for a different airlock design.

[10]http://www.lpi.usra.edu/meetings/leag2009/pdf/2005.pdf [Cited: 07-06-2017]
[11]https://ntrs.nasa.gov/archive/nasa/casi.ntrs.nasa.gov/19920010136.pdf [Cited: 07-06-2017]
[12]https://sspd.gsfc.nasa.gov/documents/5_EVA_Tools_and_Equipment_Reference_Book_Nov_83.pdf [Cited: 07-06-2017]
[13]https://imageserv5.team-logic.com/mediaLibrary/93/Dust_Mitigation_Solutions_for_Lunar_and_Mars_Surface_Systems.pdf [Cited: 07-06-2017]

6
Communication and Data Handling Subsystem Design

This chapter outlines the communication infrastructure and data handling for the lunar habitat. The data handling is explained in section 6.1, followed by the data communication in section 6.2. Lastly, the Earth - Moon link and local lunar communication links are described in section 6.3.

6.1. Data Handling

Every system of the lunar habitat and the lunar village creates a specified set of data that needs to be processed, analysed, integrated and saved. Every subsystem that generates and uses data will process the data itself for immediate use. All data will also be sent to and saved in the main Communication and Data Handling System (CDHS). There it is processed for other subsystems that rely on the data and it is analysed to perform system diagnoses. Raw data and processed data is also sent to Earth for verification and further analysis.
Figure 6.1 gives an overview of the entire data flow of LEAP. This includes internal and external data handling as well as the data flow between Earth and the Moon.

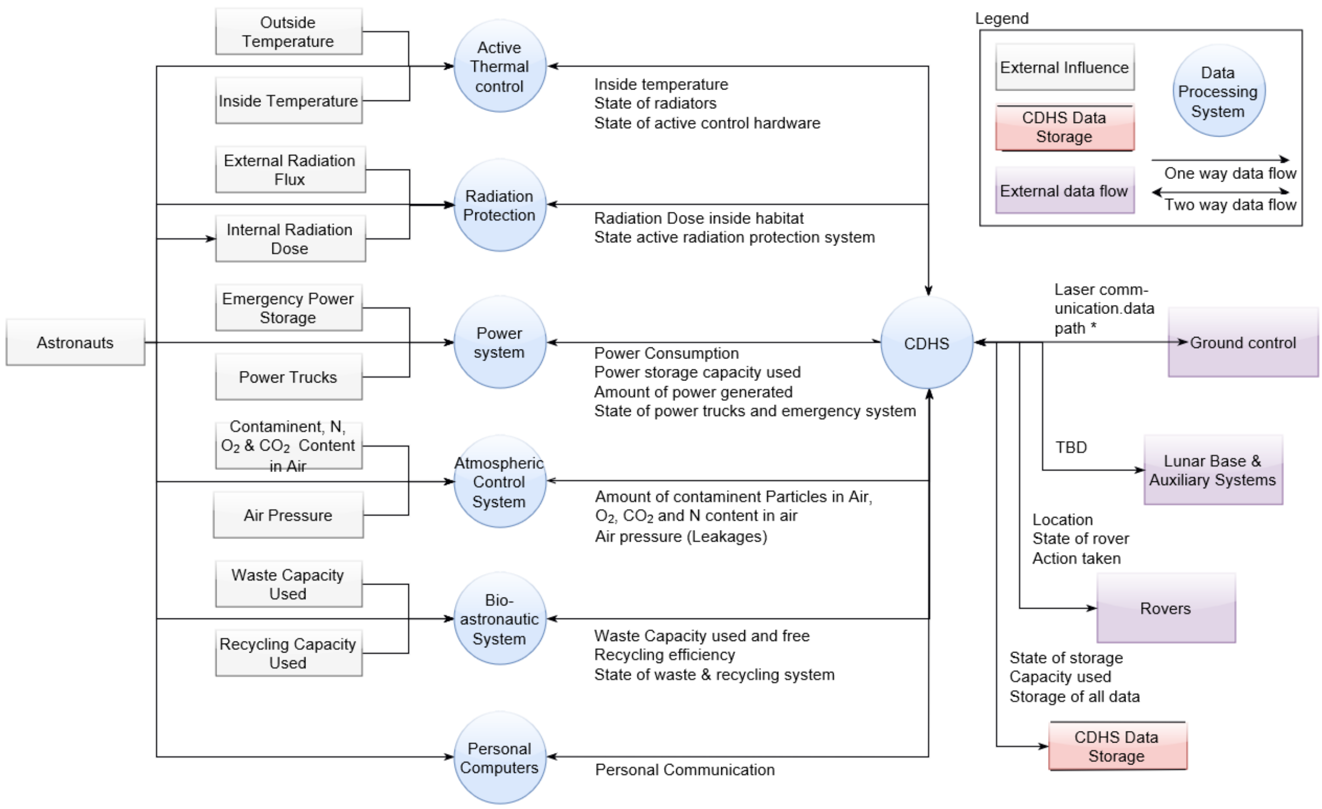


Figure 6.1: Data flow of LEAP. The detailed communication data path can be found in section 6.2.

Regarding the hardware components of the data handling system, standards have to be met for the design of the CDHS. The main standard is the AS1773 which defines the design of an optical serial data bus with a 20 Mbps bandwidth. This standard has been applied in various space systems and has

demonstrated to be capable of performing the highly irradiated space environment [34]. Optical systems are preferred over electrical mechanical systems as they accommodate greater bandwidths, have a lower data loss and a lower weight.
For the astronauts, to interact with the CDHS, laptops will be provided. These laptops have to be selected trough a certification process and are tested for radiation, off-gassing, thermal properties and fire hazards and suppression[1]. They also have to be modified for cooling (air circulation is less with less gravity) and the effects of higher radiation. In the habitat, a personal laptop is provided for every astronaut and also separate laptops for the power system, the atmospheric control, active radiation protection, communication, and as a server. Also, one back-up computer is taken into account for the initial supply, which brings the total amount of laptops at ten.

6.2. Earth - Moon Link

For the sake of optimal communication for mission directives and contact with astronauts' family and friends, it is advised to ensure an uplink that can support live video communication. Furthermore, the uplink will be needed for direct commands and scientific data from Earth. This would require a minimal uplink bandwidth around 3 Mbps. However, to increase the scientific output and social connection of a lunar mission the uplink should be increased if possible. This would enable larger data sets to be sent to the base or would support a better internet connection to interact with the public. So far, the largest possible uplink to the Moon ever was achieved by the Lunar Reconnaissance Orbiter (LRO) and was only 4 Kbps[2].

The downlink should be maximised as well in order to achieve high scientific output. The highest downlink achieved from the Moon was 100 Mbps, also by the LRO [3] using a Ka-band frequency. The downlink from the ISS has been upgraded from 150 Mbps to 300 Mbps in 2013[4] using Ku-band communication. The downlink data rate for the lunar base should be similar and preferably higher.

As part of the Lunar Atmosphere and Dust Environment Explorer (LADEE) mission, NASA performed a Lunar Laser Communication Demonstration (LLCD) [35]. The results of this experiment exceeded all expectations. Compared to the Radio Frequency (RF) communication link of the LRO, as can be seen in Table 6.1[5], the improvements are impressive. Unfortunately, the exact Bit Error Rates (BER) of the LRO could not be retrieved, but the values provided are what has been found as BER requirement for the primary communication links of current and future spacecraft. The requirement for the LEAP has been set at a BER of 10^{-9} or less, just like the optical link requirement of the ISS [36].

Table 6.1: Comparison of LLCD and LRO communication link.

Characteristic	LLCD	LRO
Downlink	622 Mbps	100 Mbps
Uplink	19.44 Mbps	4 Kbps
Weight	30.7 kg	60+ kg
Average power	90 W	115 W
Uplink Bit Error Rate	$\sim 10^{-7}$	$10^{-8} \sim 10^{-9}$
Downlink Bit Error Rate	$\sim 10^{-7}$	$10^{-8} \sim 10^{-9}$

This experiment, preceded by multiple others like sending a picture of the Mona Lisa to the Lunar Orbiter Laser Altimeter (LOLA) on NASA's Lunar Reconnaissance Orbiter (LRO)[6], the Optical Payload for Lasercomm Science (OPALS) mission [37] and the use of high-speed lasers for the European Data Relay System (EDRS) [38], show that laser systems are a promising communication method for space systems. Also, future missions are planned with laser communication, mainly NASA's Lunar Relay Communication

[1]blog.lenovo.com/en/blog/thinkpad-laptop-nasa-youtube-spacelab [Cited 23/06/17]
[2]pds.nasa.gov/ds-view/pds/viewHostProfile.jsp?INSTRUMENT_HOST_ID=LRO [Cited 19/06/17]
[3]www.nasa.gov/mission_pages/LRO/news/LRO_twta.html [Cited 19/06/17]
[4]www.nasa.gov/directorates/heo/reports/iss_reports/2013/ISS_Daily_Summary__040213.html [Cited 19/06/17]
[5]www.youtube.com/watch?v=F7oJg3KUW9g [Cited 20/06/17]
[6]www.nasa.gov/mission_pages/LRO/news/mona-lisa.html [Cited 20/06/17]

Demonstration [39] and the use of a laser comb terminal on the Mars 2020 rover [40]. It is safe to say that future (deep) space optical communication systems will meet the BER requirements and become smaller, lighter and more efficient over time. However, for LEAP, a similar system to the LLCD is considered due to the available data, for which the following characteristics will apply:

The Lunar Lasercom Space Terminal (LLST) comprises three different modules: an optical module, a modem module, and a controller electronics module. They can be seen in relative size in Figure 6.2.

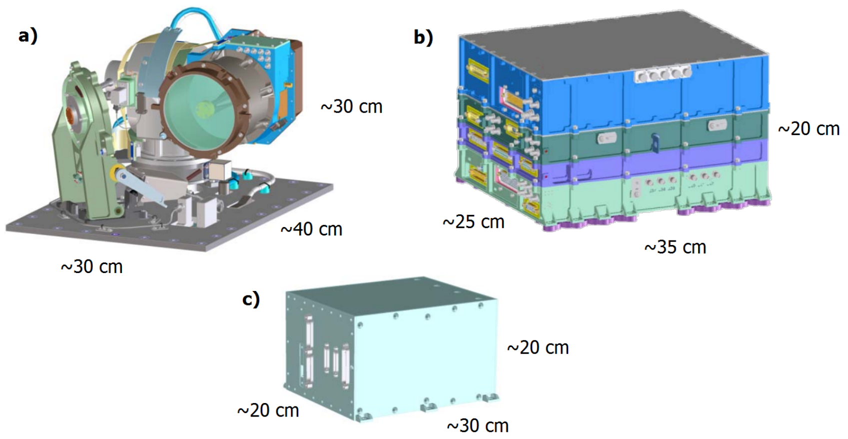


Figure 6.2: Lunar Lasercom Space Terminal. a) optical module, b) modem module, c) controller electronics module.

The optical module will be mounted on the exterior of the lunar habitat. The modem module and controller electronics module will be installed on the inside of the habitat and need to be adapted for or connected to a direct human interface. Further, a specific ground station, the Lunar Lasercom Ground Station (LLGT), was design for the LLCD mission. It consists of a gimbal system with four large receivers and four smaller transceivers, and a control room, as can be seen in Figure 6.3. The entire LLGT is designed to be transportable. This will be necessary to easily install multiple ground stations around the globe. More technical information about both stations, which is outside the scope of this project, can be found in [35]. The data path between the LLGT and the LLST is presented in Figure 6.4.

Figure 6.3: Lunar Lasercom Ground Terminal.

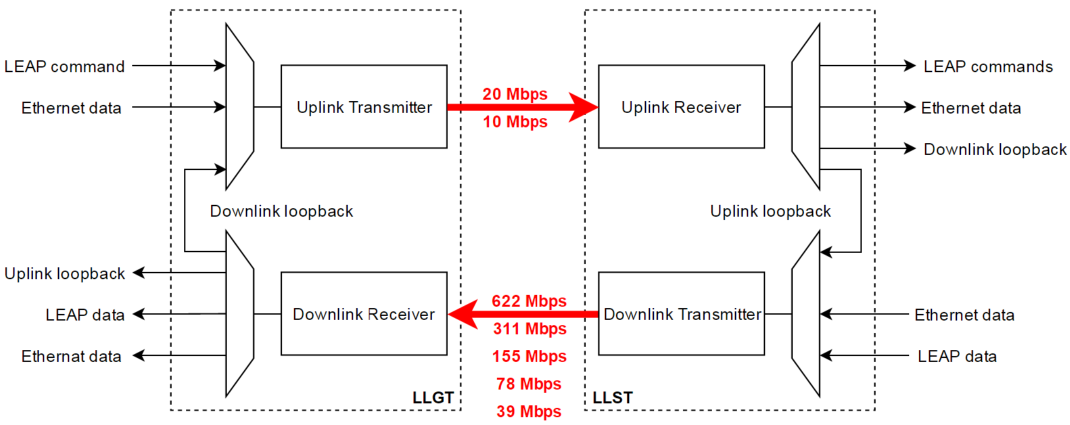


Figure 6.4: Data path of the lunar laser communication system [35].

The main issue with the communication system will be the communication window. Since one ground station will not always be in line of sight of the LEAP, at least three different ground stations will be needed. Preferably these stations are installed in regions with low cloud coverage as was done for the LLCD, where a 92% availability [7] was achieved with a station in White Sands (New Mexico), Wrightwood (California) and Tenerife (Canary Islands). In order to further minimise the cloud coverage over ground stations, four or more ground stations have to be employed to be able to switch to a cloud-free station. Also, a relay and data storage system have to be installed to allow a minimal amount of time for pausing the data stream (to swith the link to a different ground station) and retrieving it when communication is restored. Such a system was also implemented and tested in the LLCD and could bridge short cloud-outages.

A second problem might be the presence of lunar dust flying around when a landing is performed in the proximity of the LEAP. To preserve the telescope lens it will need to be temporarily shielded and thus optical communication will not be possible, but a minimal back-up Earth-Moon link can be established with traditional RF communication. The Ka-band antenna used for the precision landing beacons (see section 11.2) can be reused in this case in order to transmit and receive minimal required data. Large data bulks can be stored temporarily and be sent once the optical link is reinstated again.

6.3. Communication on the Moon

The communication on the Moon consists of internal communication within the habitat and external communication like between the rovers.

External Communication

For external communication, the use of UHF bands is preferable for the size of the antennas, that can range between 2.5 and 25cm, depending on the wavelength. Even though UHF waves can travel through walls and small obstacles, the antennas need to be in line of sight. For the 4x4km area that has been defined for LEAP's habitat mission, the antenna height h_{UHF} would need to be at least 2.26m, assuming the Moon to be a dentless sphere:

$$h_{UHF} = 1000 \cdot \left(\frac{\sqrt{2 \cdot L_{UHF}^2}}{2} \cdot \left(sin\left(tan^{-1}\left(\frac{\sqrt{2 \cdot L_{UHF}^2}}{2 \cdot R_{Moon}} \right) \right) \right)^{-1} - R_{Moon} \right) \tag{6.1}$$

where the Moon radius R_{Moon} = 1737km and one side of the square landing area L = 4km. The antenna on the lunar habitat will be elevated 5m above the ground, attached directly on top of the hard-shell module. Since the habitat will be landed somewhere in the predefined landing area, connection with all four scouting and beaconing rovers is possible after landing. All rovers and astronauts are then equipped with a UHF transceiver which would create a large local UHF network of approximately 87km^2.

Internal Communication

For communication within the habitat, a traditional wireless network is proposed. A router will be installed in every module to ensure a wireless connection.

The hardware-software interaction can be found in Figure 6.5 it shows the link between hard- and software within the habitat. Furthermore, it shows the vital link between the CDHS and all other systems. Although most systems are designed to run autonomously, it is possible for the astronauts to give direct commands to the subsystems.

[7]https://www.youtube.com/watch?v=F7oJg3KUW9g&t=43m6s [Cited 20/06/17]

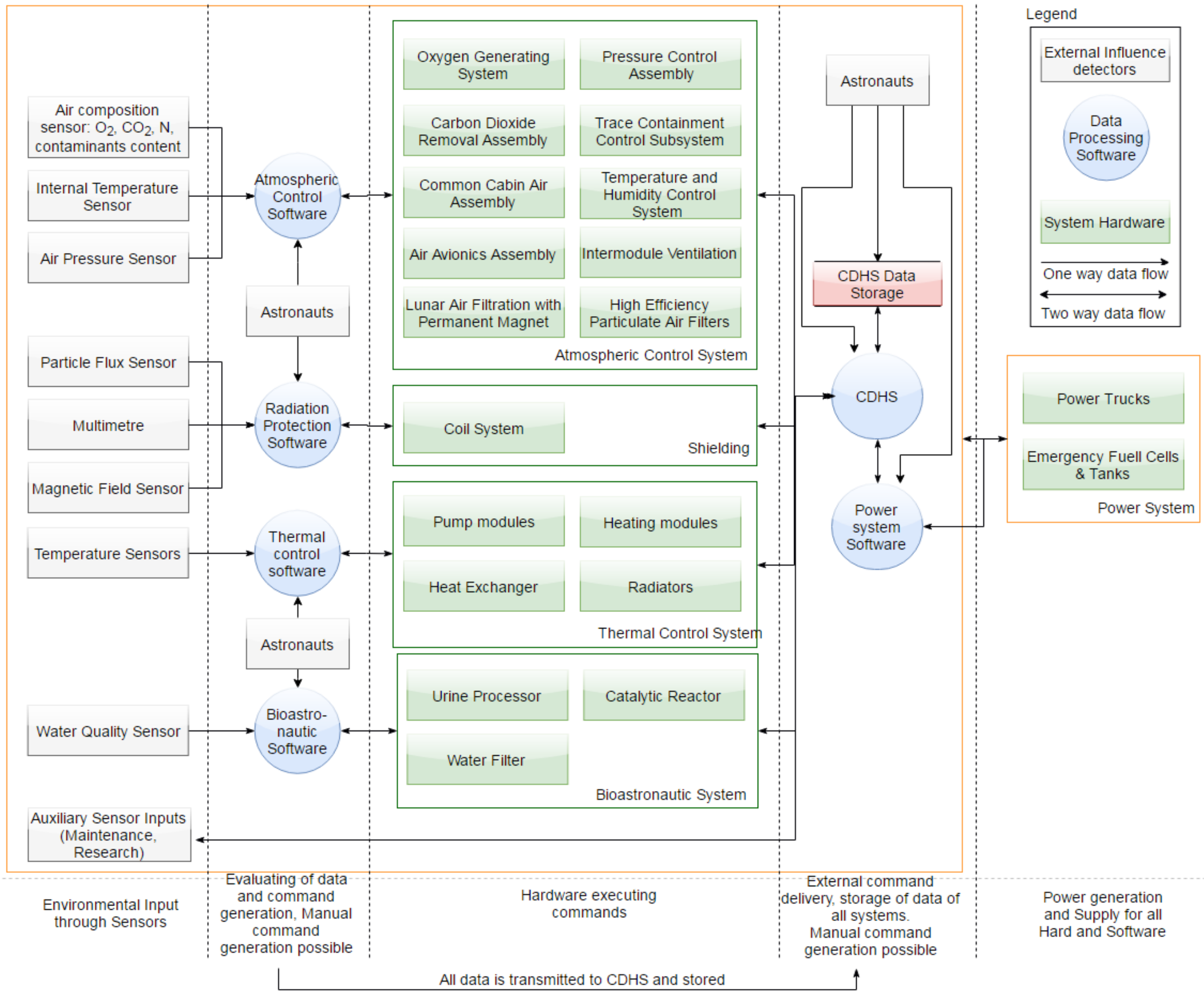


Figure 6.5: Hardware-Software interaction in the habitat system.

RAMS Characteristics and Sensitivity Analysis

The CDHS is deemed reliable since similar systems have been used on multiple spacecraft. However, the probability of the system working flawlessly for at least ten years is pretty small and it will require hardware and software update to either replace malfunctioning parts or to improve system performances. The technology for the RF communication links, wired optical data transferring, processing, analysing and saving are readily available. The technology for optical lunar communication is at TRL 7 and will have been advanced to at least TRL 8 and probably TRL 9 before final manufacturing of the habitat and its systems begins.
For the entire system, it is required that at least one of the astronauts has experience with photonics in order to maintain it. Furthermore, replacement of parts on the outside of the habitat and on rovers requires moonwalks by the astronauts. Replacements on the inside are straightforward and do not require a lot of physical effort.
The CDHS will have a common user interface and will thus be easy and safe to operate.

The design of the communication system is not heavily influenced by significant design changes to other systems. The most profound change would be relocating the antennas and providing other means of elevating them instead of simply putting them on top of the habitat and the SBR's.
The data system is subject to change when future modules will be added. Increasing the number of connections and data rates internally will only imply expanding the hardware installations until the transmittable data limit is reached.

7
Interior Design

Interior design is known for its artful integration of necessities within a living space and is often of secondary importance when technical issues are addressed. For the lunar habitat, however, the interior is of importance, as it plays a vital role in the astronauts' mental health and social well-being which, in turn, are closely linked to work performance and concentration. This chapter presents the discussion and outcome of the interior design phase. The purpose of the interior design is addressed in section 7.1. Furthermore, interior spacing and mass calculations are performed in section 7.2 while a design is proposed in section 7.3.

7.1. Purpose of Interior Design

In the following, several issues encountered during past space missions are considered and countermeasures are proposed.

Mental Health and Living Environment

The mental health of the astronauts is of vital importance and has been under considerations for many space missions. Prior space missions revealed that the astronauts often complained about the living space provided being sterile, and not accommodating [41].
One possible means of making the astronauts feel comfortable is granting the permission of personalising private and work spaces. Personalisation may include adding decoration, plants or personal items but it may also be extended to moving furniture.
Furthermore, due to the long mission duration during which the astronauts spend most of their time within the habitat, a sense of boredom of the interior and even the feeling of being trapped in a small space can occur [42]. This issue leads to a need for a design which gives the illusion of being in a large space. This can be achieved by merging rooms instead of placing walls in between such as the dining and common room. Room dividers such as shelves and cupboards can divide the room effectively without reducing the spacious effect. The appropriate application of mirrors can also help giving depth to a room. This is common practice in stores and may be applied to the habitat as well.

Social Environment and Privacy

An inviting social place facilitates social activities and thus, aids in group bonding. This is of vital importance as personal conflicts can hinder the workflow and may have an impact on work quality. A positive social environment and frequency social interactions ease the communication between crew members and therefore, facilitates discussions about issues and finding of solutions. Therefore, a large common room with entertainment and other means of leisure need to be employed. In contrast to that, the privacy of all astronauts is of utmost importance. This does not only include privacy in lavatories and for personal hygiene but also means of retreat. Thus, to ensure basic privacy, all rooms containing sanitary facilities as well as the sleeping quarters can be locked. Furthermore, each crew member has its own sleeping quarters including storage. Lastly, the compatibility of the crew members is an important criterion. It is suggested that the astronauts should be chosen by not only their technical qualities but also by their social compatibility with each other. This can serve as an indicator of how well they can work and live with each other on a small space [42].

Space, Mass and Cost Saving

A smart design can reduce the amount of space needed which may result in a mass and therefore, cost reduction. For this, multipurpose furniture, tools and locations can be employed. Foldable tables and working surfaces are a typical example creating a multipurpose location.

A particularly interesting idea is being researched by ESA and involves a thermoplastic, called Polyether Ether Kethon (PEEK), serving as a multipurpose material [43]. Due to its high strength and electrical conductivity, it can be used for nearly any purpose. The main idea is the use of a 3D printer to create tools, equipment and smaller parts of the furniture with the same material. This leads to the possibility of cannibalising broken or not used materials.
A big advantage of 3D printing concept is the ease of maintenance. Broken items may be readily fixable using the 3D printer while unfixable items can serve as raw materials. Furthermore, only one material needs to be transported to the Moon which will save weight and volume in the payload and thus, reduce cost.

Launch Considerations

In order to transport the furniture and other interior items to the habitat site, all items need to be fixed. The items and attachment need to withstand all launch loads for mission success.

For the hard-shell module, all furniture needs to be fixed and stored during launch. In the inflatable, buses are present which may be filled with furniture and other items during launch. Furniture which does not find space in the buses needs to be sent with supply missions. Then, the furniture needs to be sized in such a way that it fits through the airlocks.

Rising Issues

With all mentioned interior design aspect, several contradictions can be found:

Spacious Habitat and Space Reduction: As discussed above, for a good mental health, astronauts require spacious rooms as they provide a sense of freedom. However, extra room generates more mass and, therefore, is a large cost factor. In order to come up with a sufficient design, an appropriate space to cost ratio has to be found. The main concern is the minimum living area which is subject to personal impression.

Social Interaction and Privacy: Social interactions can be intimidating and difficult. Particularly during disputes. In the case of personal disputes, people react differently: some are searching for a discussion to find a solution while others retreat and never talk about it again. An unhealthy social environment leads to the astronauts being tense and may result in a decrease in work performance and quality. To ensure mission success, it is crucial to diminish any disputes as soon as possible. Privacy and social areas can play a vital role in dispute resolution: The more comfortable the private quarters are, the less likely the astronauts are to leave them and interact with the rest of the crew. Therefore it is argued that the private quarter should contain as little as possible means of entertainment [43]. On the other hand, however, the astronauts have the right of retreat and spend their free time as they please. Thus, a good balance between social and private quarters have to be found to facilitate a good social environment while still providing an adequate amount of privacy.

Fixed and Movable Furniture: For the sake of mental health, moving the furniture in the living space is given as a means of personalisation. However, due to safety concerns and launch fixation (hard-shell only), heavy or large furniture should not be free standing without any means of fixation. Therefore, a temporary fixing method has to be found.

7.2. Interior Spacing and Mass Calculations

Before the interior setup of the lunar habitat can be designed, a logistical bookkeeping including all the furniture will be made. The actual layout of the interior will be shown and elaborated on in section 7.3.

Two aspects will be important for the interior design: the quantity of each piece of furniture, and the associated mass. First, each piece of furniture will be categorised in interior subsystems, after which they will be quantified based on research about the needs of humans, and interiors of previous space habitats. This can be seen in Table 7.1. Some things should be noted, however: ”other entertainment” consists of multiple smaller objects which provide leisure for the astronauts. Furthermore, sleeping bags and inflatable mattresses are provided for emergency situations, when the astronauts can not access the living inflatable anymore. Additionally, ”extra cabinets” will store all objects except for food, e.g. medical kits, tools, foldable furniture etc. Finally, 12W LED lamps will be used for illumination inside the lunar habitat since they are very efficient and durable. The mass is taken into account in the ”other” subsystem, while the power consumption is taken into account in chapter 8.

Table 7.1: The categorisation of each piece of furniture, including quantity.

Subsystem + part	The Shell	The Nest	The Hive	Total
Sleeping facilities				
Bed	0	4	0	4
Closet	0	4	0	4
Cupboard	0	4	0	4
Sleeping bag	5	0	0	5
Inflatable mattress	4	0	0	4
Sanitary facilities				
Toilet	1	1	0	2
Shower	1	1	0	2
Sink	1	1	0	2
Cabinet	1	1	0	2
Consumption facilities				
Table	1	1	1	3
Chair	4	4	1	9
Tableware	1	0	0	1
Other				
Walls [m]	6	24	10	40
3D printer	0	0	1	1
Lighting	9	14	14	37

Subsystem + part	The Shell	The Nest	The Hive	Total
Leisure				
Couch	0	1	1	2
Armchairs	0	2	2	4
Projector	0	1	0	1
Other entertainment	1	0	0	1
Exercise facilities				
Cycle ergometer	0	0	1	1
Rubber yoga mat	0	0	1	1
Threadmill	0	0	1	1
Storage				
Food Cabinets	1	1	5	7
Extra cabinets	4	2	10	16
Space suit storage	4	0	1	5

Following this, an estimation has to be made of the mass of each part of the furniture. To be able to keep the total mass of the interior low, most of the hard furniture will be made from lightweight materials, such as the previously mentioned 3D printable PEEK. This means that the pieces of furniture which are used in the lunar habitat, will be lighter compared to their counterparts designed for Earth[1]. An estimation is made of each piece of furniture, based on the average weight of space applied furniture, and the ratio between the rough density of the actual furniture and the density of PEEK. A summary of the logistic accountancy, including the mass of all subsystems of the interior in each module, is shown in Table 7.2.

Table 7.2: Summary of the logistic accountancy of the interior system.

Interior Subsystem	**Mass hard-shell [kg]**	**Mass living infl. [kg]**	**Mass storage infl. [kg]**	**Total Mass [kg]**
Sleeping facilities	21	228	0	249
Sanitary facilities	41	41	0	82
Exercise facilities	0	0	117	117
Leisure facilities	15	65	55	135
Consumption facilities	41	31	19	91
Storage facilities	35	21	105	161
Others	22	86	337	445
Total interior system	175	472	633	1280

[1]http://democracy.york.gov.uk/documents/s2116/Annex%20C%20REcycling%20Report%20frnweights2005.pdf [Cited: 07-06-2017]

7.3. Interior Design Proposal

Once the quantity, dimensions and mass of every piece of furniture are known, together with the dimensions of the various other subsystems which will be located inside the modules, an interior design proposal can be made. This will consist of a proposed lay-out for the lunar habitat, which takes all previously mentioned considerations for mental health and social environment into account, as well as research performed on deep space habitat interior design [44].

The Shell

When designing the interior for The Shell, attention should be paid to several aspects. In the first place, two airlocks are present in this module, which take up a significant surface area inside the hard-shell dome structure. Next to this, The Shell will function as a safe room in case of emergencies such as solar particle events, which means the module has to be prepared at every moment in time to accommodate four astronauts for a certain duration of time. The way this safe room is realised can be seen in Figure 7.1.

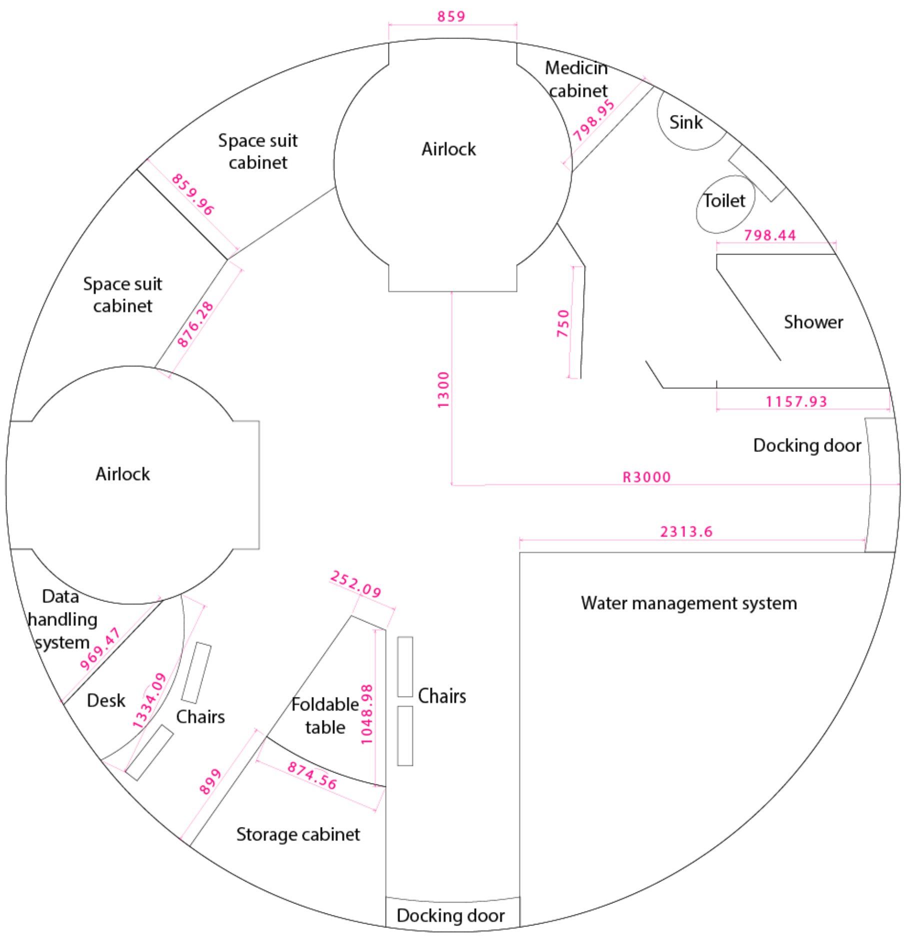


Figure 7.1: The ground plan of the The Shell.

The Shell will contain sanitary equipment, such as a toilet, a sink and a shower. These are deemed necessary for an extended stay of four humans in deep space. The shower will be a fully closed off cabin, to prevent the loss of water from splashing up from the bodies of the astronauts, due to the lower gravity. The sink will also be designed in such a way, that this effect will be minimised as much as possible. Furthermore, a full water management system will be integrated into The Shell, to provide clean water at all times for the astronauts. This water management system is located close to the shower and other sanitary equipment, aiming to reduce the complexity of tubes and pipes in this module. Next to the water management system, the atmospheric control system will also be present in this module, integrated into the floor and ceiling.

The space suits will be stored in cabinets between the two airlocks, where extra space is available for additional tools and systems, which will be used during Moon walks. This way, it will be easier for the astronauts to leave the habitat once their space suits are on.

The data handling and communication system will be located in this module, including a working desk at which the astronauts can contact Earth and communicate with ground control. Lastly, a bigger storage cabinet will be located inside The Shell, where a food storage will be present, together with sleeping bags, foldable chairs and other (emergency) tools and objects. From this larger cabinet, a multipurpose table can be folded out. During emergencies, where the astronauts have to live only inside this module, some inflatable mattresses can be pulled out of this storage cabinet. The Shell is designed in such a way, that there is enough space for four astronauts to lay out their sleeping bags.

From The Shell, the astronauts can walk into both inflatable modules through docking doors, on the opposite sides of the two airlocks.

The Nest

The inflatable modules will both have an interior living area of 50m^2. The four bedrooms will be located inside The Nest. These are designed in such a way that an astronaut will be able to live there for one full year.

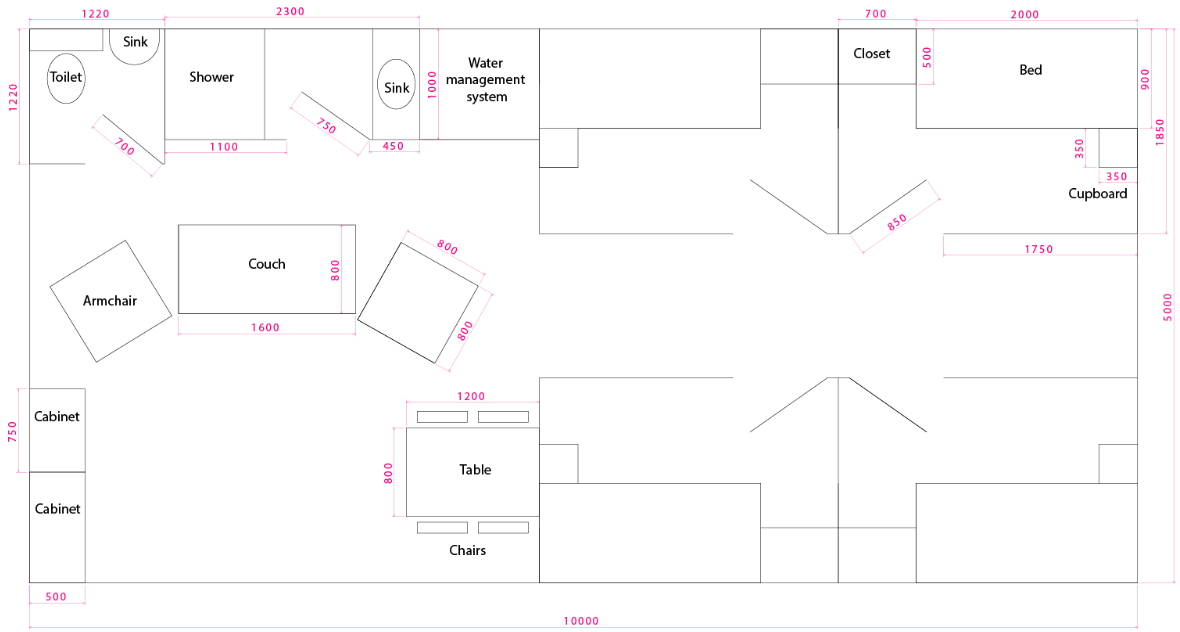


Figure 7.2: The ground plan of The Nest.

The furniture namely, which consists of a bed, a small cupboard and a closet, is modular and fully composable. This means, that the layout does not necessarily have to stay the way it is shown in Figure 7.2 for the entire ten years. The bed, for example, can be put on 3D printed long legs, creating space underneath it to either store objects, or as a working/leisure area.

Next to the four bedrooms, a living space opens up. In here, the astronauts will be able to eat food and relax on the soft couch and armchairs. A projector will be available inside the habitat, which can provide entertainment in the form of films and series, but also project scenes on the walls from nature and other phenomena on Earth, to relieve the psychological pressure on the astronauts during their one year stay. These armchairs and couch will be designed in such a way, that they can be moved towards the lower wall of The Nest in a folded position, to create space during other periods of the day.

A toilet and a shower are also present in this module, to create redundancy, but also to create comfort for the astronauts. Lastly, a part of the water management system of this module will be above the floor, while the other part will be positioned under the floor. This water management system is located next to, and under the sanitary facilities, following the same principle as in The Shell. The atmospheric control system, which will be present in each module, is located under the floors in the inflatable modules.

The Hive

The second inflatable module will contain both the main storage of the lunar habitat, as well as the sport and exercise equipment for the astronauts. This equipment is necessary to prevent muscle deterioration during the long duration of decreased gravity.

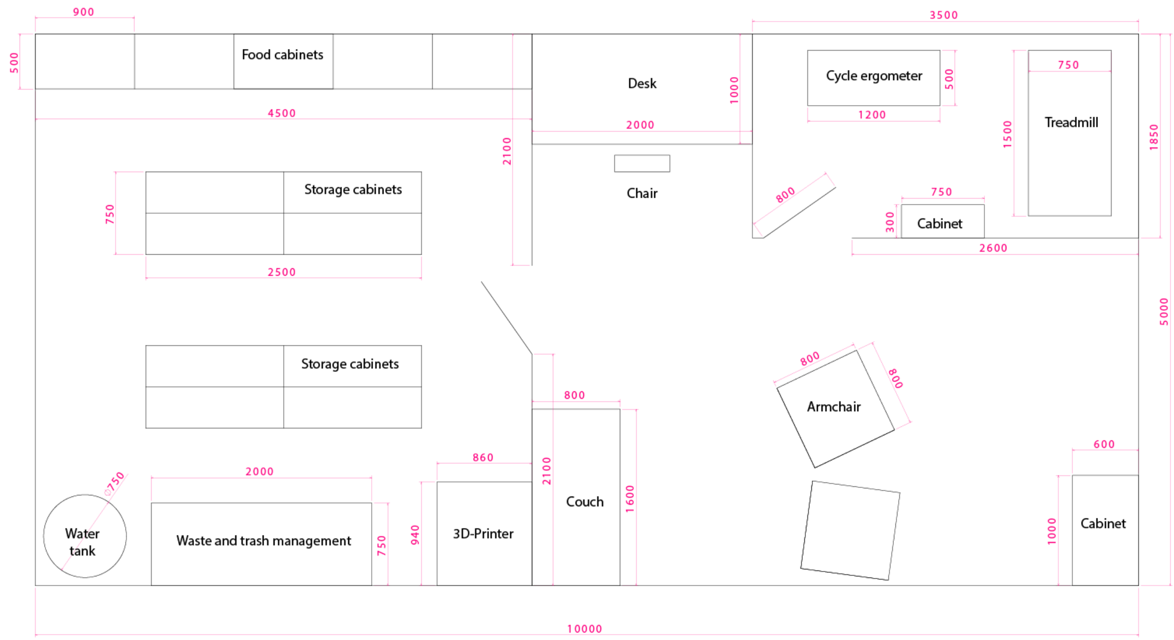


Figure 7.3: The ground plan of The Hive.

A specially designed treadmill and cycle ergometer for lower gravitational acceleration will be located in this room. The storage area, which can be seen in Figure 7.3, contains a large food storage, as well as an emergency water tank.

Furthermore, there are storage cabinets for tools, spare parts, an extra space suit, and bulk material of PEEK, which can be used to create parts using the 3D printer. The 3D printer which will be used for the lunar habitat can print parts of 45x45x60cm in volume. This is decided after a trade off between different 3D printers was made since printers with a larger build volume are automatically heavier and larger in dimension, which becomes unacceptable for the lunar habitat mission. Smaller and lighter printers, on the other hand, have a very small build volume and thus serve no purpose for a lunar habitat mission.

The waste and trash storage management system is also located in this room, next to the 3D printer. Furthermore, The Hive contains another set of armchairs and a couch, which is again foldable. This set of furniture not only gives redundancy but also comfort to the astronauts, when they want to leave the ambience of the living quarter. A working desk, including extra cabinets, concludes The Hive.

RAMS Characteristics and Sensitivity Analysis

PEEK as a material is quite well known, and much research has already been performed on it. This means that the physical properties of PEEK are well documented. Since most of the interior inside the lunar habitat will be made from this material, a thorough analysis can be made on the RAMS characteristics of the interior.

The reliability of the interior can be seen as high since the material of which it is made is a strong and tough material. This means that the chance that a table or chair will break, will be quite small, even more on the lunar surface with decreased gravity. The softer materials which will be used for the beds and sofas for example, will have to be made of a strong and durable material.
The availability of the materials here on Earth is abundant, which means that this will not be a problem for the interior of the lunar habitat. On the Moon, however, these materials are not present, which means that everything has to be brought from Earth. This will not be a problem since the design philosophy is to anyway bring a piece of bulk material, which can be 3D printed into parts in the lunar habitat. This increases the availability for each furniture part made out of PEEK, since it can always be 3D printed there, without the need of a specific resupply mission to bring that piece of furniture.
Similar to the availability, the interior is designed in such a way that the maintenance of the furniture is very easy. A designed model of the broken part will just be sent to the 3D printer, after which it can be mechanically placed to repair the broken piece of furniture. Furthermore, the layout of the furniture is proposed in such a way, that other subsystems are easily reachable, and thus facilitates the maintenance of these subsystems.
Finally, the interior is designed using materials and geometries, which enhance the safety inside the lunar habitat. The PEEK material for example, has low out-gassing[2] and flammability properties [45]. The interior contains small and larger rooms and areas, which may cause decreased safety in case of emergency or panic. This can be prevented by setting up thorough emergency plans for the lunar habitat.

There are multiple possibilities for the interior surface area to change in size. If the shielding layers of the environmental protection system in The Shell become thicker, the inner size of the habitat will become smaller. This is because the outer diameter is constrained to the inner diameter of the payload fairing of the SLS launch vehicle. The shielding layers can thus only increase inwards. This decrease in surface area and volume will have a major impact on the layout of the interior. It can cause some systems or pieces of furniture to not fit anymore inside the habitat.
Furthermore, if PEEK appears to be unusable for inside the habitat, another material has to be chosen. This could have implications for the total mass of the interior if the density and the specific strength of the material differ significantly.

[2]http://www.boedeker.com/outgas.htm [Cited: 23-06-2017]

8
Power System Design

In this chapter the concept, sizing and diagrams concerning the power system are treated. Firstly, the chosen concept will be explained along with its design philosophy and reasoning in section 8.1. Secondly, section 8.2 explains the sizing of every part and the number of chosen components. Thirdly, section 8.3 explains the Electrical Block Diagram of the power modules and the habitat.

8.1. Concept

During the detailed design of the power system, two main design concepts were fully worked out. Both designs entail power generation by solar arrays and energy storage by hydrogen and oxygen, which will provide power during the lunar night by Proton Exchange Membrane(PEM) fuel cells. The main concepts were to either have one central power system that would have to be sent in a separate launch or to have a modular power system that would be sent in multiple launches with other rovers, systems and/or modules. The main advantage of having a modular power system is the redundancy: in case one of them would fail, there will be sufficient power left for the main systems. In this report the sizing is done on the chosen design only.

A modular design of the power subsystem was the most preferable. A consequence of this is that the weight of the power system of the entire habitat becomes 22% heavier. However, the system will be much more redundant and can be scaled up easily when other modules are integrated into the lunar village. The amount of power modules is defined by the power demand of the first set of rovers that will be sent. This defines the minimum required peak power and average power that one power module would need to produce. Since the Power Rangers will be sent in several launches, including the first launch with sintering rovers, it is recommended that the Power Rangers can provide their own means of transportation and become actual "Power Rangers" so that they will be able to automatically drive from the landing site to the habitat. This also allows astronauts to have power consuming exploration or research missions outside the lunar habitat.

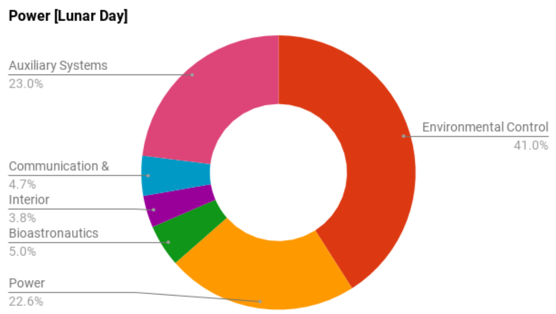


Figure 8.1: Power demand contribution of each subsystem during the lunar illumination time.

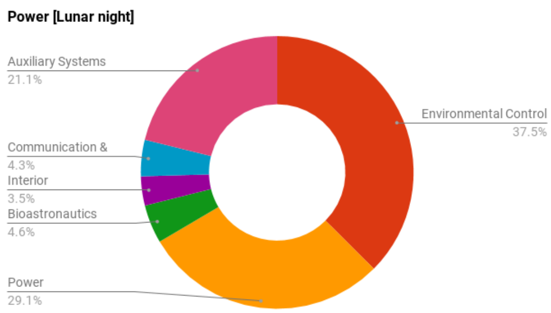


Figure 8.2: Power demand contribution of each subsystem during the lunar night.

In Figure 8.3 and Figure 8.4, the mass contribution of the components of the central power system and the modular power system can be seen, respectively. They are both designed for the power demand of the habitat, so the power output will be the same. The components that contribute in the mass calculation of the power system are water (to produce hydrogen and oxygen for energy storage), fuel cells/electrolyzer cells (to store electrical energy for during the lunar night), solar arrays (to provide sufficient power for the habitat and energy storage), storage tanks (for the hydrogen, oxygen and water), pumps (includes compressors, cryopumps and water pumps) and inverters (to convert the Direct Current (DC) output of the solar arrays and the fuel cells to an Alternating Current (AC) with a stable voltage). The structural contribution of the central design consists mainly of meteorite protection by a stuffed whipple shield, radiation protection and passive thermal control, while the Structural contribution of the "Power Rangers" consists of structural integrity and wheels. Since the redundancy of the modular power system is that high, the addition of a stuffed whipple shield would overdesign the system and make it unnecessarily heavy. The last component taken into consideration is the thermal control, this will be necessary to cool the pressure tanks while being pressurised and heat those tanks while being depressurised. Furthermore, the thermal control of the power system also ensures that the water, liquid hydrogen and oxygen does not freeze.

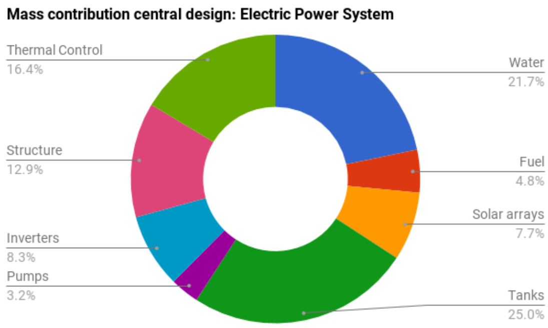


Figure 8.3: Mass contributions of the central power system design.

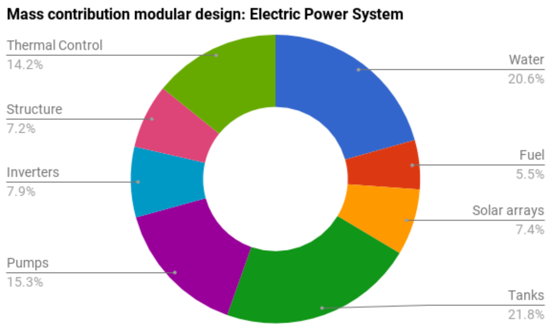


Figure 8.4: Mass contributions of the modular power system design.

8.2. Sizing

In this section, the sizing of the modular power system will be done. Since the design of the power system fully depends on the power consumption of the habitat, the average and peak power of every subsystem was calculated for lunar day and lunar night operations separately. An average power consumption during the lunar day and lunar night was calculated using a safety factor of 1.3. This value comes from the probabilities of storage tank penetration. If a meteorite hits an oxygen tank, half of the energy will be lost and if a meteorite hits a hydrogen tank, only one quarter will be lost. The average safety factor to be implemented was 1.3. The power contribution of each subsystem can be found in Figure 8.1 and Figure 8.2.

Table 8.1: Power demands for the habitat and its consequences on the required power characteristics of the five Power Rangers.

Inputs | Power [kW]	Total required	Module inputs
Average power day	13.82	2.76
Average power night	15.06	3.01
Peak power day	67.62	13.52
Peak power night	90.09	18.02

From the contribution of each subsystem, the total average and peak power demand can be calculated. To obtain the amount of power modules needed, the power demand of the set of rovers that will be sent with the first power module must be satisfied. The Power Rangers therefore need to be as small as possible, but big enough to satisfy the load demand of these first rovers. Using this approach, the lunar habitat would need five Power Rangers. The required power characteristics for a single power module can be seen in Table 8.1.

Water

Firstly, the amount of water needs to be estimated. This is necessary to have sufficient energy during the lunar night. To be able to relate water (from the reaction of hydrogen and oxygen in fuel cells) to an amount of energy, the bonding enthalpies[1] in $O_2 + 2\,H_2 \longrightarrow 2\,H_2O$ are summed. In this chemical

[1]https://chem.libretexts.org/Core/Physical_and_Theoretical_Chemistry/Chemical_Bonding/General_Principles_of_Chemical_Bonding/Bond_Ener
[Cite: 06-06-2017]

reaction, the $O = O$ and $H - H$ bonds are broken and $O - H$ bonds are created. The net enthalpy of this reaction is -509.00 kJ/mol as can be seen in Table 8.2. In this table it can also be seen how the enthalpy is converted to the specific energy of 3.9 kWh/kg of the water.

Table 8.2: From bonding energies on molecular level to the required specific energy of water.

Bonding energies [kJ/mol]		**Conversion to specific energy**		
H-H	432.00	M(H_2O)	18.02	g/mol
O-H	467.00	Energy per gram H_20	14.13	kJ/g
O=O	495.00	Energy per gram H_20	3.92	Wh/g
Enthalpy Broken bonds	1359.00	Specific energy H_20	3924.20	Wh/kg
Enthalpy Created bonds	-1868.00			
ΔH total	-509.00			

The energy required to be stored can be calculated with Equation 8.1. In this equation the required energy is calculated with the average lunar nighttime power consumption, the time of a lunar night and efficiencies of the fuel cells and the inverters that will be used during the lunar night. From the energy that needs to be stored, a mass of 477.12 kg of water is calculated in Equation 8.2.

$$E_{stored} = \frac{P_{night} \cdot t}{\eta_{fc} \cdot \eta_{inv_{fc}}} = \frac{3.01kW \cdot 354.36h}{0.6 \cdot 0.95} = 1872.31kWh \tag{8.1}$$

$$m_{water} = \frac{E_{stored}}{SpecificEnergy(H_2O)} = \frac{1872.31kWh}{3.9242kWh/kg} = 477.12kg \tag{8.2}$$

Solar Arrays

To size the solar arrays, the necessary solar power is calculated first. This is done with the peak power during the lunar day and the energy that would need to be stored for the lunar night, the time of a lunar day, the efficiency of the electrolyzer cells and the efficiency of the solar power inverter. From these values, the necessary power output for the solar arrays will be 21.44 kW as shown in Equation 8.3.

$$P_{solar} = \left(P_{peak} + \frac{E_{stored}}{t \cdot \eta_{el}}\right) \cdot \eta_{inv_{Sol}}^{-1} = \left(13.52kW + \frac{1872.31kWh}{354.36h \cdot 0.7}\right) \cdot 0.98^{-1} = 21.44kW \tag{8.3}$$

Now the amount of installed solar power is known, the necessary solar cell area can be calculated with the solar irradiance, the solar cell efficiency[2] and the degradation of photovoltaic cells over the timespan of ten years are taken into account. The Power Rangers will be provided with a multiple axis system to point the solar arrays at the Sun to obtain the optimal efficiency. As shown in Equation 8.4, the necessary calculated solar cell area is $58.05m^2$. With the size of the solar arrays known, finally the mass of the solar arrays can be calculated. This is done with the average cell weight and the weight of a two centimetre thick honeycomb plate that will be used to support the photovoltaic cells. It is shown in Equation 8.5 how the mass of the solar array is calculated.

$$A_{solarcells} = \frac{P_{solar}}{S_R \cdot \eta_{cell} \cdot \eta_{degradation}} = \frac{21.44kW}{1.361kW/m^2 \cdot 0.3 \cdot 0.99^{10}} = 58.05m^2 \tag{8.4}$$

$$\begin{aligned} m_{arrays} &= m_{cells} + m_{panels} = A_{solarcells} \cdot (m_{cell/area} + m_{panel/area}) \\ &= 58.05m^2 \cdot (0.86kg/m^2 + 2kg/m^2) = 116.10kg \end{aligned} \tag{8.5}$$

Fuel Cells & Electrolyzer Cells

Since there are no exact details on the mass of PEM fuel cells in space environment, commercially available fuel cell stacks of 1kW and 4kg[3] are used for the sizing of the Power Rangers. The same values are used for the PEM electrolyzer cells. The necessary amount of them influences the weight of the power module. Since the power module needs to handle peak power demands during the lunar night, sufficient fuel cells will be installed to satisfy the peak demand. The peak power during the lunar

[2]http://www.azurspace.com/images/0003429-01-01_DB_3G30C-Advanced.pdf [Cite: 07-06-2017]
[3]http://www.fuelcellstore.com/horizon-1000watt-fuel-cell-h-1000 [Cite: 07-06-2017]

night per module is 18.02kW, therefore 19 fuel cells will be needed. The amount of electrolyzer cells is dependent on the average amount of power that would be needed to store the hydrogen and oxygen. In case that the power required per module is not the peak power, some spare power from the solar arrays will be left. This can be used to have extra power to the electrolyzer cells. The probability that the power will be near the average power is high and the probability that the power will be near the peak power is low. Therefore, it is chosen that the Power Rangers will have four extra electrolyzer cells. Doing the calculation of Equation 8.6, the required amount of electrolyzer cells would be twelve. When the required power during the lunar day is not the peak power, these four extra electrolyzer cells can be powered. This would speed up the energy storage and the Power Rangers likely will have stored the required amount of energy before the end of the lunar day. With the number of fuel cells and the number of electrolyzer cells, the estimated mass can be calculated in Equation 8.7.

$$\#_{EC's} = int\Big(\frac{E_{stored}}{t \cdot \eta_{EC} \cdot P_{max_{EC}}}\Big) + 4 = int\Big(\frac{1872.31kWh}{354.36h \cdot 0.7 \cdot 1kW}\Big) + 4 = 12 \quad (8.6)$$

$$m_{FC\&EC} = (\#_{FC's} + \#_{EC's}) \cdot 4kg = (19 + 12) \cdot 4kg = 124kg \quad (8.7)$$

Inverters

To ensure a steady voltage, inverters are added to the Power Rangers. There will be two types of inverters; solar inverters and fuel cell inverters. The inverters convert the rough power output of the solar arrays and fuel cells with variable voltages and currents depending on the acting power to stable voltages. The solar inverters are also equipped with Maximum Peak Power Tracking (MPPT) technology to optimise the power output by adjusting the ratio of voltage and current of the solar cells. The maximum power the inverter[4] can have is 10.8kW. To satisfy the peak power demand, two solar inverters need to be installed. The fuel cell inverters[5] can convert up to 6kW. To ensure that the Power Rangers can handle the peak demand during the lunar night of 18.02kW, three of them need to be installed.
With a mass of 26kg for the solar inverters and a mass of 40kg for the fuel cell inverters the total inverter mass will become 212kg.

Pressure Tanks

Because the electrolyzer cells produce hydrogen and oxygen constantly, the Power Rangers will be provided with a sophisticated storage management system. The hydrogen and oxygen will be stored cryogenic, but the cryopump that compresses these gasses consumes too much power to operate continuously. In this system, a compressor will compress the hydrogen and oxygen into intermediate tanks up to 10 bars first. The compressors will operate continuously, but they require a lot less power than the cryogenic pump. When the intermediate tanks are filled, the cryopump compresses the 10 bars content into the main cryogenic storage tanks up to 700 bars. To know the dimensions of the tank, the amount of moles hydrogen and oxygen is firstly derived from the amount of moles of water. Using the molar mass of hydrogen and oxygen, the masses of the hydrogen and oxygen are calculated. With the mass density of liquid hydrogen and oxygen, the volumes were calculated in Table 8.3.

Table 8.3: Derivation of mass and volume of oxygen and hydrogen for the energy storage.

M(H_2)	2.02	g/mol	Mass (H_2)	53.39	kg
M(O_2)	32.00	g/mol	Mass (O_2)	423.73	kg
Moles (H_2O)	26500	moles			
Moles (H_2)	26500	moles	Volume (H_2) 700bar	0.75	m^3
Moles (O_2)	13242.27	moles	Volume (O_2) 700bar	0.37	m^3

To make the power system more redundant, the liquid oxygen will be stored in two tanks. When a meteorite would be able to penetrate the walls of the power module and strike an oxygen tank, the power module will still be able to provide half its stored energy. To make the system even more redundant, the liquid hydrogen will be stored in four tanks, then the tanks for hydrogen and oxygen storage can

[4]http://www.solar-inverter.com/en-GB/6627.htm [Cite 10/06/17]
[5]http://cspower.en.made-in-china.com/product/nXtJRTAvVKhI/China-8000W-72V-Inverter-Charger-DC-to-AC-Power-Inverter.html [Cite 10/06/17]

have the same dimensions. When a hydrogen tank would be struck by a meteorite, the power module shall still have 75% of its initial energy storage. Fortunately the probability that this will happen is very low and there are five Power Rangers powering the habitat, which will make the overall energy storage drop with maximum 10% for which the Power Rangers will be designed. The six cryogenic storage tanks will have a storage of $0.19m^3$ and a diameter of 40cm. With these constraints the length of the tanks is calculated and will be 1.23m. With these values the thickness of these tanks can be calculated in Equation 8.8. The material of preference will be carbon fibre because of its high yield stress and low mass density.

$$t_{tank} = \frac{P_{internal} \cdot r}{\sigma_y} = \frac{700 \cdot 10^5 Pa \cdot 0.2m}{600 \cdot 10^6 Pa} = 0.023m = 23mm \tag{8.8}$$

From this thickness, the required amount of volume of the carbon fibre composite can be calculated, which is $0.05m^3$. From this volume, the weight of the tanks can be calculated using a mass density[6] of $1600kg/m^3$, which results in a tank mass of 82.27kg. The two sets of two intermediate tanks will have a volume of $0.24m^3$ each and a diameter of 40cm, which will make the length of the intermediate tanks 64cm. The calculation of the necessary thickness is also done with Equation 8.8, but with 10bars instead of 700bars of pressure. The necessary thickness for the intermediate tanks will then be 0.208mm which results in a composite volume of $8.04 \cdot 10^{-5}m^3$. The mass of a single intermediate tank will then be 0.13kg. Now that the masses of the high pressure tanks and the intermediate tanks are calculated, the total mass of the tanks is shown in Equation 8.9.

$$m_{tanks} = 6 \cdot m_{cryotank} + 4 \cdot m_{intermediate} = 6 \cdot 82.27 + 4 \cdot 0.13kg = 491.88kg \tag{8.9}$$

Pumps

There will be three different kind of pumps: compressors (to pressurise the continuously outflowing gasses from the electrolyzer cells), cryopumps (to store the gasses coming from the intermediate pumps in cryogenic conditions) and water pumps (to supply water to the electrolyzer cells during the lunar day). In Table 8.4, the average gasflows are calculated to be able to determine the duty time of each pump. The compressors will work all the time, because the gasflow will be constant. The time it takes to fill the intermediate tanks has been calculated as well as the time it takes to empty the intermediate tanks with the cryopump. The ratio of the fill time and the total time defines the duty of the cryopumps.

Table 8.4: Volume flows of hydrogen and oxygen during average lunar nighttime power consumption.

n(H_2)	1.81E-02	mol/s	Time to fill intermediate H_2 tank	11.57	minutes
n(O_2)	9.04E-03	mol/s	Time to fill intermediate O_2 tank	23.15	minutes
Volume flow H_2	5.76E-04	m^3/s	Time to empty intermediate tanks	12.00	seconds
Volume flow O_2	2.88E-04	m^3/s	Duty H_2 cryopump	0.02	-
Volume flow H_2 10bar	5.76E-05	m^3/s	Duty O_2 cryopump	0.01	-
Volume flow O_2 10bar	2.88E-05	m^3/s	Duty H_20 pump	6.73E-04	-

Each power module will be provided with three waterpumps for redundancy, and four compressors; one for each intermediate tank. Also, three cryopumps will be installed; one will be a spare one which will automatically connect to either the hydrogen or oxygen storage system, when one of the pumps would have a malfunction or failure. With a mass of 91kg for the cryopumps[7], 10kg for the compressors[8] and 7.6kg for the waterpumps[9], the total mass estimation for the pumps inside a power module is 333.40kg.

[6]http://www.performance-composites.com/carbonfibre/mechanicalproperties$_2.asp[Cite : 08-06-2017]$
[7]http://ridl.cfd.rit.edu/products/manuals/CTI/9600Compressor.pdf [Cite: 12-06-2017]
[8]http://sigma-electronic.com/products/oil-free-compressors/low-power-compressors/ [Cite: 12-06-2017]
[9]https://www.tuinslangcenter.be/product/572084/category-214114/metabo-p-2000-g-tuinpomp.html#product_specifications [Cite: 12-06-2017]

Thermal Control

The thermal control of the Power Rangers will ensure that the cryogenic tanks are kept at desirable temperatures. When the gasses are compressed, the temperature will increase. Therefore, the storage tanks will need to be cooled when they are being filled, and heated when the hydrogen and oxygen are flowing out, to prevent them from freezing while depressurising them. Furthermore, the solar arrays need to be cooled during the lunar day to increase their efficiency and the water inside the water tank may not freeze. To isolate the Power Rangers, a 2cm layer of aerogel will be used, which has an extremely low thermal conductivity in vacuum and an extremely low mass density. To know exactly how much power needs to be dissipated from the storage tanks, the ideal gas law was used. From this law, it is clear that the temperature difference will be the highest when the pressure in the tanks is the highest. Using the average molar mass flow values of Table 8.4, the temperature difference can be calculated, considering that keeping the storage in liquid state would require a maximum temperature of 236.13K. For simplicity the sizes of the oxygen and hydrogen tanks are considered the same but because they will be stored at the same temperature, the pressure in the two cryogenic tanks will be different. The maximum pressure for the hydrogen tanks will be 700 bars and the pressure on the oxygen tank will be 689.50 bars. For simplicity, the amount of compressed gasses will be the last 30^{th} of the lunar daytime gas production and the result can be seen in Equation 8.10 and Equation 8.11. Of course the temperatures of 7045.11K and 7057.02K are not meant to be interpreted literally, because the heat produced of these incoming gasses will directly be distributed to the cryogenic oxygen and hydrogen throughout the whole tank. But with this temperature, the energy coming into the pressure tanks can be calculated.

$$T_{H_2} = \frac{p_{H_2} \cdot V_{H_2}}{n_{H_2} \cdot R_i} = \frac{689.50 \cdot 10^5 Pa \cdot 0.75 m^3}{882.82 mol \cdot 8.3145 J/(molK)} = 7045.11K \tag{8.10}$$

$$T_{O_2} = \frac{p_{O_2} \cdot V_{O_2}}{n_{O_2} \cdot R_i} = \frac{700 \cdot 10^5 Pa \cdot 0.37 m^3}{441.41 mol \cdot 8.3145 J/(molK)} = 7057.02K \tag{8.11}$$

The power needed to cool the tanks is calculated with the required temperature difference, the thermal capacity of hydrogen and oxygen[10], the mass of hydrogen and oxygen that will be stored during the last 30^{th} of the lunar night and the time of this period. In Equation 8.12 and Equation 8.13, these values are used to calculate the thermal power that needs to be dissipated. Furthermore, it is assumed that the tanks would require an equal amount of heating power during the lunar night to prevent the liquid storage to freeze. It should also be noticed that the power values are peak values and are linearly dependent of the pressure which is at most 700 bars. Therefore, the average power dissipation or heating is calculated by a program that calculates the heat that needs to be dissipated at every instant of the lunar day, summed those values up and divided by the time of the lunar day. This is how the average power dissipation or heating of 0.45kW was found.

$$P_{H_2} = (T_{in} - T_{cryo}) \cdot \frac{m_{H_2} \cdot C_{H_2}}{t} = (7045.11K - 236.13K) \cdot \frac{1.78kg \cdot 14.12kJ/(kgK)}{(354.36/30) \cdot 3600s} = 4.02kW \tag{8.12}$$

$$P_{O_2} = (T_{in} - T_{cryo}) \cdot \frac{m_{O_2} \cdot C_{O_2}}{t} = (7057.02K - 236.13K) \cdot \frac{14.12kg \cdot 0.91kJ/(kgK)}{(354.36/30) \cdot 3600s} = 2.06kW \tag{8.13}$$

During the lunar night, the hydrogen and oxygen will flow from the storage tanks through a heat exchanger to the fuel cells. The temperature difference will be the difference between the storage temperature and the required operation temperature of the fuel cells. The heating of these gasses is calculated considering the average power consumption during the lunar night, therefore the average massflows of the gasses are calculated and put into Equation 8.14 and Equation 8.15. The combined heating power will then be 84.27W. If the same ratio of peak power mass flow over average mass flow is used to approximate the peak power of the heater, this maximum power will be 504W.

$$P_{H_2} = (T_{FC} - T_{cryo}) \cdot C_{H_2} \cdot \dot{m}_{H_2} = (333.15K - 236.13K) \cdot 13.53kJ/(kgK) \cdot 4.19 \cdot 10^{-5} kg/s = 54.94W \tag{8.14}$$

$$P_{O_2} = (T_{FC} - T_{cryo}) \cdot C_{O_2} \cdot \dot{m}_{O_2} = (333.15K - 236.13K) \cdot 0.91kJ/(kgK) \cdot 3.32 \cdot 10^{-4} kg/s = 29.33W \tag{8.15}$$

[10]http://www.engineeringtoolbox.com/hydrogen-d_976.html [Cite 13/06/2017

$$P_{H_2O} = (T_{FC} - T_{room}) \cdot C_{H_2O} \cdot \dot{m}_{H_2O} = (333.15K - 295.15K) \cdot 4.18kJ/(kgK) \cdot 3.74 \cdot 10^{-4}kg/s = 67.22W \tag{8.16}$$

From the fuel cells, hot water can be obtained and this water can be guided through the heat exchanger, to partially reuse the heat of the water to heat the incoming hydrogen and oxygen. The power transfer in the heat exchanger from the water to the hydrogen and oxygen is calculated in Equation 8.16. The water will still have a temperature of 295.15K after going through the heat exchanger, to be able to reuse the last heat of the water to keep the water in the watertank at a desirable temperature during the lunar night. By using this heat exchanger, the average power for heating the hydrogen and oxygen will be reduced from 84.27W to 17.05W.

For sizing the thermal control, the thermal control system of the ISS was used to estimate the mass and size of the radiators. The maximum heat that can be dissipated from one of the radiators of the ISS is 14kW and the mass of the radiator and its control unit combined equals 847.40kg. This means that this control system has an average mass fraction of 60.53kg/kW of dissipating heat. From summing up all thermal heats that need to be dissipated, it can be concluded that the peak amount of heat that needs to be dissipated for one power module is 6.08kW. Multiplying this value with the mass fraction of the ISS radiator results in a mass of 368.02kg.

Structural Integrity

By using the modular concept for the power system, a fully stuffed Whipple shield for meteorite protection will not be necessary, because sufficient redundancy will be applied within each power module and there will be extra redundancy in the number of Power Rangers. However, even though the mass of the structure will be lower due to the lowered meteorite protection, there will be a mass increase due to the fact that the Power Rangers will have the undercarriage of a rover with six wheels. It also needs to have sufficient structural integrity, such that vibrational loads will not cause the module to break while driving. The Power Rangers will have two maximum design speeds at which they will be able to travel. The lowest maximum speed is when the solar arrays and thermal radiators are fully deployed, since this will decrease the critical structural integrity. The higher maximum speed of the Power Rangers can be achieved when the solar arrays and the thermal radiators are retracted. This can be done only temporarily during the lunar day. After a couple of hours, the systems will require cooling and the thermal radiators need to be deployed again. During the lunar night, heat losses through the walls will compensate the heating of the systems. Therefore, the Power Rangers will be able to travel at its higher maximum speed as long as it has sufficient hydrogen and oxygen left. For simplicity, the structural mass is calculated by summing up the estimated mass of the structural integrity and the estimated mass of the wheels. The mass of the structural integrity is estimated to be 5% of the total mass, assuming that lightweight materials and structures will be used. In addition to that, the estimated weight of the wheels including electric motors will be 8kg per wheel, 48kg in total. By combining these two masses, a total structural mass of 151.47kg will be assumed.

Power Consumption

The power consumption of the power system is mainly dependent on the thermal control of the storage tanks, the preheating of the hydrogen and oxygen and the power of the pumps. The power consumed for transporting the Power Rangers will be negligible in comparison to the systems that have to operate at a more frequent rate. For the cooling in thermal control, the power fraction has been calculated by using the ISS as a reference, which is 50.91W(dissipated)/W(consumed). When dividing the average heat to be dissipated by this fraction, an average consumed power of 7.60W is calculated. For the heating of the storage tanks during the lunar night, 0.39kW of thermal power is required, which will be provided using resistive heating inside the heat exchangers.For the cooling of the solar arrays, the power consumption of the ISS is assumed. However, as the power output of the solar arrays of the Power Rangers is considerably smaller than the solar arrays of the ISS, it is assumed that for one power module, the power consumption required for cooling the solar arrays of the ISS is divided by the number of Power Rangers that will be sent, which is on average 55W. The average power consumption of the pumps can be calculated by multiplying the duty rate of each pump that has been calculated in Table 8.4 with their maximum power output. The compressors will work all the time while producing hydrogen and oxygen, therefore their duty rate will be 1. Using this approach, the average power consumption of the hydrogen cryopump is 82.81W, the average power consumption of the oxygen cryopump is 41.40W and the average power consumption of the water pumps is 0.26W. The power consumption of a compressor is 0.55kW. Two of them will be needed to compress the hydrogen and the oxygen, the other two compressors are installed as redundancy and will start functioning when the other compressor fails or has a malfunction. Therefore, only two out of four compressors will contribute to the power consumption. Summing up all previous average power consumptions related to the pumps, an average pump power consumption of 1.22kW can be obtained.

The total average power consumption is the sum of all previously mentioned powers. This is found to be 1.67kW and this power is directed to the outputs of the power system. This often resulted into a new power input as well, therefore the power system had to undergo multiple iterations until the deviation of the new power input would be less than 0.01kW. Of course the values used in the power sizing section were already the results of this iteration, which makes the sizing of the Power Rangers definite.

8.3. Electrical Block Diagram

The Electrical Block Diagram of a power system gives a clear overview on how the power provision will satisfy the power demand. In Figure 8.5, the diagram of a single power module is shown. During the lunar day, the solar arrays will provide all necessary power. The power will flow from the solar arrays to the solar inverters, to have an input with a stable voltage in the Power Control Unit (PCU). The PCU will prioritize the power output over the Electrolizer Cells (EC's), considering that the power output does not demand a higher power than the maximum peak power for which the power module is designed. This will make the EC's dynamically balanced with the power output, when the power output will be less than the produced power by the solar arrays, as more power will go to the EC's. The EC's convert power and water into hydrogen and oxygen which will go to compressors first. Those compressors will store the hydrogen and oxygen in intermediate tanks up to 10bars. When they are fully filled, the content of those intermediate tanks will go through cryopumps into cryogenic storage tanks. All pumps, compressors and pressure valves will be controlled by the Pressure Control Unit (PrCU).

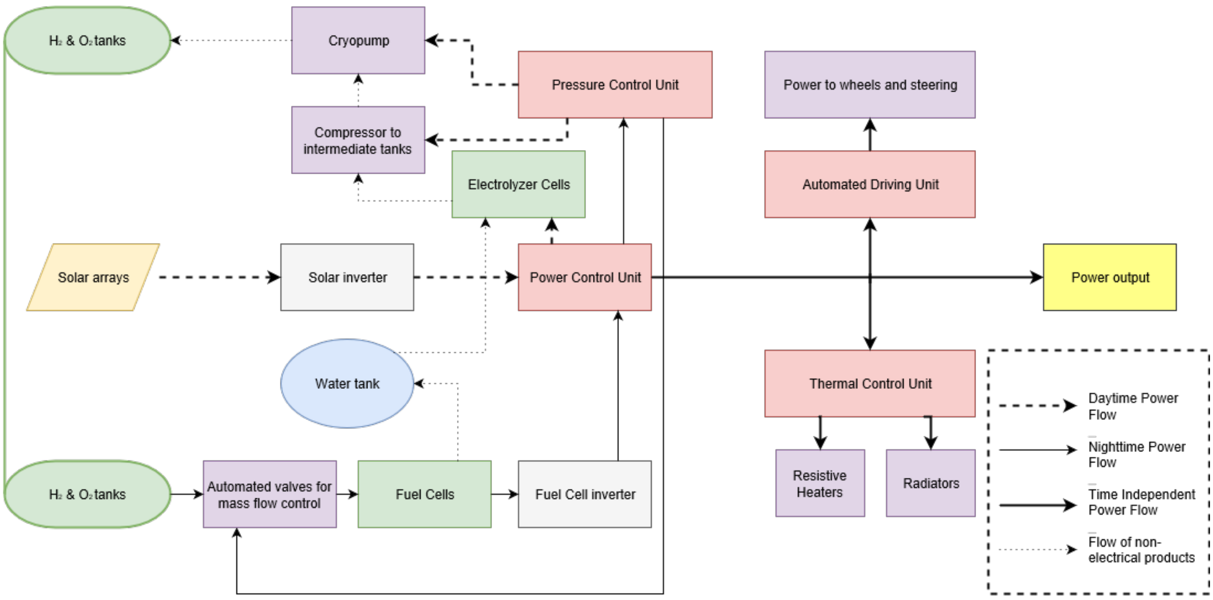


Figure 8.5: Electrical Block Diagram of one power module.

During the lunar day, hydrogen and oxygen will flow through pressure valves connected to the pressure control unit, to control the mass flow of the hydrogen and oxygen accurately. The hydrogen and oxygen will flow into fuel cell stacks that will produce electrical power and water. The water will be contained in a water tank and the electrical DC power will be converted into AC power with the fuel cell inverters. The power will flow from these inverters to the PCU. The Thermal Control Unit (TCU) is not always directly dependent on the load demand, therefore it is not coupled to the PCU. The TCU will ensure that the hydrogen and oxygen tanks will remain at correct temperatures at all time, that the water will not freeze and that all components will be within the right temperature boundaries. When cooling is required, an outer loop that will flow through radiators will cool the inner thermal loop with a heat exchanger. When heating is required, the radiators are folded within the structure and heaters in the heat exchangers will heat the inner loop. Furthermore, the Power Rangers are provided with an Automated Driving Unit (ADU) which enables the modules to automatically transport themselves when needed. When the power module is not driving the ADU will not consume power. The power output will provide the Power Rangers with the ability to connect to the habitat structure and to rovers and some other auxiliary systems directly.

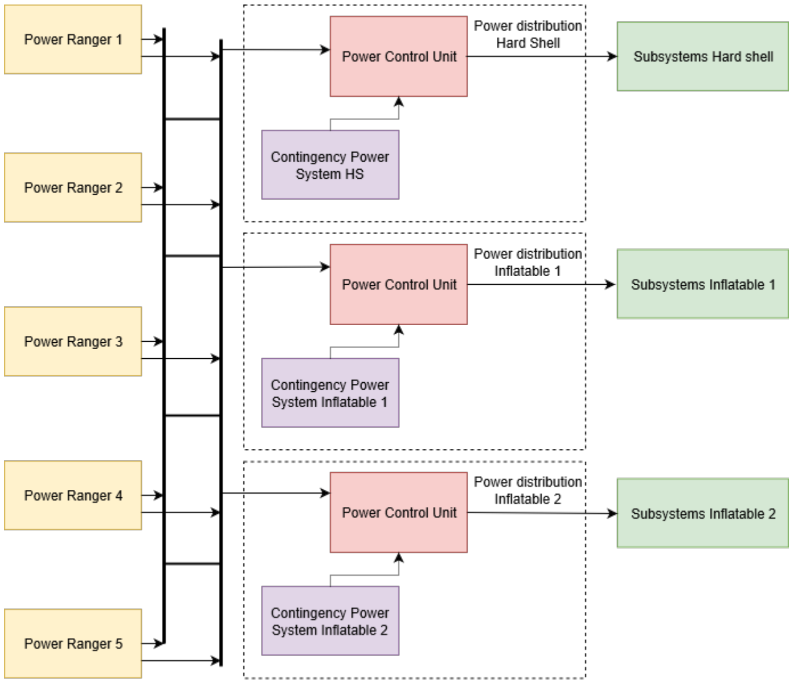


Figure 8.6: Electrical Block Diagram of the habitat.

The Electrical Block Diagram of the habitat is shown in Figure 8.6. The five Power Rangers are connected by cables to two busses in a dual power distribution bus system. The power distribution bus is designed to collect and distribute the power from the Power Rangers to the separate habitat modules. The benefits of having a dual power distribution bus is that power peaks that would occur in a single habitat module will result in equally distributed load demands for each power module. The dual power distribution bus has also an interconnection between the two busses between each power module, to ensure power supply when one of the two busses would have a failure. From the dual power distribution bus, each habitat module will gather its power through a Power Control Unit (PCU) which controls and distributes the incoming power to the subsystems onboard the habitat modules. Additional to the power supply through the power distribution bus, a contingency power system is installed in each habitat module. This will ensure that, in case of emergency, the habitat modules will have power for seven days for evacuation.

RAMS Characteristics and Sensitivity Analysis

The power system was designed with the philosophy that the reliability of the power system is of utmost importance. Without power, no systems will be able to operate. There will be no communication with Earth, no atmospheric control, no active thermal control and no active radiation protection. To increase the reliability, the modular design is chosen over the central design. Also multiple tanks will be present to store the oxygen and hydrogen. In case of a meteorite impact, the energy loss will be at most half of the energy storage of a Power Ranger. Since five Power Rangers will be present, this will only be 10% of the total energy storage and the lunar habitat will be able to operate with the power from four Power Rangers. While selecting elements and materials for the Power Rangers, available products and materials were selected to get a good impression of what is possible today. Therefore, the availability will be sufficient. The maintenance for the power system will be limited. The fuel cell stacks have an expected lifetime of 40,000 hours. This means that they will need to be replaced two or three times during the ten year operationality of the lunar habitat. The advantage of having the modular design is that a Power Ranger can be sent with ordinary resupply missions when extra power is requested with the expansion to a lunar village. The safety is assured by having the power system outside the habitat. Since the energy will be stored in large tanks of hydrogen and oxygen, installing these in the habitat would pose a risk. A very limited amount of energy will be stored within each module of the habitat to satisfy an emergency load demand of seven Earth days.

The power system design is dependent on the power consumption of every system. To perform a sensitivity analysis, the necessary total power which is used as inputs was increased with 1kW. The result was that a mass increase of 118kg per Power Ranger was observed. Therefore the total mass of the power system would be 12118kg which is 5.12% with respect to the original design.

9
Subsystem Summary

This chapter provides an overview of all designed modules: The Shell, The Nest and The Hive, and the Power Rangers. Firstly, the summary of the Power Rangers is split into two parts: the mass is presented in section 9.1 and the power consumption in section 9.2. section 9.3 gives the mass and composition of the hard-shell module, called The Shell. Furthermore, section 9.4 and section 9.5 display the inflatable module designs with their respective masses and compositions, as well as their wet masses.

The mass and power contributions are determined using a spreadsheet containing information about each subsystem. For that, all subsystems calculations are located on a separate sheet within the same file. The sheet facilitates the connection of dependent subsystems. This enables immediate updates of all subsystems connected to the updated system resulting in a faster optimisation process. All masses, power and dimensions of all subsystems are linked to the main spreadsheet. Finally, a mass budget of the entire habitat can be generated which is automatically updated.

9.1. The Power Rangers Factsheet - Mass Composition

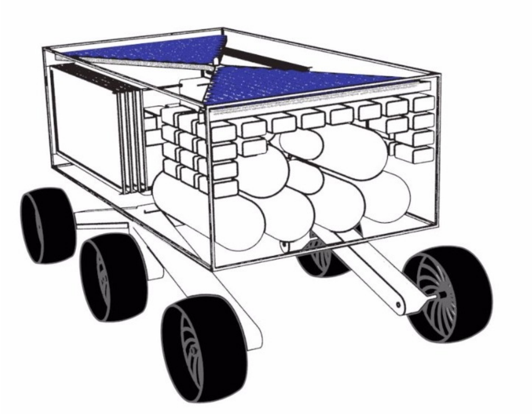


Figure 9.1: Sketch of one folded Power Ranger.

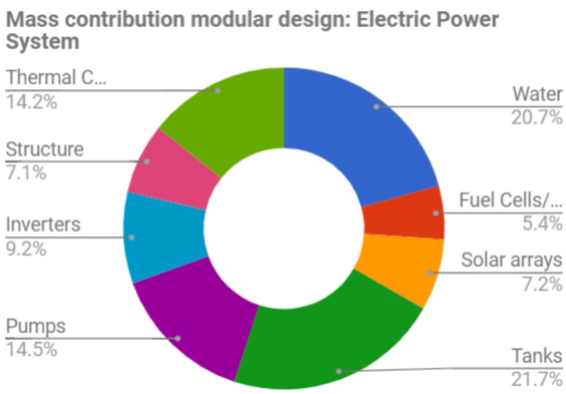


Figure 9.2: Mass composition of one modular power supplier: the Power Ranger.

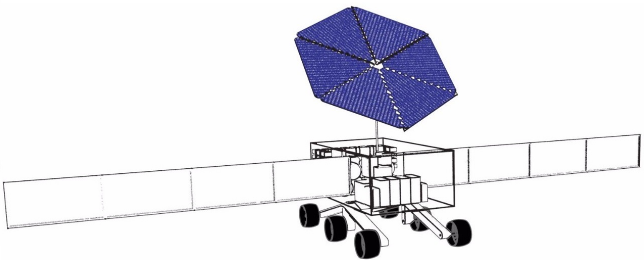
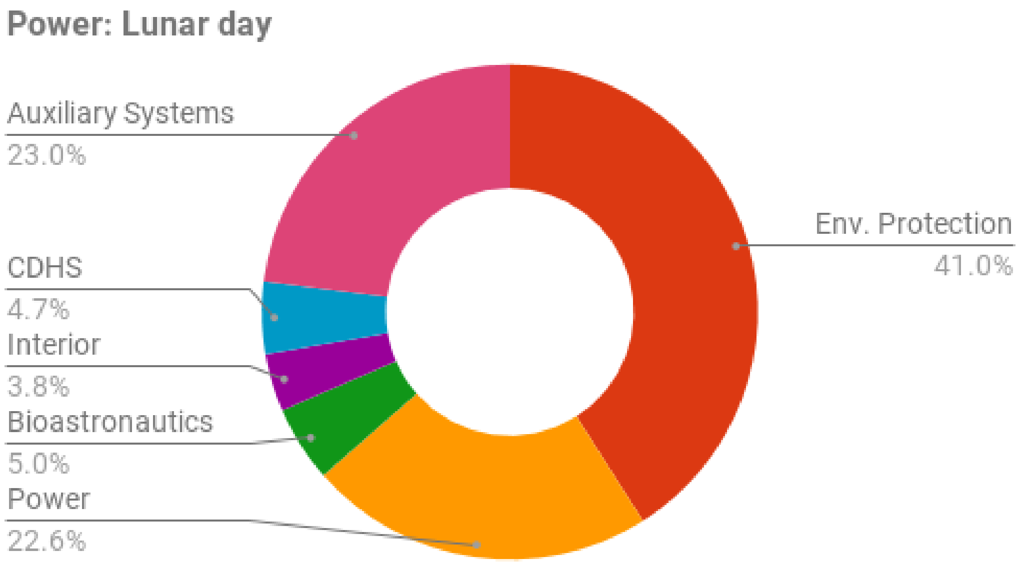


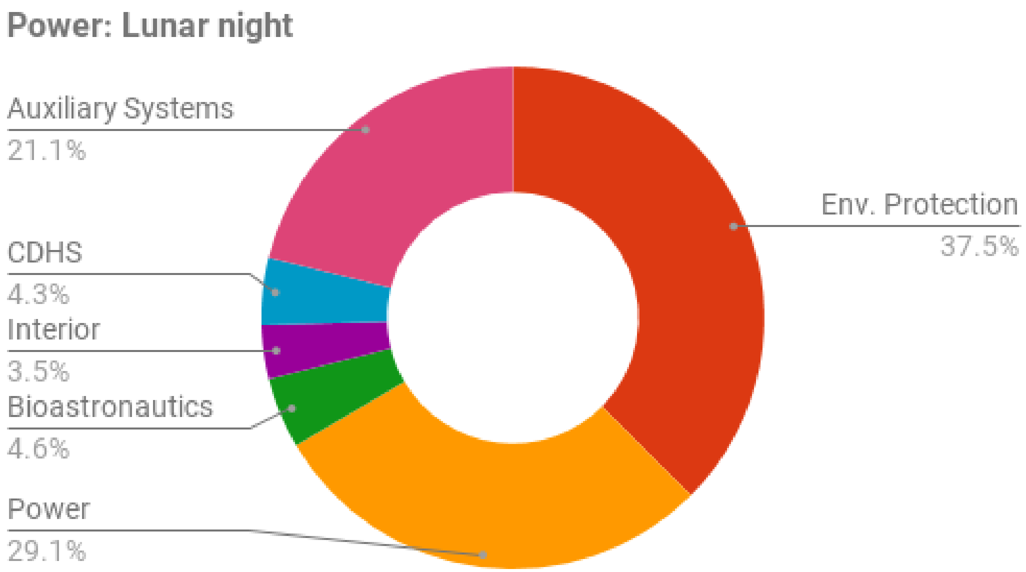
Figure 9.3: Sketch of one extended Power Ranger.

Table 9.1: Mass contributions of the power ranger's own subsystems.

Mass Contributions	**Mass [kg]**
Water	477.12
Fuel/Electrolyzer Cells	124.00
Solar arrays	166.03
Tanks	501.33
Pumps	333.4
Inverters	212
Structure	163.28
Thermal Control	328.38
Total mass for 1	**2305.54**

9.2. The Power Rangers Factsheet - Power





Inputs | Power	**Total required [kW]**	**Truck required [kW]**
Average power day	13.82	2.76
Average power night	15.06	3.01
Peak power day	67.62	13.52
Peak power night	90.09	18.02

Subsystem | Average Power	**Lunar day [kW]**	**Lunar night [kW]**
Environmental protection	4.358	4.341
Bioastronautics	0.5277	0.5277
Communication	0.5	0.5
Interior	0.4	0.4
Power	2.40	3.37
Auxiliary systems	2.45	2.45
Total $*1.3$	13.82	15.06

Subsystem | Peak Power	**Lunar day [kW]**	**Lunar night [kW]**
Environmental protection	4.608	4.591
Bioastronautics	30.493	30.493
Communication	2.7	2.7
Interior	0.6	0.6
Power	12.55	35.04
Auxiliary systems	16.67	16.67
Total	67.62	90.09

9.3. The Shell Factsheet - Mass Composition

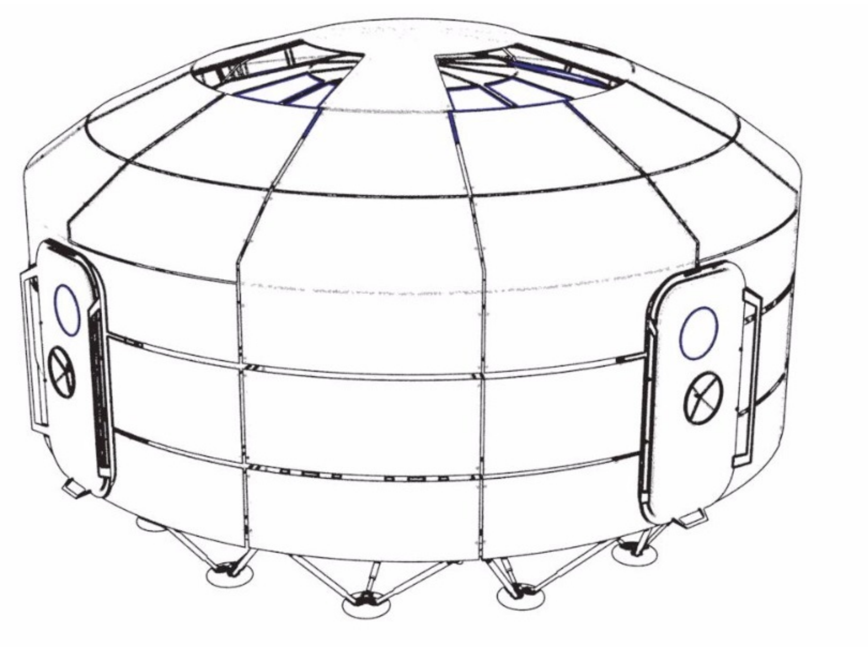


Figure 9.4: Sketch of the hard-shell module The Shell.

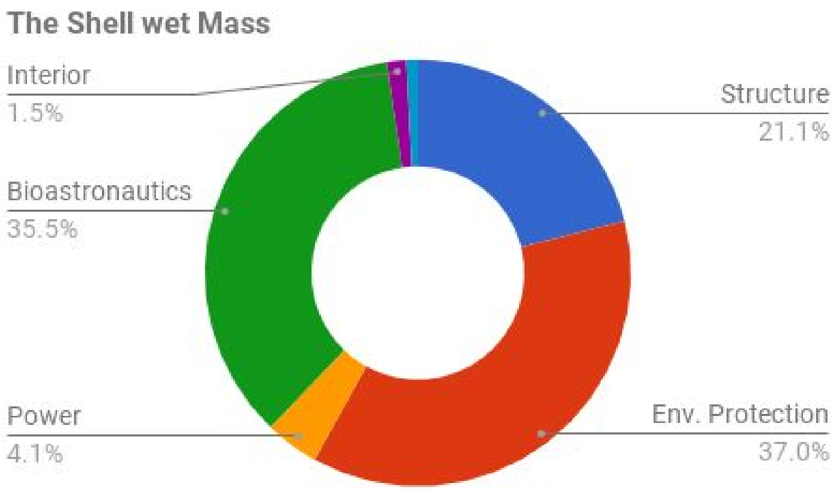


Figure 9.5: Mass composition of The Shell structure.

Environmental Protection	**System Part**	**Mass [kg]**	**System Part**	**Mass [kg]**
Active Thermal Control	Interface Heat exchanger	82.6	Tubes and Valves	50
	Pump modules	154	Radiators	149
	Cold plates	65.9	Tanks	17.6
Atmospheric control	Pumps, tanks, valves and fans	198.37	Oxygen generation	27.07
	Monitoring	38.95	Filters	67.98
Radiation Protection	Active	2579	Passive	None added

MMOD protection	**Layer**	**Material**	**Thickness [m]**	**Mass [kg]**
Whipple shield	Front Bumper	Al 6061-T6	0.11	249.35
	Upper stuffing	Nextel AF10	0.13	287.33
	Lower stuffing	Kevlar 29	0.08	95.78
Passive Thermal Control	MLI	Mylar, Dacron	4.40	380.7

Structure	**Layer**	**Material**	**Thickness [mm]**	**Mass [kg]**
Load Bearing	Pressure chamber	Al 2219-T87	6.4	2316.84
Flooring	Floor Panels	Al. Honeycomb	25	200

Other	**System Part**	**Mass [kg]**	**System Part**	**Mass [kg]**
Supporting items	Tubes and Feet	10.3 (18pcs)	Pins	6.44 (177pcs)

Bioastronautics	**System Part**	**Mass [kg]**	**System Part**	**Mass [kg]**
Water-management	Urine processor	159.4	Catalytic reactor	398.5
	Water Filter	239.1	Tanks	109.9
Astronaut supply	Water supply	615.4	Food, tools and medkits	1407.9
Airlock Design	Airlock	1262.2	Dust control	1.8
Communication	LLST incl. backup	61.4	Laptops and data bus	25.0
	Cables	20	Router	0.2
Interior	Furniture	152.71	Walls	21.93

9.4. The Nest Factsheet - Mass Composition

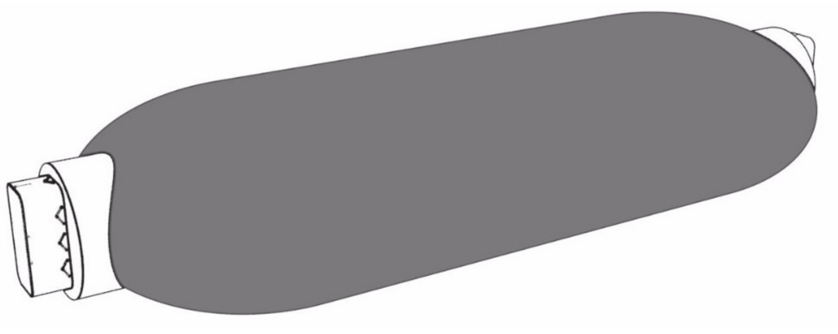


Figure 9.6: Sketch of the inflatable modules set-up: valid for the Nest and the Hive.

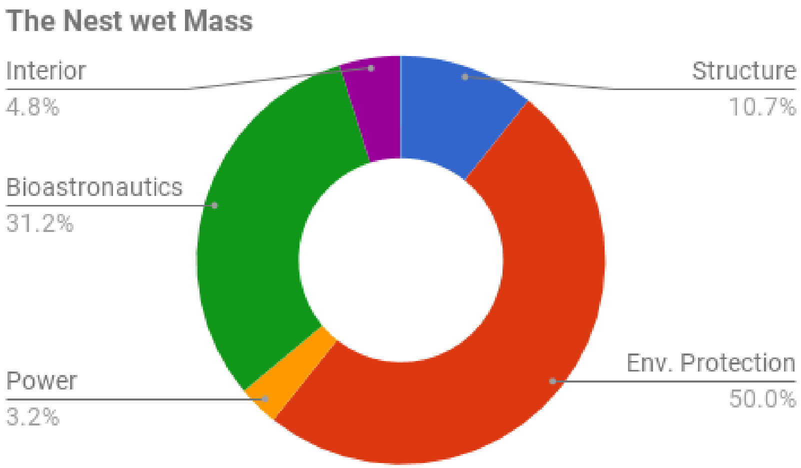


Figure 9.7: Net wet mass composition of the Nest. It consists of the MTB blanket featuring MMOD protection, thermal insulation, a load bearing layer and an airtight atmospheric layer.

Environmental Protection	**System Part**	**Mass [kg]**	**System Part**	**Mass [kg]**
Active Thermal Control	Interface Heat exchanger	82.6	Tubes and Valves	50
	Pump modules	154	Radiators	160.6
	Cold plates	57.6	Tanks	19.2
Atmospheric control	Pumps, tanks, valves and fans	346.01	Oxygen generation	54.79
	Monitoring	78.81	Filters	137.54
Radiation Protection	Active	-	Passive	-

MMOD protection	**Layer**	**Material**	**Thickness [mm]**	**Mass [kg]**
Multishock thermal blanket	Cover layer	Beta cloth	0.25	65.0
	Disrupter	Nextel AF10	1.98	694.98
	MLI	Mylar, Dacron	9.86	1,017.75
	Spacer	AC550	456.3	421.16
	Stopper	Kevlar 149	8.11	1,550.2
	Back cover	Mylar	0.07	12.38

Structure	**Layer**	**Material**	**Thickness [mm]**	**Mass [kg]**
Load Bearing	Restrainer	Kevlar 149	0.09	207.34
Airtight layer	Bladder	Elastomer FKM	1.0	468
Flooring	Floor Panels	Al. Honeycomb	25	375

Bioastronautics	**System Part**	**Mass [kg]**	**System Part**	**Mass [kg]**
Water-management	Urine processor	159.4	Catalytic reactor	398.5
	Water Filter	239.1	Tanks	193.4
Astronaut supply	Water supply	1436.0	Food, tools and medkit	-
Airlock Design	Airlock	631.1	Dust control	0.9
Communication	Cables	5	Router	0.2
Interior	Furniture	386.1	Walls	85.47

9.5. The Hive Factsheet - Mass Composition

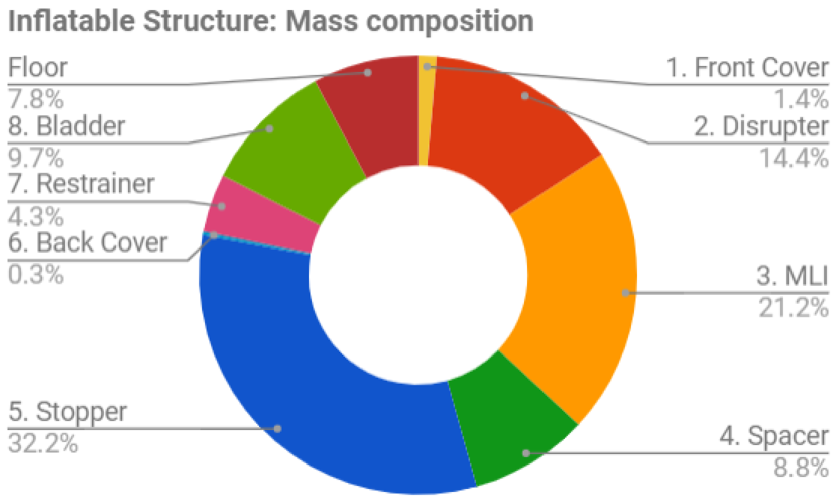


Figure 9.8: Composition of the complete inflatable structure: valid for the Nest and the Hive.

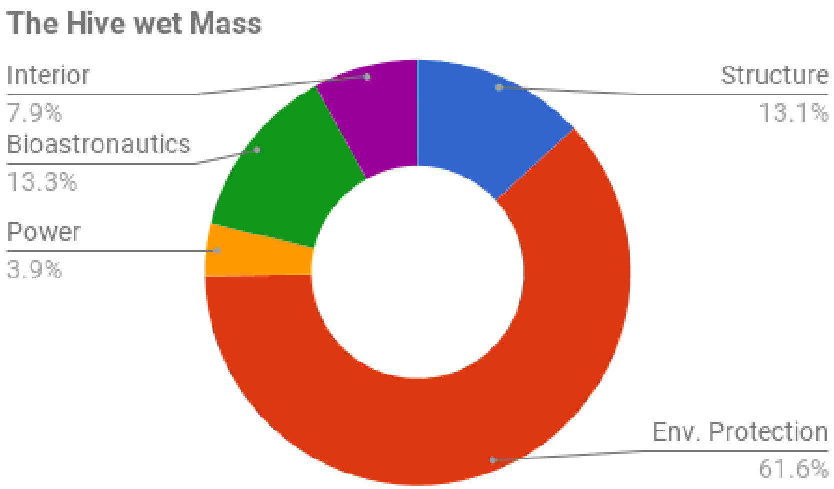


Figure 9.9: Net wet mass composition of the Hive. It consists of the MTB blanket featuring MMOD protection, thermal insulation, a load bearing layer and an airtight atmospheric layer.

Environmental Protection	System Part	Mass [kg]	System Part	Mass [kg]
Active Thermal Control	Interface Heat exchanger	82.6	Tubes and Valves	50
	Pump modules	154	Radiators	160.6
	Cold plates	77.6	Tanks	19.2
Atmospheric control	Pumps, tanks, valves and fans	346.01	Oxygen generation	54.79
	Monitoring	78.81	Filters	137.54
Radiation Protection	Active	-	Passive	-

MMOD protection	Layer	Material	Thickness [mm]	Mass [kg]
Multishock thermal blanket	Cover layer	Beta cloth	0.25	65.0
	Disrupter	Nextel AF10	1.98	694.98
	MLI	Mylar, Dacron	9.86	1,017.75
	Spacer	AC550	456.3	421.16
	Stopper	Kevlar 149	8.11	1,550.2
	Back cover	Mylar	0.07	12.38

Structure	Layer	Material	Thickness [mm]	Mass [kg]
Load Bearing	Restrainer	Kevlar 149	0.09	207.34
Airtight layer	Bladder	Elastomer FKM	1.0	468
Flooring	Floor Panels	Al. Honeycomb	25	375

Bioastronautics	System Part	Mass [kg]	System Part	Mass [kg]
Water-management	Urine processor	-	Catalytic reactor	-
	Water Filter	-	Tanks	75.3
Astronaut supply	Water supply	357	Food, tools and medkit	-
Airlock Design	Airlock	631.1	Dust control	0.9
Communication	Cables	5	Router	0.2
Interior	Furniture	596	Walls	36.95

9.6. Resource Budget

Finally, a total breakdown of the subsystems for the entire habitat is presented in Figure 9.10. This mass does not include the Power Rangers, as they are auxiliary units and not part of the structure. Furthermore, in order to judge the progress made since the delivery of the midterm report, the mass budget of the midterm is considered in Figure 9.10.

Figure 9.10: Percentage Fraction of the subsystem masses obtained in the final and midterm report. For the final report 100% = 29.8t; for the midterm report 100% = 38.8t [2].

Subsystem | Mass	Final [%]	Midterm [%]
Env. protection	47.86	46
Structure	15.54	12
Bioastronautics	28.14	12
CDHS	0.39	9
Interior	4.29	0
Power	3.77	21

Figure 9.11: Percentage Fraction of the subsystem peak power consumption (lunar night) of the final and midterm report. Final report 100% = 11.58kW (excl auxiliary units); Midterm report 100% = 40kW [2].

Subsystem | Power	Final [%]	Midterm [%]
Env. protection	37.48	40
Structure	0	1
Bioastronautics	4.56	22
CDHS	4.32	19
Interior	3.45	0
Power	29.09	18
Aux. systems	21.15	-

It becomes evident that the mass fractions of all subsystems are substantially different and therefore, are impossible to compare. The same situation can be found for the power budget. The values for the final habitat and the ones from the midterm review can be found n Figure 9.11. The explanation for this situation is simple; in the midterm review, the mass fractions are determined using reference missions [9]. However, there has never been a mission comparable to LEAP. The only, slightly related missions are the ISS mission, which features the long-term hosting of the ISS, and the Apollo missions, as they landed humans on the Moon. Thus, the fractions used are estimations only, which turn out to be unsuitable for this mission. This conclusion is certainly not a failure of the mission, however, the uncertainties and contingencies estimated need to be evaluated and new ones established. This is a challenging task, due to the lack of expertise in this field and the stage of the design. Thus, the uncertainties of the mass and power fractions are estimated with respect to the sensitivity findings of the performed calculations. Thus, the subsystem calculations are subjected to an increase in their respective requirements, such as higher thermal flux for thermal control or higher power needs for the Power Rangers. As the subsystems are already iterated and optimised, the change in design is estimated to be low. Table 9.2 shows the newly estimated uncertainty values.

Table 9.2: Estimated uncertainties with respect to power and mass.

Uncertainties	Mass		Power	
	[%]	[t]	[%]	[kW]
Env. protection	20	2.85	15	0.65
Structure	15	0.70	0	0
Bioastronautics	10	0.84	10	0.05
Communication	10	0.01	20	0.1
Interior	10	0.13	10	0.04
Power	10	0.11	10	0.34
Average	12.5	2.41	10.9	1.00

For the environmental protection, however, it needs to be said that the uncertainty of thermal control, MMOD protection and radiation protection differ significantly while only the average is presented here. For the uncertainty in mass, the thermal protection estimate lays at 15%, the radiation protection at 30% and the meteorite protection at 20%. Regarding the power uncertainty, especially the active radiation protection is striking as the uncertainty is high. This can be explained due to the fact that the active radiation protection is still at a low TRL and the assumptions made may be invalid. Furthermore, the thermal control features an uncertainty of 20% regarding power consumption while the MMOD protection does not require power at all.

The uncertainties given provide a brief overview of how sensitive the subsystems are to changes and indicate their design stage. Thus, it can be seen which ones require more attention in the future to ensure mission success. Generally, all systems need to be further optimised and iterated in order to decrease the uncertainty. Further, some subsystems need more research to confirm their applicability and the calculations performed.

10
Transportation and Lander Configuration

In order for the habitat to be deployed, it first needs to be transported to the Moon. Although a lot of experience has been gained in launching spacecraft into near-Earth orbits, less experience has been gained in the past years regarding beyond Earth exploration and landing on other celestial bodies. In section 10.1, the launch vehicles that are capable of, or are being developed for delivering payloads to lunar orbit or Trans Lunar Injection (TLI) are described. Once the launch vehicle is treated, the lander that delivers the payload on the lunar surface is described and sized in section 10.2. The chapter is concluded in section 10.3, with a description of the strategy and the spacecraft used for transporting the astronauts to The Moon.

10.1. Launch Vehicles

In order to assemble the habitat and provide sufficient supplies at all times, a launch plan has to be set up. Table 10.1 provides an overview of all launch vehicles to be considered for the LEAP successful mission. For each phase of the mission, a suitable launch vehicle has to be determined. For this, the cost and useful payload are the most important criteria to be considered. For these missions, three different launch vehicles are considered. This selection is based on the ability of a launch vehicle to deploy a payload into TLI. All of them are currently under development for this purpose and are expected to be ready by the time the first launch is scheduled. Older launch vehicles which have deployed payloads to the Moon are not considered, due to their lower effectiveness and thus lower sustainability. Furthermore, current launch vehicles have reduced the cost per launch significantly and are therefore preferred. Considering the cost again, the Angara 5V (A5V) and the Falcon Heavy are preferred with respect to the Space Launch System (SLS), due to their lower cost. Additionally, the Falcon Heavy is a more sustainable option due to its recyclable components. Despite the lower cost, these launch vehicles are not the only vehicles that are going to be used, simply due to the limitation of their useful payload mass, as can be noted in Table 10.1[1][2].

Table 10.1: Launch vehicle properties and the dimensions of the payload fairing [46].

Launch Vehicle	**Launch cost [$]**	**Useful payload [tonnes]**	**Height Fairing [m]**	**Diameter Fairing [m]**	**Height Fairing (max ∅) [m]**
SLS	>500M	17.8	19.1	7.5	10.1
Falcon Heavy	135M	7.1	12.5	5.2	6.6
Angara A5V	100M	4.3	12.8	4.3	3.8

The price and the mass the launch vehicles are not the only factors that play a role in the selection; the size of the fairing is also an element which has to be taken into account, in Table 10.1 all the different parameters of the fairing of the launch vehicles can be found.
From Table 10.1 it becomes clear that the fairing shapes of the three launch modules differ significantly, the Space Launch Systems distinguishes itself in having a very high fairing while the Angara possesses a relatively small diameter this results in a very small useful volume, this is confirmed by Figure 10.4, where the lander already makes up of large part of the payload fairing.
The different launch vehicles have different launch pads due to their origin. The Space Launch System and the Falcon Heavy are American spacecraft while the A5V belongs to the Russian Angara family and are therefore launched from their country of origin. For the Angara 5V, the new Vostochny platform has

[1]http://www.spacex.com/news/2013/04/12/fairing [Cited On: 23-06-2017]
[2]www.russianspaceweb.com/angara5v.html [Cited On: 23-06-2017]

been built in the past years specifically for this new part of the Angara family. It is situated near the Uglegorsk in the Amur region, in south-east Russia on the border with China. The first launch from this new site has already been carried out by a Soyuz rocket, while the first Angara launch is planned for 2021 [3]. On the other hand, the American based launch vehicles will be launched from American soil. SpaceX currently uses four different launch sites: Cape Canaveral Air Force Station space launch complex 40, Vandenberg Air Force Base Space Launch Complex 4, Kennedy Space Centre Launch Complex 39A, and SpaceX South Texas Launch Site [4]. According to SpaceX, the Falcon Heavy is able to take off from both the Florida Launch and the Cape Canaveral sites, so these will be used according to their availability[5]. Finally, the Space Launch System will be launched from the Kennedy Space Centre as the test flight will be executed at this site, enlarging the experience in the launch site resulting in a higher probability of success [6].

10.2. Lander Configuration

The design of the mission and the habitat depend heavily on the lander configuration. The way the lander deploys the payload on the lunar surface dictates actions that have to be undertaken in order for the payload to become useful. The goal is thus to reduce as much as possible the amount, and complexity of the supporting mechanisms needed on the Moon to extract the payload. Keeping this in mind, for the landing of the habitat modules a dual thrust axis lander is used [47].

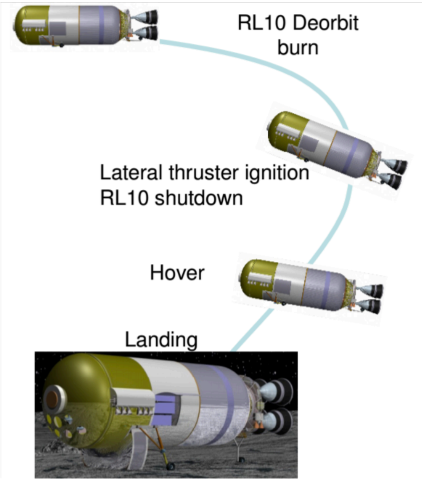


Figure 10.1: Landing sequence [47].

The lander configuration can be seen in Figure 10.1. the method that is going to be used entails two different stages. The first stage decelerates the payload from the high trans-lunar injection speed (2952m/s) to the lower speed required for the final precision landing (3m/s). Once the payload is slowed down the second stage is engaged which performers the final controlled descent of the payload. This method is referred with respect to the traditional single stage lander because the second slow speed stage enables more freedom in the landing orientation of the payload. This is because the payload can be tilted changing the orientation of the payload and thus the large RL10 engine is not under the payload when on the lunar surface, facilitating the accessibility of the habitat module, which is the goal in this design.

Lander First Stage

The first stage entails one RL10 engine to perform the deceleration. There are three different types of RL10 engines, in order to select the optimal engine for the first stage, calculations are performed for all three in order to detect the differences in the design and select the optimal solution. The specification of the three different thrusters can be found in Table 10.2.

Using specifications of the three engines, the first stage of the lander is sized in order to be able to select the optimal engine for the lander. This is done by using the ideal rocket equation of Tsiolkovsy which gives the mass relationship between the start and end of the deceleration, as is described in Equation 10.1. This equation also largely dictates what the useful payload is which are depicted Table 10.1. This is because the first stage is the heaviest part of the lander, because of the large deceleration it needs to perform. The values reported also consider the weight the second stage entails which is explained later in this section.

[3]https://spaceflightnow.com/2016/04/28/first-launch-from-russias-new-cosmodrome-declared-a-success/ [Cited On: 20-06-2017]
[4]http://www.spacex.com/about/capabilities [Cited On: 20-06-2017]
[5]http://www.spacex.com/missions [Cited On: 20-06-2017]
[6]https://www.nasa.gov/sites/default/files/atoms/files/sls_october_2015_fact_sheet.pdf [Cited On: 20-06-2017]
[7]http://www.rocket.com/rl10-engine [Cited On: 12-06-2017]

Table 10.2: RL10 engine specifications[7].

Characteristic	RL10A-4-2	RL10B-2	RL10C-1
Thrust [N]	99200	110090	101820
Weight [kg]	167.8	301.2	190.5
Fuel	Liquid hydrogen	Liquid hydrogen	Liquid hydrogen
Oxidiser	Liquid oxygen	Liquid oxygen	Liquid oxygen
Mixture Ratio	5.5:1	5.88:1	5.5:1
Specific Impulse [s]	451.0	465.5	449.7
Length [m]	2.296	2.197 (stowed)	2.184
	-	4.153 (deployed)	-
Nozzle Diameter [m]	1.168	2.146	1.448

$$\Delta V = I_{sp} g_0 ln \frac{m_0}{m_f} \tag{10.1}$$

One needs to keep in mind that the precise orbital location and time span are not considered in this equation, it only considers the amount of delta V the engines need to deliver based on the engine efficiency. Using the specifications of the engines depicted in Table 10.2, the required velocity change, and the masses the two selected launch vehicles are able to deploy into TLI the sum of the required fuel and oxidizer per RL10 engine is found. The fuel and oxidizer mass is then multiplied by a safety factor of 1.1, which is the classical safety factor for fuel amounts in space missions. Once the combined mass of the fuel and the oxidizer are determined, the mixture ratio is used such that the single masses can be determined in order to be able to determine the respective volumes. The densities that are used to determine the volumes are 70.8 kg/m^3 for the liquid hydrogen and 1141 kg/m^3 for liquid oxygen. In Table 10.3 a summary of the fuel fractions and the respective volumes can be found.

Table 10.3: Mass and volume fractions for the high speed deceleration.

	Falcon Heavy			Space Launch System			Angara 5V		
RL10 engine type	A-4-2	B-2	C-1	A-4-2	B-2	C-1	A-4-2	B-2	C-1
Total mass [tonnes]	8.56	8.37	8.58	21.04	20.57	21.08	5.3	5.2	5.4
Fuel mass [tonnes]	1.32	1.22	1.32	3.24	2.99	3.24	0.8	0.8	0.8
Oxidiser mass [tonnes]	7.25	7.16	7.26	17.80	17.58	17.84	4.5	4.5	4.5
Total volume [m^3]	24.96	23.47	25.01	61.31	57.64	61.44	15.6	14.7	15.6
Fuel volume [m^3]	18.61	17.19	16.65	45.71	42.23	45.81	11.6	10.7	11.6
Oxidiser volume [m^3]	6.35	6.27	6.36	15.60	15.41	15.63	4.0	3.9	4.0

Once the volume of the required fuel and oxidiser are determined. it needs to be determined how much volume the entire lander occupies in the payload fairing of the launch module, in order to determine the available volume for the useful payload. The upper part of the payload fairing has a curved shape and therefore it is difficult to use a section of the launch module. In order to prevent this curved shape to dictate the configuration of the useful payload, the lander is designed to fit in the curved section. This can be done because the fuel tanks can be designed such that they make efficient use of this otherwise difficult to use space. Therefore the engine is located in the nose of the launch module and the fuel tanks then follow the inner shape of the fairing. A program is made to optimise the shape of the fuel tanks, the result for the Falcon Heavy can be found in Figure 10.2 while for the SLS can be found in Figure 10.3 and finally in Figure 10.4 for the Angara launch vehicle.

Due to the low useful payload mass of only 4.3 tonnes (Table 10.1 and the small available payload volume in the fairing (Figure 10.4, the A5V is no longer considered as a viable option for the launch of the habitat modules. The Angara is still considered for other parts of the missions because of its low cost, which is favourable for missions with a smaller payload such as resupply or scouting mission.

The program used for the sizing of the first stage of the lander first performs the steps explained above to determine the volume of the fuel and the oxidizer. Once the volume is determined the program

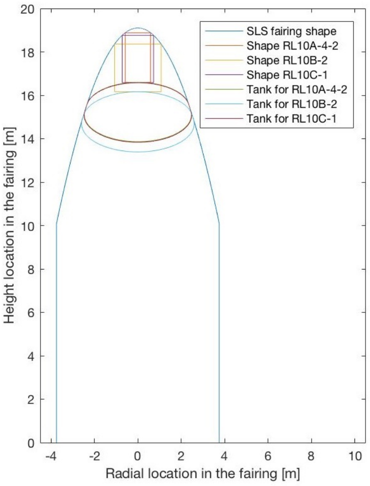


Figure 10.2: Fitting of the lander into the Falcon Heavy.

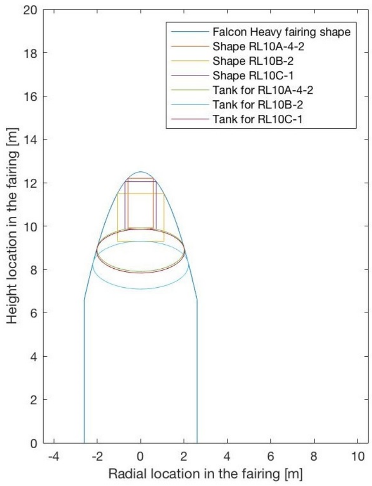


Figure 10.3: Fitting of the lander into the SLS.

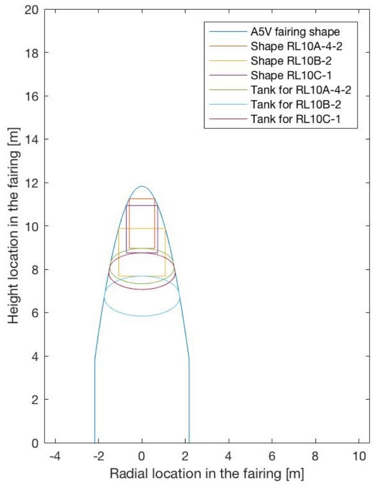


Figure 10.4: Fitting of the lander into the A5V.

inserts the engines into the nose of the fairing which in Figure 10.2, Figure 10.3, and Figure 10.4 are depicted by the squares, such that the engine does not point towards the payload. Afterwards, it starts calculation the shape tanks based on the required volume. The shape of the fuel tank is made up out of two different shapes, namely an ellipsoidal cap and the section of a paraboloid. The ellipsoids are used to close off the fuel tanks, this is preferred to a simple sphere as this is a more efficient use of the available diameter and is still possible to be used in pressurised fuel tanks. The paraboloid, on the other hand, is used because of the parabolic shape of the fairing. The program aims at maximising the available space for the payload, therefore making the lander as short as possible while respecting the shapes required for a pressurised tank, this entails respecting the tangency between the two ellipsoidal caps and the paraboloid section, preventing stress concentrations. The final shapes of the tanks for the different engines and launch modules can be found in Figure 10.2, Figure 10.3 and Figure 10.4.

Figure 10.5: First stage lander.

One must consider that the lander in this case has an unconventional orientation and this will only be used if the volume of the useful payload is a restriction for the payload bus of the launch vehicle. Otherwise, it will be placed at the bottom of the payload bus. In Figure 10.5 the first stage of the lander can be seen, here the RL10 engine is situated in the centre, while the oxidizer and fuel tanks surround the engine. When one looks only at the size of the lander there is no big difference in the sizes of the different landers and therefore no lander is preferred, while if one looks at the weights here also is a small difference, but as this is a more strict constraint the lightest option is selected and thus the RL10B-2 is used for the landers.

In order to perform the structural analysis of the modules, the loads during the descent have to be determined. This is done by another program, the results of this program can be found in Figure 10.6. The program calculates the incremental change in mass due to the consumed fuel, the acceleration due to the thrust of the engine and then the resulting velocity change. The figures depict this, one can note that as the spacecraft become lighter the acceleration it experiences becomes higher because the thrust remains constant. It can be noted that the amount of g-force the payload experiences is very low and thus can be omitted in the structural analysis as it is minor with respect to the other loads that the payload endures. Figure 10.7 represents the same values, in this case, it is a payload of the Falcon Heavy, also here the g-force is relatively low with respect to the launch loads.

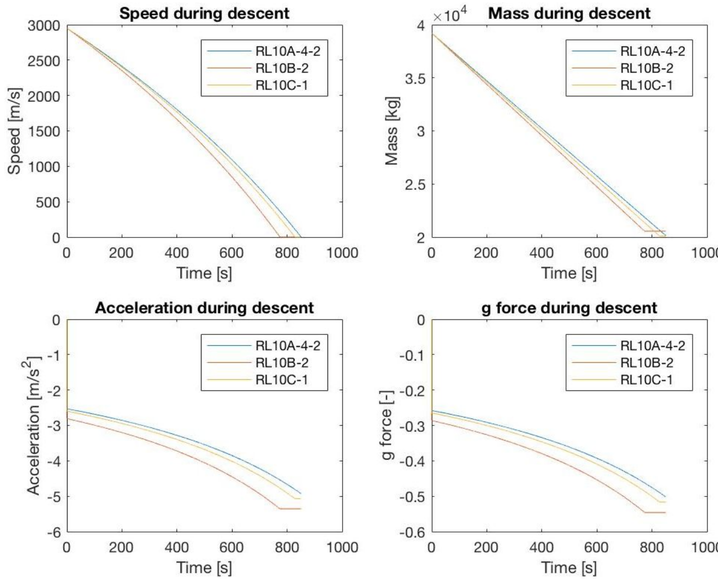


Figure 10.6: Loads during the deceleration of the SLS payload.

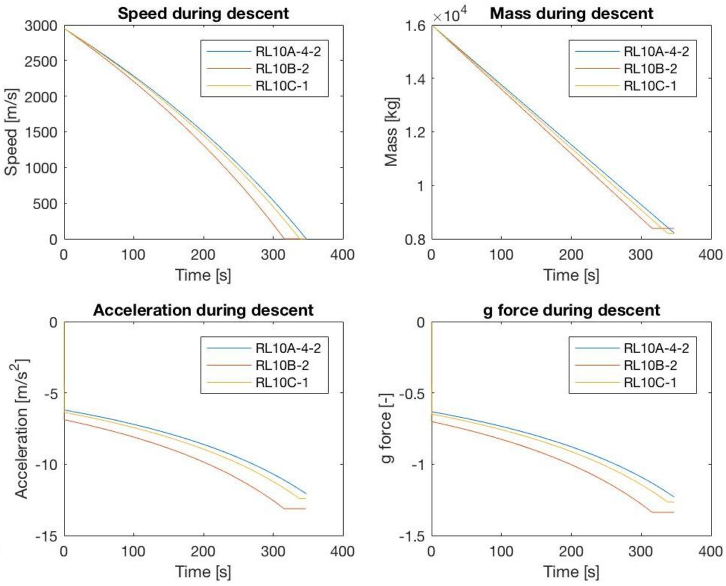


Figure 10.7: Loads during the deceleration of the FH payload.

The two payloads due to their different mass will experience different trajectories during the deceleration as they are subjected to the same thrust force. The differences in this orbit can be seen in Figure 10.8. In this figure the Moon can be noted and the two different trajectories. These trajectories depict the deceleration from 2.9km/s to the lower speed of 3m/s. It can be noted that the SLS payload needs to start the deceleration earlier and at a higher altitude with respect to the FH payload, due to its higher mass. In Figure 10.9 the image is enlarged in order to enhance the difference.

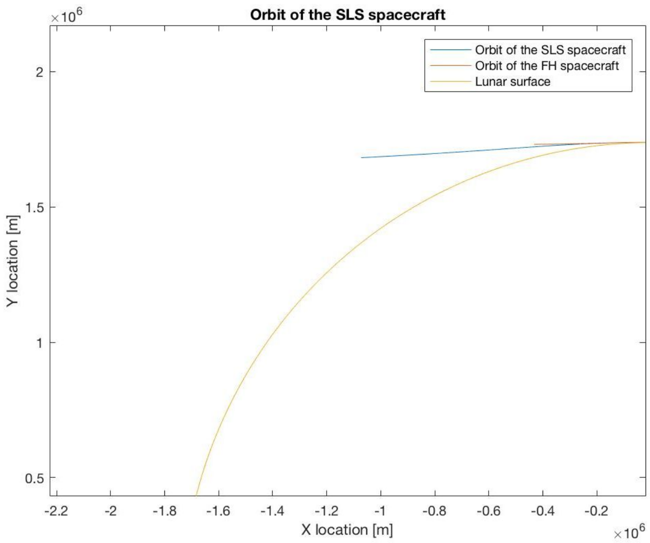


Figure 10.8: Difference in the orbits of the two payloads.

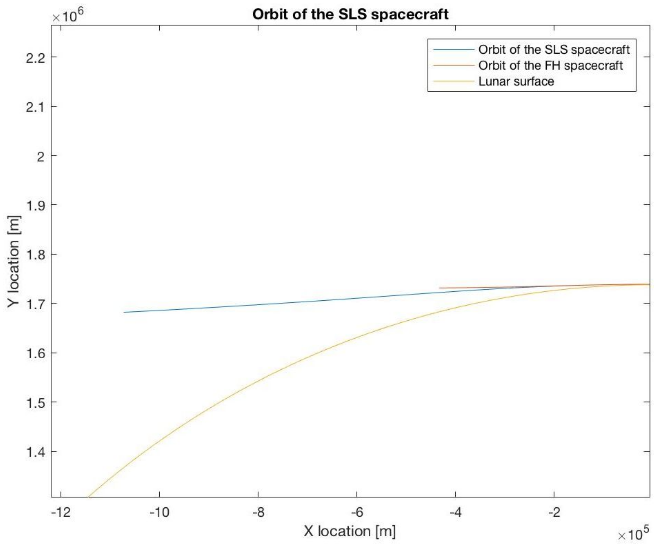


Figure 10.9: Closeup of the difference in the orbits of the two payloads.

Lander Second Stage

Now that the payload is slowed down to a speed of 3.048 m/s there are two options that can be undertaken either the RL10 engine is discarded together with its fuel tanks during descent or it is kept attached to the payload ad is landed with it. The first option enables more freedom in the landing orientation of the payload while the second option enables the potential for cannibalisation of the RL10 engine and its fuel tanks, this is more difficult with the first option as the engine and tanks crash on the lunar surface and thus are difficult to recover. In both cases, small engines are used to decelerate the payload to the ground.

In this case, the SuperDraco is used, this is because there is already research in using these thrusters for controlled flight. These engines are capable of producing 71 kN at maximum thrust, but they are also able to be throttled down in a range from 20 to 100 percent in order to perform a controlled descent, more properties of the SuperDraco engine can be found in Table 10.4, while a model of the SuperDraco can be seen in Figure 10.10[8].

Figure 10.10: SuperDraco engine.

In order to have full control four engines are used, but in order to have redundancy the engines are placed in pairs, resulting in a total of eight engines[9]. In this case, the Tsiolokvsky rocket equation can not be used as more forces start playing a role with respect to the first stage, mainly the gravitational pull of the Moon. Therefore a different method is used to size this stage.

For this model, a descent is assumed which starts at a speed of approximately three meters per second and an altitude of one kilometre, based on the requirements for the precision landing. This speed is then left free to accelerate due to the lunar gravity until the last moment the thrusters need to decelerate the module in order for the speed to be zero on the surface.
During the thrusting phase the throttle is reduced to 50% in order for it to be controlled and extra throttle is available if deemed needed. The SuperDraco has a mass flow of 31 kg/s at full throttle, this is then reduced proportionally to the throttle and the amount of engines in order to obtain the necessary fuel flow for the throttle setting.

The program determines the altitude at which it has to start deceleration and the amount of time it requires. The profile of the last descent including the loads the habitat experiences can be found in figure Figure 10.11 and Figure 10.12. In the figures, the speed, the altitude, and the amount of g-force the habitat experiences are depicted. Furthermore, the change in mass as fuel is consumed can be noted. These images represent the last descent which includes the high-velocity stage.

Table 10.4: Properties of the SuperDraco engine.

Thrust [kN]	71
Specific impuls [s]	235
Density monomethylhydrazine [kg/m^3]	880
Density nitrogen tetroxyde [kg/m^3]	1440
Mass flow at full throttle [kg/s]	31
Mixture ratio [-]	1.08

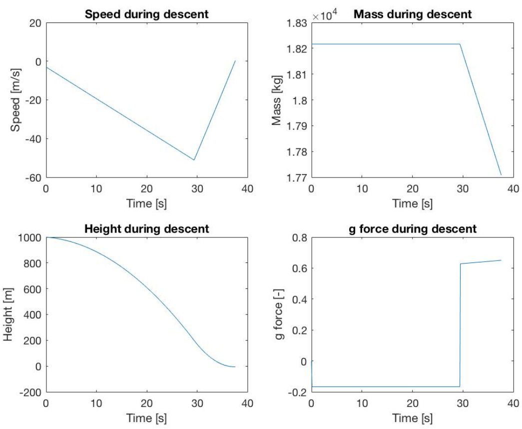


Figure 10.11: Loads during the last descent of the SLS payload.

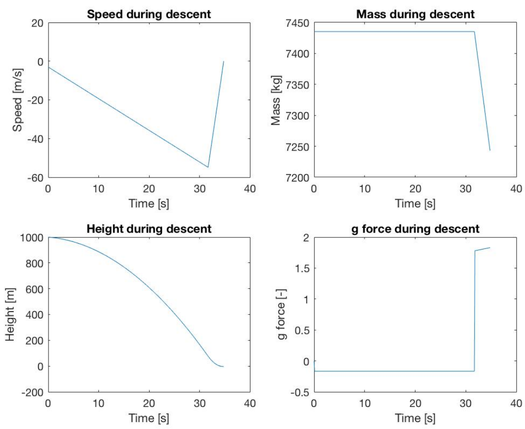


Figure 10.12: Loads during the last descent of the FH payload.

The graphs for when the high-velocity stage is removed are similar to the ones depicted in Figure 10.11

[8]http://www.spacex.com/press/2014/05/27/spacex-completes-qualification-testing-superdraco-thruster [Cited On: 12-06-2017]
[9]http://www.spacex.com/news/2014/05/30/dragon-v2-spacexs-next-generation-manned-spacecraft [Cited On: 19-06-2017]

and Figure 10.12, the change in mass is different and thus also the time at which the SuperDraco engines ignite and the relative burn time. This phenomenon can already be noted in the difference between the Space Launch System and the Falcon Heavy and are therefore not depicted in this report. Instead in Table 10.5 all the properties of the last descent of the different module can be found. When calculating the fuel fractions once again a safety factor of 1.1 is taken as was done before. The most important plot in the figures is described by the amount of g-force the habitat experiences during the last deceleration as this has to be accounted for during the design of the structure. It can be noted that for both cases the loads are significantly lower with respect to the launch loads the habitat has to endure and therefore are already accounted for in the design for those loads.

Table 10.5: Properties of the last descent of the habitat modules for the different configurations.

	Space Launch System		**Falcon Heavy**	
High speed stage	Attached	Discarded	Attached	Discarded
Maximal descent velocity [m/s]	-50.95	-50.95	-54.88	-54.88
Burn time [s]	8.3	8.3	3.2	3.1
Amount of g force [-]	0.63	0.64	1.78	1.86
Mass of fuel and oxidiser [kg]	572.88	566.06	218.24	211.42
Useful payload [tonnes]	17.8	17.8	7.1	7.1

In Table 10.5 one can note that keeping the first stage of the lander attached to the lander has little influence on the amount of final useful payload. Therefore in a big part of the landed payload, the high-speed stage will be kept attached to the payload such that the materials can be used to execute eventual repairs or construction. The only case where the first stage is separated is when the orientation of the module is important, this is for example the case with the hard-shell module which will land and the desired spot and not is moved.

10.3. Manned Missions

The manned mission requires a different approach with respect to the payload missions because the astronauts delivered on the Moon also need to be able to return to Earth. The best way to do this is to reuse the capsule that brought them to the Moon. In this way, no extra capsules need to be landed on the lunar surface increasing the sustainability of the mission as more objects are re-utilised. Furthermore, the safety is increased as the capsule is ready for the astronauts to leave in case of emergency, with respect to the option where a different extraction capsule has to be sent to the habitat.

There are two capsules currently under development for sending astronauts beyond LEO, one is the Boeing Orion Multi-Purpose Crew Vehicle, while the other one is SpaceX' Dragon V2 capsule. While both modules are able to land a crew on the Moon, the Dragon V2 capsules is also able to lift off from the Moon as it has Incorporated thrusters in the crew module, which enables it to perform the last descent, but also can be used for lift off[10]. The thrusters used in this case are the same as the ones used for the last descent phase as described in section 10.2. Due to this capability, the Dragon V2 capsule is a more suitable option due to the reasons described at the beginning of this section.

The Dragon capsule is capable of delivering up to seven astronauts to The ISS and to The Moon according to SpaceX, which is more than the required four astronauts in this mission. It has eight SuperDraco engines incorporated into the module which enables it to go from the launch pad into orbit. These engines are the same as used during the landing as described in section 10.2, therefore these engines are used for the descent, using the same method described in the previous section, but also for the ascent of the crew module[11]. It is not clear whether the Dragon capsule is able at bringing the module back to Earth or solely into orbit. In case the latter case is true the Dragon capsule will rejoin with a module which it has left there during arrival, which propels it back to Earth.

[10]http://www.spacex.com/news/2014/05/30/dragon-v2-spacexs-next-generation-manned-spacecraft [Cited On: 20-06-2017]
[11]https://www.inverse.com/article/28534-everything-to-know-about-dragon-2 [Cited On: 20-06-2017]

11
Functional Analysis and Logistical Approach

In order to produce a feasible design for the lunar habitat, it is necessary to identify all operations related to the mission and to keep these in mind during the design process. For this reason, all operations have been categorised in a Functional Breakdown Structure (FBS) in section 11.1. Additionally, the flow of all activities is described in section 11.2, with special attention paid to the logistic challenges of the mission. Finally, the launch logistics are described in section 11.3.

11.1. Functional Breakdown Structure

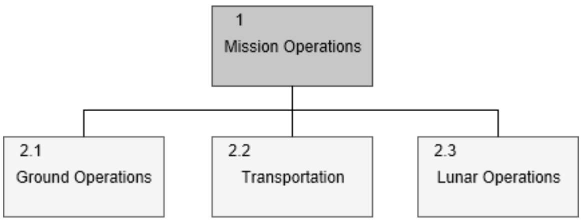


Figure 11.1: Top Level Functional Breakdown Structure.

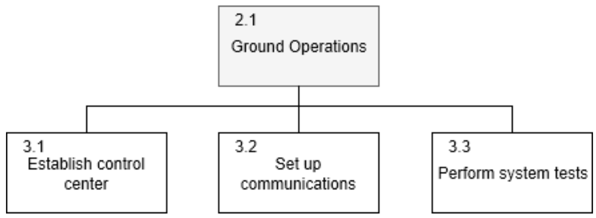


Figure 11.2: Functional Breakdown Structure of Ground Operations.

The FBS for the system at hand can be broken down into four chronological operation types: Design Operations, Manufacturing Operations, Mission Operations and Post-Mission Operations. Of these four phases, the Mission Operations phase acts as the main source of habitat design requirements. Given the main objective of this DSE, which entails the design process of the habitat, it is therefore chosen to break down the Mission Operations phase only. The top level breakdown of Mission Operations can be found in Figure 11.1.

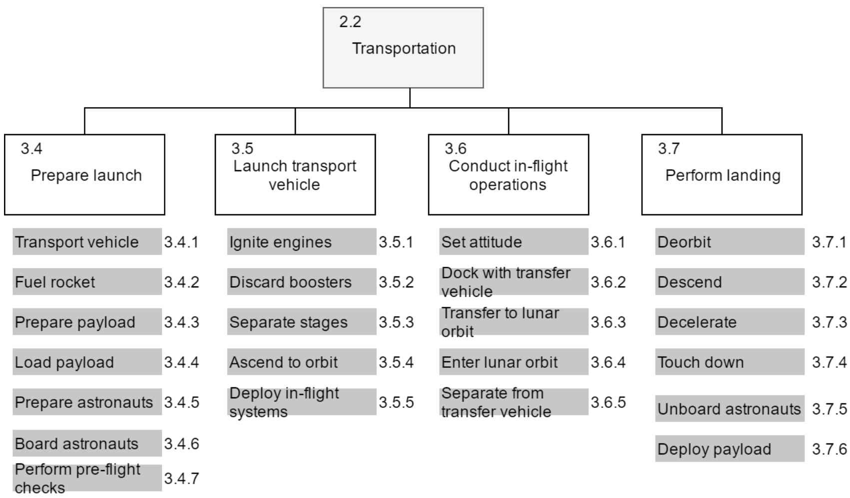


Figure 11.3: Functional Breakdown Structure of Transportation.

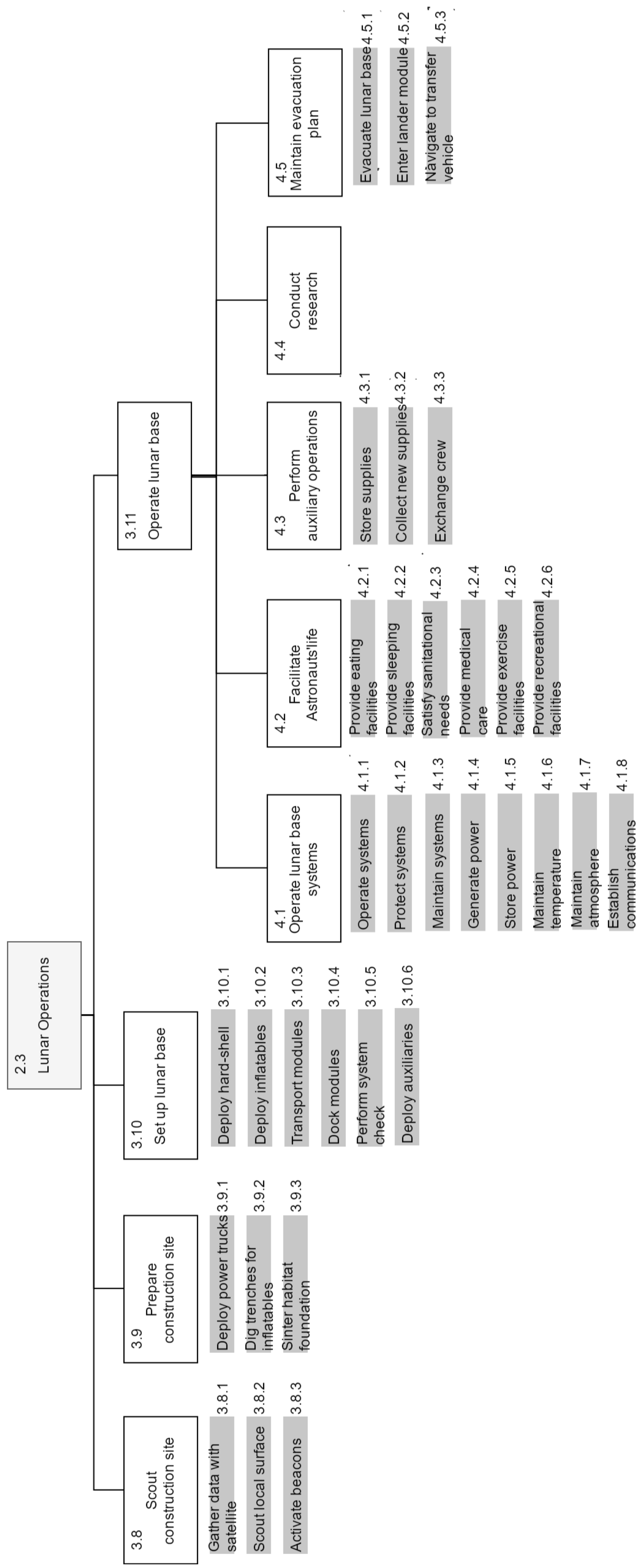


Figure 11.4: Functional Breakdown Structure of Lunar Operations.

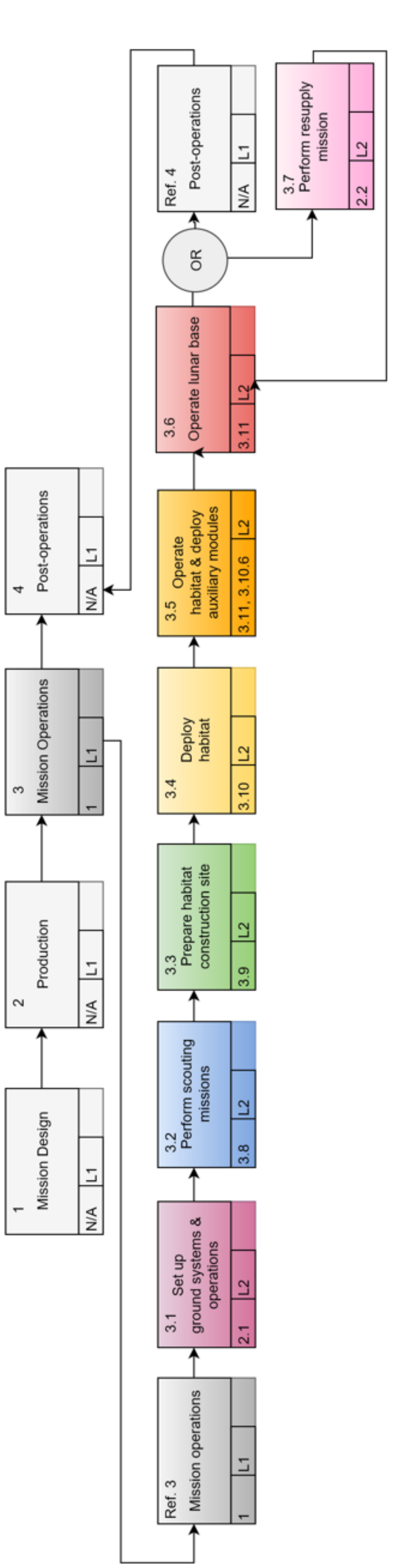


Figure 11.5: Top level Functional Flow Diagram.

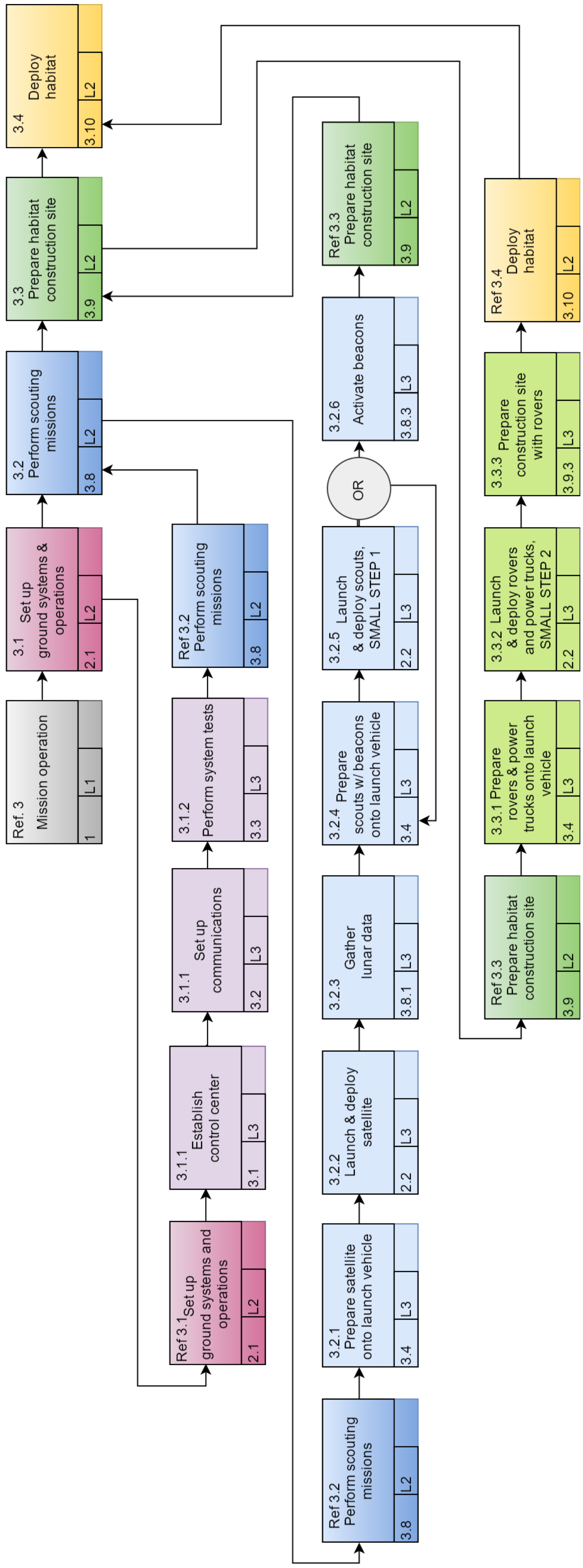


Figure 11.6: Functional Flow Diagram of the preparation phase.

The Preparation Phase

The Ground Operations, although important for the success of the mission, are not considered in detail since they are not part of the scope of this DSE. For that reason, Ground Operations is only been broken down into 3 major operations, as can be seen in Figure 11.2. The second functional branch, namely Transportation, entails all operations which are related to the conveyance of man and material to and from the Moon. The overview in Figure 11.3 shows that Transportation is broken down into four distinctive operations: Launch Preparation, Launch, In-flight Operations and Landing. These operations have all been dissected into smaller actions. It can be noted that not all transportation missions have the same required operations, e.g. the transportation missions concerning building material do not require the preparation of astronauts (3.4.2). The third functional branch of Lunar Operations, which can be found in Figure 11.4, encompasses four important processes: scouting the construction site, preparing the construction site, setting up the lunar base and operating the lunar base. The latter is broken down into five different functions that need to be fulfilled for the lunar base to function properly:

- 4.1 Operate lunar base systems: contains all functions related to the operational systems within the lunar base. This includes the actions performed by the operational systems as well as the operation/maintenance of these systems by astronauts.
- 4.2 Facilitate astronaut's life: entails all habitat functionalities which cater for basic human needs.
- 4.3 Perform auxiliary operations: this entails functions that are not directly related to the aforementioned categories, but which are of importance to the functioning of the lunar base.
- 4.4 Conduct research: as the name suggests, this contains any kind of operation during which scientific data is collected.
- 4.5 Maintain evacuation plan: briefly describes the main actions that need to be performed to return the astronauts to safety in case of an emergency. After these actions, normal transportation operations are continued.

11.2. Mission logistics

The system of operations related to this mission can be organised effectively in a Functional Flow Diagram (FFD). A general description of FFDs can be found in the Midterm Report [2]. Project LEAP can be divided into a sequence of four top-level operations, namely Mission Design, Production, Mission Operations and Post-operations. This can be seen in Figure 11.5. Each of these four elements can be broken down several times into sets of smaller operations, each level containing more detailed actions to be performed. The Work Flow Diagram (WFD), which has been created in the Project Planning [48], covers the entire process of Mission Design, as this is the primary objective of the DSE itself. For the design of the entire mission, the expansion of phase three (Mission Operations) is particularly interesting. The Mission Operations can be divided into three major phases: the preparation phase (3.1, 3.2 and 3.3), the deployment phase (3.4 and 3.5) and the operational phase (3.6) which will be described in the following: As can be seen in Figure 11.6, the preparation phase consists of three major operations.

Operation 3.1: Set up Ground Systems & Operations

Due to the scope of this DSE, the deployment of a ground station has not been considered in detail. The most important milestones defined for this stage were the construction of the ground station itself, the establishment of a communication network around Earth, and a final system check.

Operation 3.2: Perform Scouting Missions & Beaconing

This operation contains the launch and deployment of several payloads which will scout the lunar surface. First, a lunar orbiter will perform accurate measurements of the environment at the target location. Afterwards, four scouting and beaconing rovers (SBR) will be sent up to scout specific locations selected for the lunar base and the different landing sites for the modules and resupplies. During this scouting mission, the SBR will check the local radiation levels, soil mechanics and obstacles in the terrain that went undetected by the satellite. These tasks will be performed using a soil mechanics tester, a camera, a radiation sensor and a Sun sensor and Earth sensor for navigation [49]. In addition to this payload, the SBR will have its own solar panel to power its systems. During the lunar night, the SBR will be

hibernating.
Once the selected locations have been deemed suitable for their purpose, the SBR will drive to one of the beacon locations and deploy its beacon and UHF-antenna on an extension device of at least 2.3m above the ground to ensure communication between the base and the landed payload.
The four beacons make use of a Doppler radar sensor, which requires a K_a-band CW Doppler transceiver and a Lidar system. [50]. They will be placed at the corners of a square surface area of 4 by 4km, as this was found to be the optimal distance [51]. Depending on its trajectory and the landing position it is aiming at during its descent phase, the lander will use three of the four beacons to triangulate its position to land within at most 7m distance from its target [51] [52].
Figure 11.7 shows how the beacons that are used are dependent on the region in which the target lies and the trajectory of the lander. If this construction is adhered to the landing accuracy will always stay within the 7m distance from the target [52].

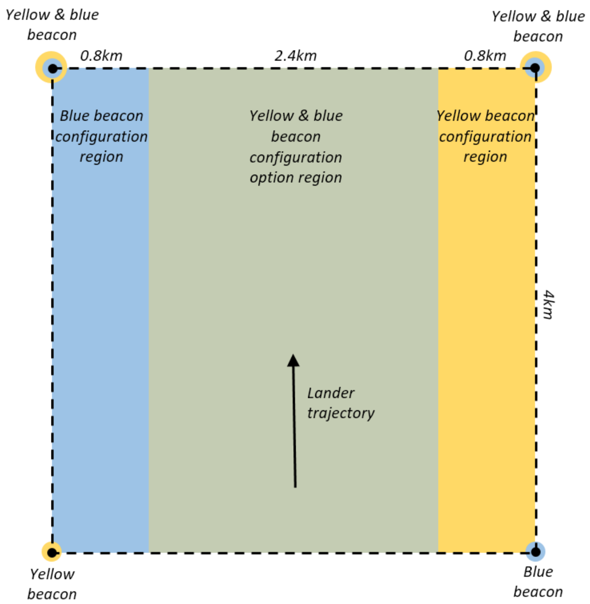


Figure 11.7: Beacon dependencies on target region and lander trajectory.

Operation 3.3: Prepare Habitat Construction Site with Rovers
When a suitable location has been found, rovers will be sent up to prepare the infrastructure for the lunar base. Besides the SBR, three other types of rovers are required. These are the multifunctional lunar rover (MLR), the lunar sintering rover (LSR) and the Space Exploration Vehicle (SEV). As the name suggests, the MLR will need to serve a multitude of purposes. These purposes address base deployment as well as base maintenance. In short, the MLR should be able to:

1. Remove craters and even out the ground with sufficient accuracy.
2. Transport the modules to the assembly site.
3. Carefully align the modules for connection.
4. Extract resupplies from a lander and bring them to the habitat.
5. Transport landers to a dismantling location.

To perform these tasks, the MLR should be equipped with a large pushing blade to scoop away regolith, a powerful arm to push/pull payloads they have been mounted on wheels, a 3D laser scanner for connection alignment, laser sensor and radar for payload lander location determination and optical sensors for controlling the arm.

The LSR serves only one purpose as it stands, which is to reinforce the surface for the lunar village. While the current design does not include a protective layer of regolith around the modules, the LSR might be used for this later on. The LSR will be equipped with solar panels to power its sintering device.

This means the sintering has to stop during the lunar night, which will slow down the sintering process. With current technology, the LSR should be able to sinter 0.0625m^3 of regolith in 24 hours [53]. The LSR will be similar to the All-Terrain Hex-Limbed Extra-Terrestrial Explorer (ATHLETE).

The SEV is a 12-wheeled rover that will be the astronaut's means of transportation across the lunar surface. It comes with a pressurised cabin and is capable of hosting two astronauts with tools for two weeks. The SEV should also be modified, such that it can connect to the habitat's airlocks. Finally, the SEV needs to be able to transport all four astronauts to the lander in case of an emergency.

The Deployment Phase

After the preparation phase, the deployment will start. As can be seen in Figure 11.11, the deployment phase contains two major operations.

Operation 3.4: Deploy Habitat

First, The Shell will be sent up to the target location. After it has landed on the prepared construction site, rovers and sintering robots will prepare reinforced trenches for the inflatable modules to rest in. The inflatable modules will be sent towards The Moon, after which the actual assembly occurs. In short, this assembly phase will consist of transporting the inflatable modules to the location of The Shell, docking them to The Shell and finally performing system checks. While the MLR still needs to be designed in detail, significant research has already been performed in the design of the docking system.

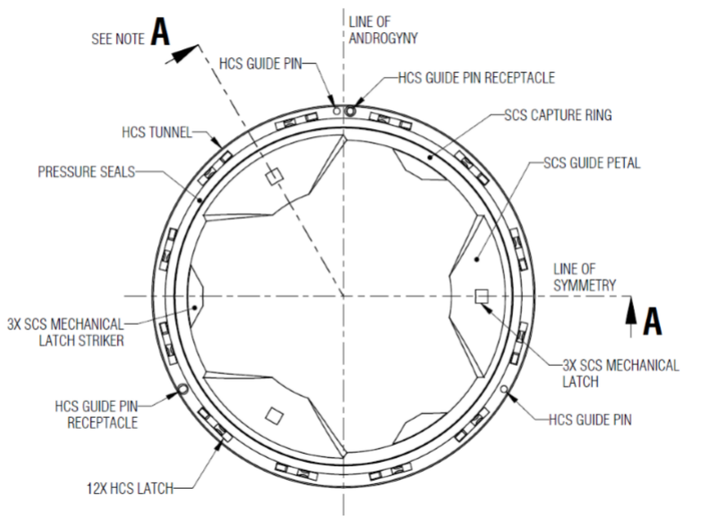


Figure 11.8: Axial view docking interface international standard docking system [54].

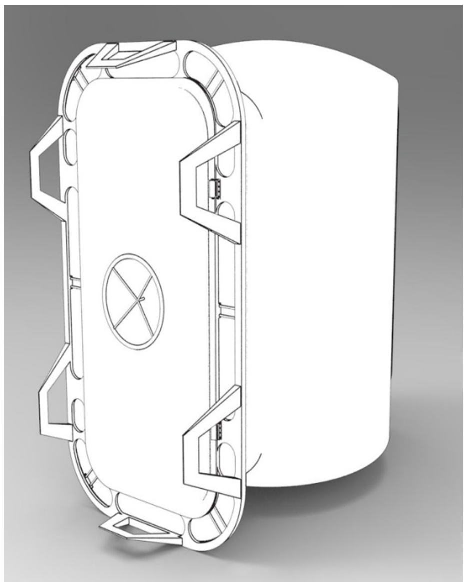


Figure 11.9: Axial view adjusted docking door for LEAP.

The design of the docking doors is based on the International Standard Docking System (ISDS) [54] with some adjustments to make the door more practical for the lunar environment. In Figure 11.8 the axial view of the locking system of the ISDS is shown. The petals in this design are there for the first coarse alignment guidance. There is also a fine alignment guidance in the form of pins in the docking ring and after this alignment, the docking rings will lock into place with their hooking system, that will make the connection airtight.
The docking system has an active and a passive docking ring. The passive docking ring is mounted on the system that is to remain stationary. In the case of the habitat, the passive docking ring is mounted on the hard-shell module. The active docking ring is mounted with a hydraulic system to the system that is to approach the stationary system. For the habitat, the active docking ring will be mounted on the inflatable modules. The precision required to be guided correctly by the coarse guiding system of petals is 10cm in the $Y_{PR}Z_{PR}$-plane, 2.3m in the X_{PR}-direction and 4° in the pitch, yaw and roll angle between

the undeployed active docking ring and the passive docking ring [54]. These distances are with respect to the coordinate systems defined in Figure 11.10.

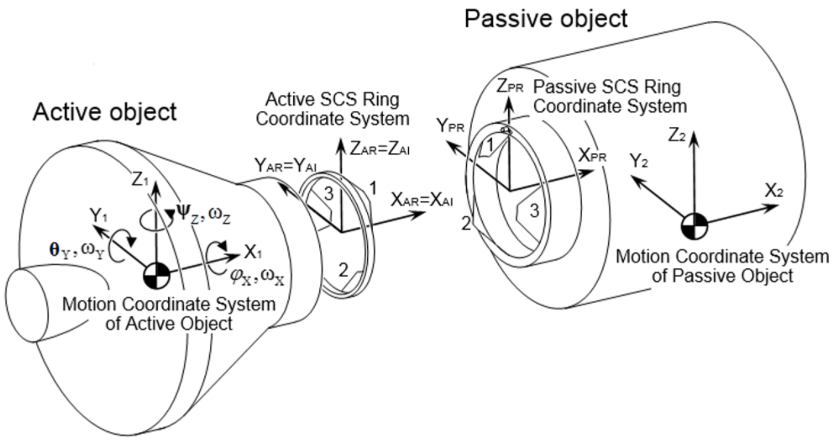


Figure 11.10: Coordinate systems docking objects (active and passive) [54].

Table 11.1: Maximum ISDS interface loads [54].

Load	**Limit Value**
Tension	3900N
Compression (static)	3500N
Compression (dynamic)	6500N
Shear	3200N
Bending	2800Nm
Torsion	1500Nm

The precision in the X_{PR}- and Y_{PR}-direction and in the yaw angle are requirements for the positioning capabilities of the MLR. The passive docking door will be equipped with reflectors and another point for the sensors of the MLR to focus on when aligning [54]. The precision in the Z_{PR}-direction and in the pitch and roll angle are requirements for the hydraulic system in the wheels mounted on the hard ends of the inflatable module. It should be noted that the hard-shell module will first be levelled with its own hydraulic system to provide a level reference point in the docking.

Table 11.1 shows the maximum loads that the ISDS interface can take. These loads should be adhered to during the docking process and while the modules are docked. Around the docking system room should be reserved for different types of interconnections. The interconnections include of course the docking door, the electrical connections and fluid connections. For the lunar habitat, the ISDS is adjusted to better suit the lunar environment that has gravity. The docking door is designed to resemble a submarine door that will better accommodate the astronauts exiting and entering the modules. Some research will still have to be performed to check how the changes to the design influence the specific characteristics of the docking door.

Operation 3.5: Operate Habitat & Deploy Auxiliary Modules
After the system checks have been performed, the first crew of astronauts will be sent to the Moon. While the habitat remains operational by itself for some time, auxiliary modules will be prepared for launch and deployment. Similar to the inflatable modules, these will be transported and connected to the habitat using the same rovers and docking systems.

The Operational Phase
After the auxiliary modules have been connected, nominal operations will happen for the majority of the habitat's lifetime. Figure 11.12 indicates the most significant operations during this phase, namely base maintenance, conducting research and facilitating the astronauts' lives. After a 10 years, a decision needs to be made between two options, as shown in Figure 11.5: either perform a resupply mission or move to the post-operational phase of the mission, which entails the extraction of the last crew and the termination of operational systems. As can be seen in Figure 11.12, the resupply mission can be in the form of consumables and/or a new astronaut crew.

11.3. Launch logistics
Evidently, project LEAP will require a variety of payload launches to The Moon. In Table 11.2 and Table 11.3, the launch options are presented of every major payload launch, or Mission Step. The payload

mass that each launch vehicle can transport to the lunar surface, is taken from Table 10.1. The very first launch is not considered a Mission Step, as this is expected to be a small payload piggybacking on another lunar orbiter. The objective of that payload is ”to take radiation and micrometeorite measurements and map the vicinity of Apollo 11”. Table 11.2 shows the items to be included in SMALL STEP 1 and SMALL STEP 2, the payload mass and the room left in these launches[1][2]. The options marked in green will be used for the mission. As can be seen in Table 11.3, the SLS is the only viable option for GIANT LEAP 1 and GIANT LEAP 2, as the habitat modules are simply too heavy for the Falcon Heavy.

Table 11.2: Preparation phase launch breakdown [50] [55].

SMALL STEP 1						
Mission	**Payload**			**Transportation**		
	Items		**Mass [kg]**	**Launch Vehicle**		**Room Left [kg]**
Scout location, check soil mechanics specific spots, place beacons	4	SBR	3360	1	Falcon Heavy	3715
	5	Beacons	25	1	Angara A5V	915

SMALL STEP 2						
Mission	**Payload**			**Transportation**		
	Items		**Mass [kg]**	**Launch Vehicle**		**Room Left [kg]**
Create infrastructure (levelling, sintering) for in-situ transportation and habitat/base foundation	1	MLR	5000	1	SLS	1383
	1	LSR	1500	3	Falcon Heavy	4883
	3	Power Ranger	6917			
	1	SEV	3000			

Table 11.3: Deployment phase launch breakdown.

GIANT LEAP 1						
Mission	**Payload**			**Transportation**		
	Items		**Mass [kg]**	**Launch Vehicle**		**Room Left [kg]**
Send the first module of the habitat	1	The Shell	12019	1	SLS	1170
	2	Power Ranger	4611	N/A	Falcon Heavy	N/A

GIANT LEAP 2						
Mission	**Payload**			**Transportation**		
	Items		**Mass [kg]**	**Launch Vehicle**		**Room Left [kg]**
Send the second and third module of the habitat	1	The Hive	7989	1	SLS	8
	1	The Nest	9803	N/A	Falcon Heavy	N/A

During the ten years of operations, every year a new crew of four astronauts will be sent to operate the habitat and the old crew will return to Earth. In addition to the astronauts being resupplied, the systems in the habitat also need to be resupplied. Table 11.4 show the minimum, maximum and average resupply mass per year.

The values in Table 11.4 for the atmospheric control, power and bioastronautics have been determined based on the precise subsystem parts and consumables that have to be supplied. The atmospheric control requires yearly resupplies of N_2 to account for leakage and filters to replace the old filters. The power trucks will twice require a resupply of fuel cells during the ten-year mission, due to degradation. The bioastronautics requires a yearly resupply of water, food, medical supplies and system parts.

For the habitat structure, it is estimated for the hard-shell module that certain connector pins will have to be replaced regularly. This together with some other small parts in the hydraulic system will add up to somewhere around 3kg for each year. For the inflatable module, it is estimated that the shell will hold for the entire mission duration. However, in case the module needs to be patched up due

[1]https://www.nasa.gov/exploration/technology/space_exploration_vehicle/index.html [Cited:22-06-2017]
[2]https://nssdc.gsfc.nasa.gov/nmc/spacecraftDisplay.do?id=1973-001A [Cited: 22-06-2017]

Table 11.4: Resupply sizing.

System	Yearly Minimum [kg]	Yearly Maximum [kg]	Yearly Average [kg]
Structure (Hard-shell)	3	3	3
Structure (Inflatable)	0	10	2
Atmospheric control	723.6	723.6	723.6
Radiation protection	0	129	25.8
Micrometeorite protection	8	12	10
Thermal control	0	50	10
Power	0	516.0	103.2
Bioastronautics	4377.3	4377.3	4377.3
Interior	0	7.2	1.4
Communication & Data Handling	0	50	5
Auxiliary systems	0	950	190
TOTAL	5111.9	6828.1	5451.4
Mass left in launcher (Falcon Heavy)	2088.1	371.9	1748.6
Margin of safety	1.41	1.05	1.32

to significant damage done by (micro)meteorites hits, a 10kg resupply is taken into account. For the average, it is assumed that this might occur two times during the ten-year operations. For the radiation protection, it is assumed that about 5% of the coils will have to be replaced twice during this time. The same assumption was taken for the interior. For the thermal control and the auxiliary systems, the 5% is increased to 10%. This is to account for the fact that the radiators and the rovers will be outside, which will increase the likely hood of them being damaged. The meteorite protection of the hard-shell module consists of separate whipple shield panels. It is estimated that yearly one of these panels will be damaged that much that it would be preferable to replace it. These panels have a mass between 8kg and 12kg. Assuming it is as likely that either size is hit the average mass to be resupplied will be 10kg. Regarding the communication & data handling system it is assumed based on current technology that the majority of hardware will require one replacement in the ten years of manned mission operations.

In conclusion, based on the values stated in Table 11.4 even in the case that all systems require the maximum resupply mass this can still be fitted in the Falcon Heavy together with 371.9kg of miscellaneous supplies. Taking in mind that the planned major supplies can be coordinated in a way that they don't have to be in the same resupply, the Falcon Heavy should provide enough room to fit the desired payload with a nice margin of about 1748.6kg. However, it should be taken into account that once the base is extended more resupplies will be required.

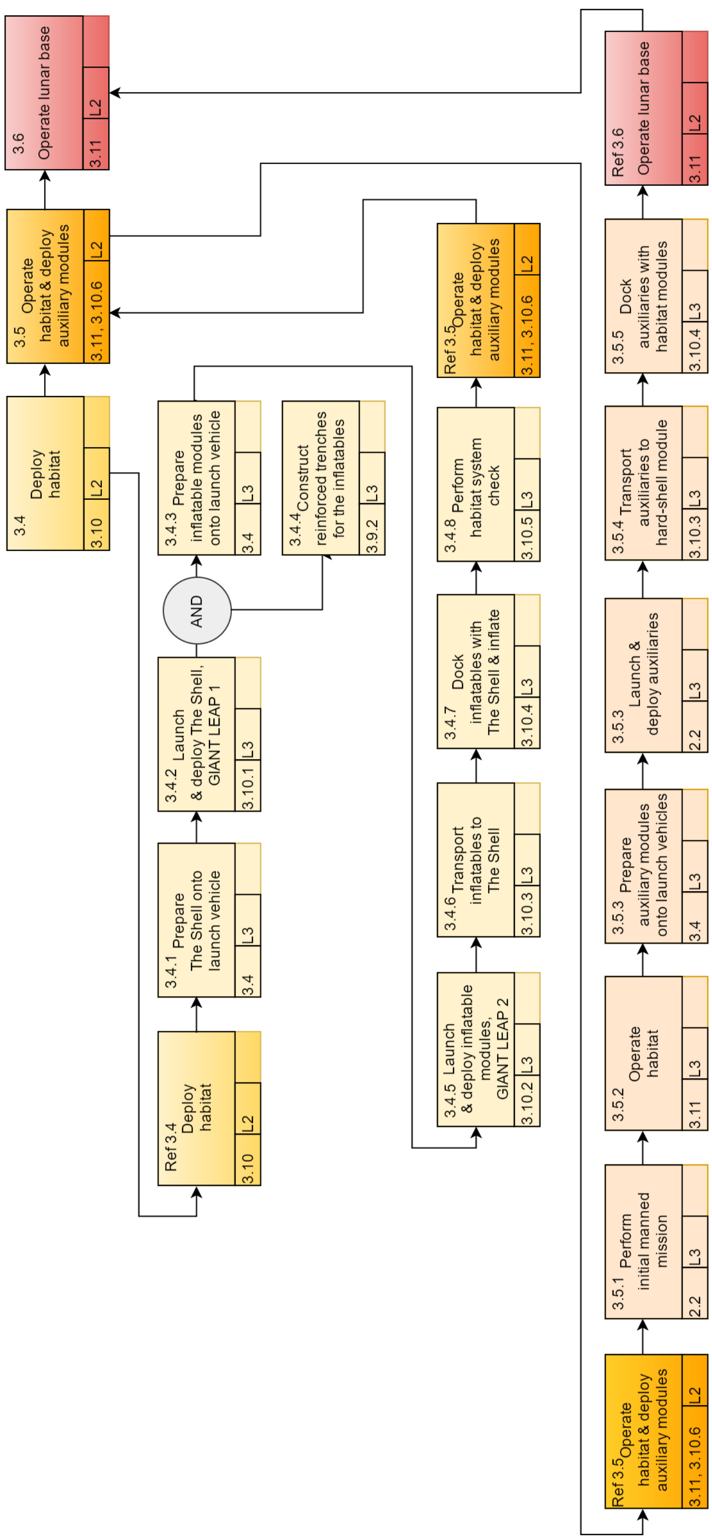


Figure 11.11: Functional Flow Diagram of the deployment phase.

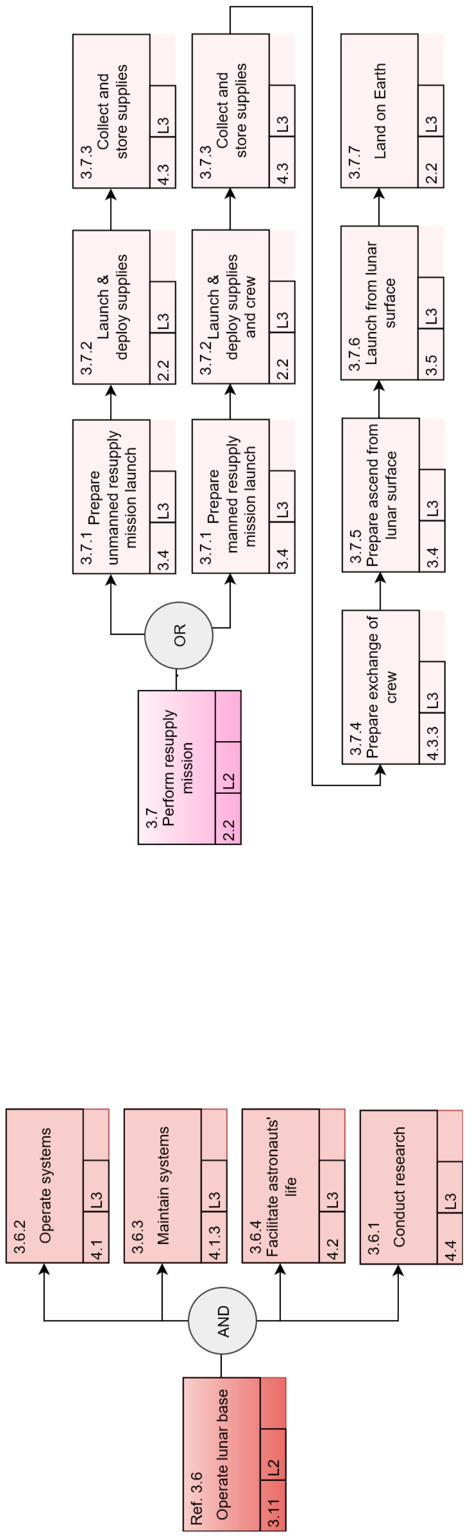


Figure 11.12: Functional Flow Diagram of the continuous/periodic operations.

12 Technical Risk Assessment

Analysing risks is a task which has to be performed at the start of every project, in order to raise awareness of which risks the design can have and design for it. This chapter addresses the technical risks during the mission. Risk management is a continuous process, as the severity or the likelihood of a risk occurring may change as the different phases of the project proceed. In section 12.1 the different risks are identified and structured, followed by the mitigation plan for the most critical risks in section 12.2.

12.1. Risk Assessment

By examining the different phases of the mission, systems and subsystems, potential hazards can be found. The determined risks are reported from Table 12.1 until Table 12.7. It can be seen that the risks are categorised based on the specific subsystem or phase in the mission, whereafter the cause is depicted, followed by the failure mode. This is then followed by the consequence the failure has on the lunar habitat itself or on the entire mission and accompanied by a clarification whether this affects the functioning of the lunar habitat or the survival of the astronauts. Afterwards, the likelihood and effect are quantified. The former is based on the maturity of the technology, while the latter is based on the impact it has on the mission. Events that endanger the lives of the astronauts are deemed to have the biggest impact for the LEAP mission. Finally, in the last column of the tables, the mitigation plan which has to be undertaken to reduce the risk is described.

The risks depicted from Table 12.1 until Table 12.7 are sorted based on the severity of their impact and the likelihood of them occurring, leading to the so-called risk map that can be seen in Figure 12.1. As can be noted, in the previously mentioned figure the risks fall under different categories: high, medium and low. The objective of the risk mitigation plan is driving the risks from the top right corner into the bottom left. Furthermore, one may observe that in general, the risks involving the survival of the astronauts have a high impact with respect to the risks which involve the operation of the lunar base. The risk map after the mitigation has been applied can be found in Figure 12.2.

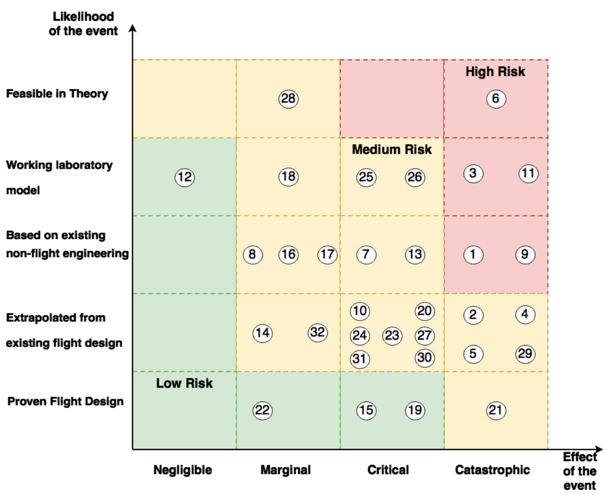


Figure 12.1: Risk map before mitigation.

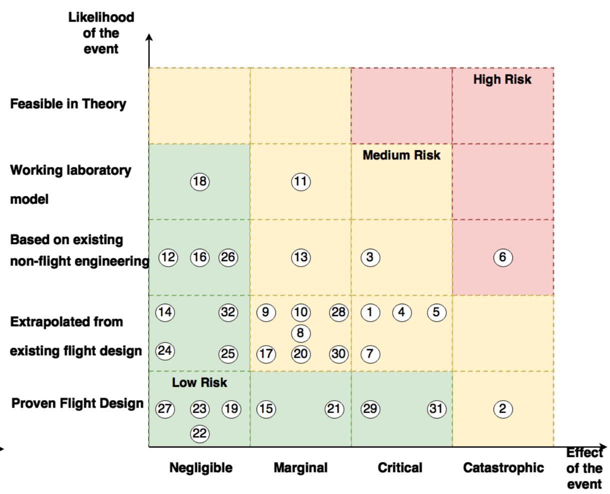


Figure 12.2: Risk map after mitigation.

12.2. Risk Mitigation Plan

The highest risks should be mitigated as much as possible in order to maximise the probability of a mission success. In this section, the most critical risks are addressed and according to that, a mitigation plan is employed and presented. As stated in section 12.1, the objective is to relocate the risks which are situated in the upper right corner of the risk map towards the bottom left corner, such that they become a medium or preferably low risk. In order to achieve this, two actions can be undertaken: the likelihood is lowered by using technically proven designs while the other option is reducing the impact by employing preventive measures.

The biggest step taken for mitigation is designing an emergency capsule and airtight safety doors in between the different modules. The airtight doors make sure that if one part gets inoperational by any cause or accident, surviving in the rest of the habitat is still possible. In case of a total failure of the survival systems within the entire habitat, a self-sustaining safe room is implemented in the design. This safe room provides means of survival for a week which is deemed to be enough for the Earth control to send a rescue mission, but will mainly be used in case of dangerous circumstances such as solar particle events. The survival time can be adjusted accordingly if concerns arise.

In general, monitoring and maintenance are the main actions to be undertaken when reducing the risk level. Preventive maintenance is employed to intercept potential failure. The main focus for this procedure lies on thermal control, power and the communication subsystems as a failure of these systems have a critical impact on the mission success and the safety of the astronauts. Furthermore, it is suggested to employ back-ups in order for the vital functions to be carried out in case of a subsystem failure.

As can be noted in Figure 12.1, one of the highest risks is the failure of an airlock. This is due to the fact that there is very little experience with respect to the design of airlocks enabling astronauts to go from the pressurised habitat to the vacuum environment regularly, for activities as repairs and research. In addition, within the lunar base, it is important that the astronauts are able to move freely between the different modules, if these are not connected through a pressurised area. Frequent use of habitat intern airlocks may lead to early failure due to fatigue. Thus, the system should be thoroughly inspected and repaired regularly as a measure of prevention.

Furthermore, the failure of the exit airlock will impede the astronauts off accessing or exiting the habitat. The use of existing airlocks is favourable in terms of reliability, however, a fatigue test should be carried out in any case. Furthermore, in order to reduce the impact of a failure, redundancy is introduced by integrating another airlock; in this way the astronauts are still able to move between the habitat and the environment, enabling the repair of the damaged airlock. Implementing these changes reduces this risk from high to medium as can be noted in Figure 12.2.

Another big risk is the moon dust infiltrating into the habitat. There is not a lot of experience yet with moon dust, there are some filtration concepts with strong magnets, but none of these systems is used on the Moon yet. It is known that it is dangerous for the health of astronauts and the operationality of systems, so it is vital to keep it outside the habitat. To make sure the system will be able to do this, tests have to be performed for the airlocks where the presence of moon dust is simulated. Experiments can be done with for example volcanic dust, which has some of the same characteristics as moon dust. Similar actions are undertaken in order to reduce the level of the different risks. The mitigation suggested to reduce the risk level of the other risks can be seen from Table 12.1 until Table 12.7.

Comparing Figure 12.1 to Figure 12.2, it can be noted that the risk of failure during launch of either the lunar habitat or the astronauts can not be mitigated. There is a difference in severity and probability, which arises from the fact that there is more experience in launching humans into space in comparison to habitat components and that the severity is naturally higher in case the launch of the astronauts fails as this will lead to casualties. It is already assumed that the most reliable technology is used, resulting in the highest probability of a successful launch and therefore can not be changed.

Table 12.1: Technical Risk Assessment and Analysis.

No	Category	Cause	Event	Consequence	Impact	Likelihood	Effect	Mitigation
1	Habitat Structure	The habitat is subjected to an unforeseen load case	The load carrying structure of the lunar habitat are subjected to higher loads than designed for	The structure of the lunar habitat collapses	Health of astro-nauts and operation of Habitat	Based on existing non-flight engineering	Catas-trophic	Perform an extended study on the conditions that may occur on the Moon and afterwards add safety factors for unforeseen situations. An extra strong safe room (The Shell) is implemented where the astronauts can survive for at least 5 days.
2	Habitat Structure	The ambient monitoring and handling system for the lunar habitat fails	There is a loss in atmospheric conditions within the habitat	The astronauts are not able to operate without additional support	Health of Astro-nauts	Extrapolated from existing flight data	Catas-trophic	Monitor the atmospheric system and perform regular maintenance in order to keep the system fully operational, healthy and prevent failures.
3	Environmental Protection	Bigger meteorites than expected struck into the habitat	The structure of the lunar habitat is struck by a bigger force than accounted for caused by meteorites	There will be a loss in atmospheric conditions within the habitat. The astronauts are not able to operate nominally and the structure will (partially) implode	Health of Astro-nauts and operation of habitat	Working laboratory model	Catas-trophic	Keep track of larger object which can damage the protective structure such that potential hazards are identified and can be accounted for beforehand. Next to that, all modules are able to be closed off by airtight doors.
4	Environmental Protection	There is a failure in the thermal control of the habitat	The habitat falls outside its operational temperature range	The astronauts are not able to operate the habitat while under extreme temperatures. Systems that have operational temperature range are not able to operate anymore	Health of astro-nauts and Operation of Habitat	Extrapolated from existing flight design	Catas-trophic	Have backup heat sources such that vital systems can be kept operational during the inoperable period of the thermal control system. The airtight doors prevent the whole habitat to become inoperational.

Table 12.2: Technical Risk Assessment and Analysis.

5	Environmental Protection	There is a failure in the atmospheric control system of the habitat	There is a loss in the atmospheric conditions within the habitat.	The astronauts are not able to operate without additional support	Health of Astro-nauts	Extrapolated from existing flight design	Catas-trophic	Have backup atmospheric control sources. The airtight doors will make sure only part of the habitat will loose its atmospheric conditions. Oxygen masks are available for for emergency situations.
6	Environmental Protection	moon dust infiltrates the habitat	moon dust penetrates the subsystems of the habitat and is inhaled, touches the skin or eyes of the astronauts.	Respiratory system, eyes and/ or skin of the astronauts get damaged by the particles of moon dust. The moon dust wreak havoc on the subsystems inside the habitat.	Health of Astro-nauts and Operation of habitat	Feasible in theory	Catas-trophic	Design the airlock in such a way that it is also able to get rid of the moon dust. moon dust consists of very small sharp particles that have a high static electricity. Tests have to be performed where the presence of moon dust is simulated, for example volcanic dust has similar properties as moon dust.
7	Environmental Protection	There is a failure in the radiation protection	The radiation protection in the hard-shell cannot be operated safely	The astronauts will receive an elevated amount of radiation	Health of astro-nauts	Based on existing non-flight design	Critical	Regular checks shall be performed during the manufacturing process and during the habitat operations to detect and repair any defects before a solar particle event is at hand.
8	Environmental Protection	There is an unforeseen interaction between the protons and the radiation shielding	The magnetic shield is unable to fully redirect the particle radiation	The astronauts will receive an elevated amount of radiation	Health of astro-nauts	Based on existing non-flight design	Marginal	Further research has to be performed to ensure all phenomena are mapped and enable applying more power to the shield to deflect more particles

Table 12.3: Technical Risk Assessment and Analysis.

9	Power Subsystem	There is a failure in the power storage subsystem	The power storage is not able to store the required amount of energy	The habitat is not foreseen of power during periods in which no power is produced or does not possess a buffer for when more power is requested than is produced	Operation of habitat	Based on existing non-flight engineering	Catas-trophic	Monitor the cycles of the storage system in order to anticipate eventual malfunctions. Have multiple power storage units in order to create redundancy.
10	Power Subsystem	There is a failure in the power supply subsystem	The power supply is not able to produce the required amount of power	There is no sufficient power supplied to the habitat impeding all systems to operate nominally	Operation of habitat	Extrapolated from existing flight design	Critical	Operate only the vital systems of the habitat, make use of the stored energy if needed. Include in the design extra power supply systems for redundancy.
11	Bioastronau-tics	There is a failure in the airlock system	The airlocks are not able to provide the desired pressure difference between the Moon and habitat environments	Astronauts are not able to move between the lunar environment and the habitat	Operation of habitat	Working laboratory model	Catas-trophic	Operate the airlocks as designed in order to prevent failure. Create multiple airlocks for the habitat module such that the failure of one airlock does not impede the astronauts to enter or exit the habitat
12	Bioastronau-tics	There is a failure in the waste system	The waste system is not able to handle the waste produced by the astronauts	There is an accumulation of waste	Operation of habitat	Working laboratory model	Negligible	Monitor the operation of the waste system in order to anticipate eventual failures
13	Bioastronau-tics	There is a failure in the water management system	The water management system is unable to recycle the water	Used water is not being filtered and potable anymore	Operation of the habitat	Based on existing non-flight design	Critical	Install emergency water tanks to provide water for astronauts during system failure

Table 12.4: Technical Risk Assessment and Analysis.

14	Communication and data handling	There is a failure in the communication system between Earth and the habitat	The astronauts are not able to communicate with Earth	The stationed astronauts are not able to receive commands and personal communication with the ground station	Operation of habitat	Extrapolated from existing flight design	Marginal	Monitor the communication system performing regular maintenance in order to prevent failure. Provide a backup communication system for critical communication with Earth for ground support. The habitat under normal conditions should be independent of ground control
15	Communication and data handling	One or more UHF antennas fail to operate	The UHF antenna network does not cover the full operational area anymore	Astronauts or system might be out of communication reach	Precision landing, astronaut safety, village operations	Proven flight design	Critical	Only 3 antennas are needed for precision landing, and 4 antennas are available initially and 5 after the soft-shell modules have landed. The operational area has to be reduced untill the antenna is repaired or replaced or expeditional rovers have to carry higger antennas.
16	Communication and data handling	Internal wifi routers fail	WiFi network does not cover the full habitat area anymore	No or limited data transfer is possible within the habitat	Habitat operations	Based on existing non-flight design	Marginal	Typical router issues are software based and can easily be fixed. Also spare routers are available, as well as cable to make wired connections.
17	Communication and data handling	A laptop experiences a hardware issue	Laptop cannot be used anymore	Operations with laptop cannot be performed as desired.	Habitat operations	Based on existing non-flight design	Marginal	A spare laptop is available and can be used entirely or in parts. It can be replaced when a following payload arrives. Some parts might be replaced by 3D printing a spare.
18	Communication and data handling	A laptop experiences a software issue	Laptop does not behave as desire	Operations with laptop cannot be performed as desired	Habitat operations	Working laboratory model	Marginal	Software issues can always be resolved by a full reboot and consequences will almost be temporary. Back-ups of data can be restored to continue operations.

Table 12.5: Technical Risk Assessment and Analysis.

19	Communica-tion and data handling	Central data bus fails	Central data bus fails	Data is not properly interchanged or saved	Habitat opera-tions and Astro-nauts health and safety	Proven flight design	Critical	A second data bus runs simultaneously with the main bus as immediate back-up. Laptops can be configured as server in case of further redundancy.
20	Transportation	There is a failure during launch of the habitat or during the transport mission to the Moon	The launch vehicle fails to deliver the habitat to the Moon	The lunar habitat does not become operational	Loss lunar habitat	Extrapolated from existing flight design	Critical	Select a proven reliable launch module.
21	Transportation	There is a failure during launch of the astronauts or during the transportation mission to the Moon	The launch vehicle fails to deliver the astronauts to the Moon	The astronauts will die and the habitat will not be operational	Loss of astro-nauts	Proven flight design	Catas-trophic	Select a proven reliable launch module.
22	Transportation	There is a malfunctioning in two of the precision landing beacons	The lander is unable to triangulate its position with three beacons	The lander lands with an accuracy of 250m	Operation of the habitat	Proven flight design	Marginal	Design payload retrieval rovers that can cover the additional distance and still retrieve the lander.
23	Transportation	There is a failure during launch of the resupply cargo	The launch fails to deliver the resupply cargo to the Moon	The habitat will not get resupplied with resources and vital resources will run out soon	Operation of habitat and health of Astro-nauts	Extrapolated from existing flight design	Critical	Plan the resupply missions on a moment that if something goes wrong a second flight can be scheduled and performed before the habitat runs out of resources.
24	Transportation	During launch unpredicted loads are experienced	The lunar habitat experiences loads for which it is not designed	The habitat structure fails under the excessive loads	Operation of habitat	Extrapolated from existing flight design	Critical	Perform extensive testing and simulation before launch and make use of safety factors during design to account for unforeseen loads.

Table 12.6: Technical Risk Assessment and Analysis.

25	Assembly	During the stationing of the lunar habitat one system or part of the habitat fails to deploy	The lunar habitat is not able to deploy on the Moon	The lunar habitat does not become operational	Operation of habitat	Working laboratory model	Critical	Perform extensive testing and simulation before launch in order to ensure a reliable deployment system.
26	Assembly	The different modules of the habitat fail to attach	It is not possible to go from one module to another via an airtight pressurised area	The lunar habitat does not get operational as it is meant to be	Operation of habitat	Working laboratory model	Critical	Focus during testing and simulation on the differences of the ISS module connections and the lunar habitat connections and how this changes the design. Make sure all the modules have an airlock themselves, so the habitat is operatable until the connections are repaired.
27	Selenology	The lunar landing location is not as expected	The lunar habitat can not be deployed on the selected location or attached to the ground	An alternative location of deployment or the chosen location have to be altered	Operation of Habitat	Extrapolated from existing flight design	Critical	Perform extensive research on the lunar landing location. Take the information gathered from the Apollo missions. In addition, make sure the habitat is not very location specific, but could be deployed in different spots.
28	Auxiliary Units	The auxiliary units are subjected to conditions for which they are not designed	The auxiliary units become inoperational	The auxiliary units become inaccessible impeding the astronauts to perform the scheduled operations	Operation of lunar base	Feasible in theory	Marginal	Make the lunar habitat independent of the auxiliary units such that survival of the astronauts is ensured

Table 12.7: Technical Risk Assessment and Analysis.

29	Operations	Short circuit occurs or hot gases inflame	A fire occurs inside the habitat	Fire damages the systems and endangers the life of the astronauts	Operation of habitat and health of astro-nauts	Extrapolated from existing flight design	Catas-trophic	Test all materials going into space for flammability. In addition, implement smoke detectors into the ventilation system to detect fire at an early stage. The compartment where the fire starts can be closed off by airtight doors. Furthermore, implement multiple fire extinguishers in the habitat.
30	Operations	Non-fatal accident or illness of astronaut happens	The health of the astronaut is damaged and will not be able to do his/her scheduled tasks	The other astronauts take over the tasks or certain tasks are omitted. In addition the life of the astronaut could be endangered	Operation of habitat and health of astro-nauts	Extrapolated from existing flight design	Critical	Have first aid systems available in the habitat. Astronauts can perform first aid on each other to minimise the non-operational time and life-threatening situation.
31	Operations	Fatal accident or illness of astronaut happens	One of the astronauts dies and the habitat needs to operate with only 3 astronauts	The other astronauts need to take over the tasks of the passed away astronaut, while coping with psychological problems.	Operation of habitat	Extrapolated from existing flight design	Critical	Perform extensive health checks on the astronauts before launch and during operations to make sure no sudden deaths will occur. In addition, make the habitat operatable for only 3 astronauts and provide mental support for the remaining astronauts
32	Interior	An accident happens and something falls or hits the floor hard	A piece of furniture breaks or gets damaged	The piece of furniture can not be used anymore. If it is an important piece, it can prevent certain actions for the astronauts.	Operation of habitat	Extrapolated from existing flight design	Marginal	Most of the furniture pieces will be made from PEEK material. A 3D printer will be available in the lunar habitat, as well as bulk material of PEEK. The broken part can thus be 3D printed and the furniture can be repaired.

13
Technical Verification & Validation

This chapter lays out the verification and validation of the DSE phase of this project. Firstly, the requirement validation is described. Secondly, the validation and verification process of all design tools is described.

Requirement Validation

Throughout the mission, every design which is made should be checked whether they are in accordance to the requirements of the customer. This means that the customer input has to be analysed and translated to requirements for the design, in such a way that the mission capability of the design can be validated using these requirements. The requirements of the habitat and its part in the LEAP mission have been checked to be VALID (verifiable, achievable, logical, integral and definitive). During this validation process, requirements have been altered to match specific data that has been determined during the research and design phase. This was necessary to account for unprecedented influence from systems on each other and new specifications that have been discovered during the design phase. The new list of requirements and the compliance can be found in chapter 14.

Design Tool Validation and Verification

Different tools have been written and used for designing different subsystems of the habitat. Initially, assumptions were identified together with every programmer. Afterwards, every programmer performed a verification and validation process of subsystem parts which are shown below.

Module Structure

The main structural component of The Shell was designed and analysed in CATIA. CATIA is a verified software and hence not the results have to be verified and validated, but the inputs. Both can be found in chapter 4. Two assumption are made and deemed valid: 1) local stress concentrations from the protective shell is taken into account, since the internal pressure forces are significantly larger anyways, 2) internal pressure only acts on the larger surfaces. To verify the inputs of the structural analysis, i.e. the atmospheric pressure and the clamped bottom, the CATIA model was inspected by several members of the group. A similar process was carried out for the analysis of the mounting pins. The design process of the truss structure was verified by inspection too, yet in the form of reproducing the calculations by hand by another person.
The inflatable load bearing structure is the restrainer layer as found in section 4.4. Here, the minimum thickness needed and the resulting structural mass is calculated by an Excel file using the which was checked by hand calculations. The properties of the materials used are taken from CES EduPack ([10]).

MMOD Design

The MMOD protection is designed using the MMOD protection handbook of NASA([5]), which is in accordance with the hard- and soft-shell micrometeorite protection approach. The tool used for the design of The Shell's meteorite shield is an Excel sheet, which only uses the equations from [5] to determine the shielding performance. Besides that, relations have been applied for the conversion of shielding reliability to projectile size, however, these follow merely from probability and statistics (as explained in section 3.1). Regarding the input of the spreadsheet, the following has been used:

1. For the thickness and density of the MLI, a cross-reference is made to the tool which is used for thermal control.
2. For the vulnerable surface area of The Shell, a cross-reference was first made with the tool that is used for structures. At a later stage, this value was updated manually in accordance with the surface area that CATIA measured for the module.

While designing the meteorite shield, there was little need to make assumptions. The design and performance equations provided by [5] left room for a few uncertainties only. For every uncertainty, it was decided to stick with a conservative approach:

1. Some of the constants in the performance equations for the intermediate and high-velocity regimes (Equations 3.5 and 3.6) could take on different values depending on certain mass ratios within the shield. However, these values were only given for limited ratio intervals, which is why the most conservative values have been assumed here.
2. The MLI could contribute to the shield's performance in two ways. On the one hand, it can improve the ballistic performance, by simply adding more material which can stop projectiles. On the other, it can also improve the hypersonic performance (i.e. the shattering of projectiles) if the MLI were located towards the front bumper. The ballistic contribution of MLI inside the shielding system was only given for the situation where the MLI was located directly on top of the rear wall. To remain conservative, it was assumed that the MLI had no ballistic contribution if it were located anywhere but there. Iterations showed that it was most beneficial to install the MLI on top of the rear wall, as the ballistic projectiles were more problematic. This is why this assumption did not affect the credibility of the outcome.
3. The meteorite flux statistics shown in Figure 3.2 assume a projectile density of only 1g/cm^3 to relate the yearly meteorite flux to the projectile size. Due to a lack of additional information, it has been decided to apply the flux values given in Figure 3.2, even though the density of the design projectile is 2.7g/cm^3. This means that the shield is actually designed for a heavier projectile than the literature suggests it should be designed for.

The design tool used for the inflatable structure is a Python program, which calculates the critical design diameter of the projectile d_{crit} hitting the MMOD shield with respect to the blanket under consideration. The minimum critical diameter the blanket needs to sustain is calculated by the same excel sheet used for the hard-shell module. The equations are set-up according to the NASA MMOD handbook ([5]) and the more recent article [13]. Then, the implementation of the equations is checked for correctness with already tested blankets (medium and thick as seen in section 3.2). The article featuring these blankets ([13]) also contains a graph displaying d_{crit} as a function of velocity. The program set-up was said to be verified if since was possible to recreate the displayed graph (Fig.5 of [13]) using the same layer and material configurations. Furthermore, the mass calculations of the MTB layers are performed in two different programs: The MMOD Python program as well as in an Excel file. Neglecting inaccuracies, the values match and the calculations are verified.

Passive Thermal Control

The passive thermal control is designed using existing formulas from SMAD [9] for thermal insulation. The material properties are either found in the CES Edupack or on material property websites.
The program written for the inflatables is an Excel file that calculates the incoming heat for the different angles of the Sun. The assumptions made are the following:

- The conductivity of the regolith in which the inflatable is laying is negligible, so it is assumed to insulate perfectly. This is a valid assumption for the time the inflatables are fully operational. During heating up of the inflatables, some heat will be lost to the regolith up until the point the regolith reaches its equivalent temperature, after which the heat loss to the regolith is negligible. Because of low conductivity regolith tends to have a stable equivalent temperature [16].
- The airlocks do not loose any excessive heat. This is assumed since the airlocks used are already existing airlocks. It is not the scope of this project to design the airlock and its thermal system.
- The outer layer of beta cloth is assumed perfectly conductive. This is assumed since the resistivity of the beta cloth is negligible. [15].

In addition, a unit test has been performed. The graph made is checked on sensibility, for example on the shape and the differences between day and night. When it was checked by someone who did not make it, it was found that the area used was still an old estimate. The changes have been implemented in the design. Furthermore, the outcome of the inflatable shells' thermal control design tool is compared to the outcome of the hard-shell thermal control design tool, which is a different program. By examining

difference in structure and materials the differences in the outcome are verified.
The program written in Python for the hard-shell is verified by previous program that was written. The first program was written for a solid shell, but after iterations of the structure and the meteorite protection a new program had to be written for using panels instead of a continuous shell. Comparing the outcome of the first program with the second program verified them since they were at the same order of magnitude. The fact that the heat flow is negative all the time is critically discussed and researched. Though, after re-examining the calculations done and doing a sensitivity analysis based on the numbers filled in for the emissivity and absorptance the program is verified. The assumptions made for the design are based on proven data, e.g. from the ISS. The most important assumptions that were taken are the following:

- The pins of the hard-shell are so thin that the heat loss due to these pins was assumed to be negligible. To be sure of this assumption a calculation was made in Table 3.1. However, this calculation showed that the heat loss of these pins could not be assumed negligible, so they were added in the model
- The heat flow going through the floor is assumed as if outside is outer space. Using this assumption the maximum possible heat flow out is calculated. The number is only used for the amount of heating needed and not for the amount of cooling needed. This way the maximum possible cooling energy and maximum possible heating energy can be generated. So the assumption is valid, though in the future optimisation should take place
- As with the inflatable, the airlocks of the hard-shell are assumed to have the same thermal properties as the rest of the habitat

The Active Thermal Control is designed comparing the heat rejection needs of the LEAP habitat with the ISS [21]. For this design it is a valid assumption, because the ISS is already operational for a long period of time and has been proven reliable. Nevertheless, in the future, optimisation can be performed.

Passive Radiation Protection

The program used to model particle interaction caused by radiation is SPENVIS. SPENVIS is a SPace ENVironment Information System developed by ESA, which is based on the GEANT4 program made by CERN. GEANT4 is written in C++ and requires knowledge in that programming language to use it properly. SPENVIS offers a more user-friendly interface, which is easier to use. The GEANT4 program has been tested by other institutes and is a validated program[1]. SPENVIS is also deemed validated, since it is a software used by ESA. GEANT4 is able to simulate particle interaction due to radiation, to calculate the total ionising dose different phenomenons, such as GCR and SPEs. However, the program has its limitations which leads to the following assumptions:

- The lunar habitat is simulated as a spacecraft with the same orbital characteristics as the Moon.
- The lunar habitat is subjected to omnidirectional radiation.
- The radiation originating from the lunar surface is omitted.
- The lunar habitat is simulated as a sphere with a similar volume as the designed habitat.

First of all, since the lunar habitat will be on the Moon, it will make the same trajectory as the Moon, therefore the first assumption is valid. Secondly, the lunar habitat will in reality experience radiation from all sides. However, the habitat will experience less radiation from the side that is exposed to the lunar surface, since GCR and SPEs will be blocked by the Moon. It is thus valid to simulate the radiation source as omnidirectional even though not all radiation levels are the same on all sides. The radiation from the lunar surface is negligible compared to the GCR and SPEs and is therefore omitted in the radiation analysis. Finally, simulating the habitat as a sphere with equal volume is the best approach for the habitat. The other option would be a simulation of a planar slab. The choice of a sphere is therefore justified as it produces the best result for this analysis. The effects and amount of deterioration of material due to radiation is not taken into account due to the limited information and knowledge about that subject. The radiation analysis is done for GCR and SPE radiation. GEANT4 requires models that predict the specific type of radiation and then runs a Monte-Carlo simulation to predict the radiation

[1]http://geant4.cern.ch/results/results.shtml [Cited: 21-06-2017]

levels. The model used for SPE radiation is the ESP model by Xapsos [56]. The ESP model is preferred over other SPE predicting models since it is the most recent model and includes three full solar cycles in the statistics. The model also takes into account worst-case scenarios. The ESP model has been published in an official NASA document and is therefore deemed valid. However, the radiation levels from SPEs are higher than expected, according to [57]. Further research must be done to verify the use of the SPENVIS program. For the GCR, the ISO-15390 model is used to predict the radiation fluxes, which has been developed by the Moscow State University and predicts the fluxes for protons and nuclei. The model is assumed to be valid and verified since it is an international standard. The results are validated by comparing the results with a study from R.A. Mewaldt [4]. A table of GCR levels in an unshielded environment is given in Table 13.1.

Table 13.1: Radiation levels due to GCR in deep space.

Period	Modulation Level [MV]	Unshielded Dose [cGy/yr]	Unshielded Dose Equivalent [cSv/yr]	Shielded Dose Equivalent [cSv/yr]
Solar max.	925	6	39	27
Solar min.	352	16	88	50
Est. 1954	230	19	109	62
Est 1890	100	30	147	83

The second column of data in Table 13.1 shows the GCR levels for an unshielded spacecraft in deep space. The generated ionising dose due to GCR for the lunar habitat layers is in the same order of magnitude with the solar maximum and minimum data. However, there are some differences. The generated data gives dose levels ranging from 6 to 11rad/yr. While the solar minima and maxima are between 6 to 16rad/yr. The differences between the generated data and the data from the study are caused by interpolation of a statistical model. The generated data is extrapolated from measured data to predict the GCR in 2035, while the data from Table 13.1 is from measurements. It can also be noted that the GCR over time has decreased: in 1890 the unshielded ionising dose was 30cGy/yr, while in 1954 it was only 19cGy/yr. Furthermore, the generated data accumulates all ionising doses over a year, while the data from the study only gives a dose rate per event. The program is validated by comparing these values. Although the analysed total ionising dose is lower than those from the study, it can be explained by the trend of decreasing GCR levels over the past years. The data used to predict GCR levels is based on rather old statistical data. To get a better approximation of the ionising dose in 2035, more research should be put into radiation activity in space. The generated ionising dose data for SPEs is found using the same procedure as for GCR, but with the ESP model from Xapsos.

Active Radiation Protection

The overall active radiation program makes use of a few assumptions:

1. **The SPR consists only of protons.** It is true that the SPR mostly consists of protons, which is why it is also sometimes referred to as ”proton storm”[2]. It also contains HZE-particles and helium atoms. These particles have a higher charge than protons, so possibly the magnetic field designed for protons can still be successful in shielding from these particles.
2. **Any secondary radiation produced will not enter the habitat.** However, the alpha particles in the SPR are going to cause secondary radiation even if they are deflected by the shield. The secondary radiation is in the form of neutrons. Neutrons, unless polarised, are not affected by a magnetic field. How much of these neutrons are generated and how much of that will enter the habitat should be verified to validate this program. For the sake of this design process, the program is validated for now and neutron research is strongly recommended for future design.
3. **The magnetic fields of neighbouring coils do not influence each other.** The influence of neighbouring coils would increase the magnetic field, which would ease the deflection.
4. **The protons of the SPR have not increased in mass before they enter the influence of the magnetic field of the habitat.** If the protons have increased in mass the initial speed will

[2]https://umbra.nascom.nasa.gov/SEP/ [Cited: 24-06-2017]

be lower, this would mean a smaller velocity increment is required to deflect it. However, a slower particle will also induce a smaller force to deflect it.

5. **The SPR enter the habitat wall perpendicular to the surface.** The magnetic field is also more effective if the particles arrive perpendicular to the field lines. However, if they arrive at an angle the particles will require less deflection.
6. **The deflection of the SPR is instantaneous.** The deflection won't be instantaneous but more gradual. The motion of the particles will be slightly curved. This will influence the amount of secondary radiation because the path through the wall is not the shortest one, which enables more matter to react.
7. **The magnetic field inside a coil is constant.** The magnetic field inside a coil has peaks of higher strength and lower strength. These peaks will make the deflection more abrupt than a constant field.

Based on literature it seems that the values found using the stated calculations for the mass and power are unrealistically low [58]. It is recommended to run the program with tested electromagnets for shielding purposes to detect unresolved discrepancies in the code. Furthermore, the active shielding was designed to counteract the large equivalent dose for SPE radiation. If further research of the SPENVIS program finds a reduced equivalent dose, this type of shielding might not be needed to have a feasible habitat design.
The magnetic field strength required is calculated using equations of the Concept of Modern Physics book [19]. The program was checked by printing separate outputs and performing sanity checks on the changing of the outputs with respect to changing input values. This showed a mistake in the momentum calculation. The converted equations were given a second opinion. A mistake in the conversion of the momentum was spotted and resolved, after which the program was successfully verified. The coil properties were determined using Python code from an external source[3]. This code is based on the equations for the magnetic field strength of a thin shell solenoid. The calculations and the outputs of the program were checked in the same way as before and found to be correct. Futher, the harmful magnetic field strength penetration radius was calculated also using code from an external source[4]. This code was based on an equation from the following citation: [59]. This code was again checked by sanity checks of changing output and the calculations were checked by multiple people. No discrepancies were found, so the code is verified.
The calculation of the mass and power mostly relied on the determination of a number of coils in the shield. This calculation was checked by running the program that takes the semi-major and -minor axis as inputs and taking the special case in which these two axes are of equal size. The output was then checked with a hand calculation of this case. This turned out to be correct. Then a sanity check was performed on the output in case of a semi-major axis being bigger than the semi-minor axis. It was seen that the angle between the points increases when approaching the top of the ellipse, which was expected. Finally, the thermal properties determination consisted of some simple calculations for the surface area, the temperature, and the thermal flux. For verification, these calculations' outputs underwent a sanity check by a different person as well.

Lander Mass & Volume Determination

In the design of the lander, two different approaches are used. For the high-speed deceleration, the ideal rocket equation is used which only considers changes in velocity giving the mass ratio needed for this delta V, provided the engine properties are given. The Matlab program which uses Tsiolkovsky rocket equation is therefore not always applicable or does not depict accurately the situation as gravitational and centripetal forces play a role in the deceleration. For the second stage on the other hand fewer assumptions are made. The only assumption here is the mass flow, which was derived proportionally to the throttle. This is not necessarily true, but as the mass flow of the SuperDraco engines are not given for the different thrust settings this was the only method which is valid. In order for this to be more precise one needs to determine the mass flow of the engines based on the 50% throttle that is used in the last descent phase.

[3]/nbviewer.jupyter.org/github/tiggerntatie/emagnet-py/blob/master/solenoids/solenoid.ipynb [Cited: 20-06-2017]
[4]http://nbviewer.jupyter.org/github/tiggerntatie/emagnet-py/blob/master/offaxis/off_axis_loop.ipynb [Cited: 21-06-2017]

For the determination of the high-speed deceleration, a different model was set up to verify the program used for the sizing of the first stage of the lander, which considers the centripetal force the spacecraft experience due to its tangential velocity and gravitational pull the Moon exerts on the spacecraft. The results of this model are then compared to the results of the previous calculation. The differences in the model are the forces that are in play and the fuel amount is determined using the fuel flow. In this case, the fuel flow is known as the RL10 engines operate at full capacity and the fuel flow for this setting is known to be of 24.1kg/s. This model starts at the point where the spacecraft is slowed down to the speed and altitude required for the precision landing, with a mass equal to that determined by the previous model. Initially, it accelerates the spacecraft parallel to the lunar surface, balancing the centrifugal, gravitation and part of the thrust normal to The Moon, using the remaining thrust for the acceleration. Once the escape centrifugal force is greater than the gravitational pull all the thrust is used to accelerate the spacecraft parallel to the lunar surface. The spacecraft is accelerated until it reaches a resultant speed of 2953m/s, which corresponds to the TLI speed.

The results of the program can be seen in the figures presented above. In the model for the SLS payload mainly two differences can be noted. First, the amount of time the engine requires to perform the speed change differs around 70s compared section 10.2. This corresponds to about 10% of the total time needed thus is quite significant. The other difference is the mass of the spacecraft. Using the model presented in section 10.2 a fuel mass of 21.0 tonnes is needed, while in the verification model only 18.6 tonnes are needed if the 1.1 safety factor is included. The safety factor is not included in the plots depicted above. This thus results in a difference of 2.4 tonnes. If one calculates the fuel deficit, using the difference in time and the fuel flow, there is still a difference of 1.7 tonnes so there are different factors that differ in the models. In this case, the percentile difference is 11% which once again is close to the 10% contingency in this phase of the design. When looking at the Falcon Heavy payload, also here the verification model requires less time to decelerate the payload, around 48s, which corresponds to a 15% change while the mass has a difference of around 8.5%, including the 1.1 safety factor, which again falls within the margin for this phase of the project. The plots of the Falcon Heavy module are not included as they resemble the ones of the Space Launch System. In order to get a final value for the lander size and mass, a more precise model has to be set up which considers the most efficient trajectory and then computes all the burn times the lander needs to execute in order for the payload to perform the desired trajectory.

Communication and Data Handling System

The characteristics of the communication and data handling system were mostly taken from reference systems used on the ISS, Earth and other space missions and if needed adapted to the conditions of the habitat. These adaptations are explained in the respective chapter about the system.

The most extensive calculations have been made for the UHF covered operational area for lunar exploration. Therefore it was assumed that the Moon is a perfectly smooth sphere. This assumption is deemed valid for the landing location that was selected since it is located in the Mare Tranquillitatis which is a more or less flat field, with the only deviations in altitude being large rocks and impact craters. It was also assumed that the general lunar radius of 1737km could be used, even though the Mare Tranquillitatis has an elevation of -2km. The calculations have been performed with both radii but the variations in antenna height were in the order of millimetres, so the Moon radius of 1737km was kept for the sake of compliance with the knowledge of the general public. The calculations itself were done in Excel and verified by performing manual derivations of the trigonometric formulas used and by replicating the calculations in Wolfram Alpha, which yielded the same results.

Electrical Power System

The power system was sized and designed by using elementary formulas and equations. To ensure their outcome, consistent verification was performed while making these calculations. With every formula, a unity check was performed validated with a sanity check. When the outcome was questionable, research within the calculations was done to pinpoint the issue. Sometimes, a calculation error occurred, sometimes a reasoning error due to invalid assumptions. The main consistent assumptions that were made were concerning the use of existing products and materials like fuel cells and electrolyzer cells since Proton Exchange Membrane technology has not been applied for lunar missions. Also, efficiencies and parameters were assumed from the average of commercially available products. It can be expected that those products will decrease in mass when they will specifically be designed for.

Interior Lay-out

For quantitative bookkeeping of the interior and its mass, a program was written in Excel. In this way, the total mass of the interior could be calculated. To determine the amount, dimensions and mass of each piece of furniture, different sources of information have been used, and a few assumptions have been made. Historical layouts of space habitats assume a micro-gravity environment. The presence of gravity on the Moon means a more Earth-like design philosophy had to be used. This is why a combination of historical space layouts and low area Earth layouts has been used [60]. For the estimation of the mass, certain assumptions have been made. For example, the average mass of pieces of furniture here on Earth has been used but corrected with the average density of PEEK to estimate their weight in case they are 3D printed. The average density is taken from the CES Edupack software, which is a verified and valid source. After the Excel program was completed, unit calculations were performed by hand. This way, the interior design calculations were verified.

14
Requirements Compliance Checks

Table 14.1: Requirements compliance matrix, including requirement validation comments.

Requirements			Compliances			
Code	Item	Requirement	Value	Margin	Com-pliant	Comments
MR-LO	Location	The lunar habitat shall be located on the near side of the Moon	yes		yes	Apollo 11 landing site
MR-LO-01	Location	The ground shall be able to endure the loads of the lunar base without slipping or sinking	yes		yes	
MR-LO-02	Location	The landing site shall not contain any debris or rocks, that are too big to be moved by the infrastructural preparation robots	yes		yes	
MR-LO -03	Location	The landing site shall be distanced <tbd>m from mountains to prevent potential dust avalanches affecting the base	yes		yes	Needs further research for bases near mountains
MR-LO-04	Location	The surface shall not contain any vast abrupt changes in composition, that are likely to cause shallow quakes	yes		yes	
MR-LO-05	Location	The location shall provide a continuous square surface area of at least $64km^2$ which is accessible by lunar vehicle	yes		yes	See rover specifics
MR-AS	4 astronauts	The habitat shall host up to 4 astronauts	yes		yes	
MR-MT	1 year	The habitat shall sustain the astronauts' life for at least 1 year at a time taking into account multiple resupplies	yes		yes	
MR-TL	10 year lifetime	The habitat shall be operational for a period of at least 10 years	yes		yes	
MR-LE	Legislation	The habitat shall comply with the international legislation regarding Moon exploration recognised by ESA members	yes		yes	
MR-SA	Survival	The astronauts shall be able to return safely to Earth at all times	yes		yes	Ascend vehicle available

Code	Item	Requirement	Value	Margin	Compliant	Comments
MR-EE	Enter and Exit Habitat	The astronauts shall be able to enter and exit the habitat	yes		yes	Airlocks as entrance and exit
MR-LB	Lunar Base	The habitat shall be able to integrate into a lunar base	yes		yes	Docking systems present
MR-ST	Stationment	The habitat shall be stationary	yes		yes	
MR-OU	Outfit	The habitat shall allow the astronauts to wear the same or simpler outfit than inside the ISS	yes		yes	Atmospheric conditions allow for casual outfits
MR-LS	Living space	The habitat shall have a usable floor area of $100m^2$	$130m^2$		yes	
MR-TR	Transport	The habitat and the astronauts shall be transported to the Moon with transportation methods with a TRL of 3 in june 2017	yes		yes	
MR-TR-01	Frequency	The habitat shall be resupplied at least once every year	yes		yes	
MR-TR-02	Payload mass	The mass of the payload in payload configuration should not exceed the useful mass capacity of the chosen launch vehicle	yes		yes	Rephrased when launch vehicle was selected first
MR-TR-03	Payload size	The volume of the payload in payload configuration should not exceed the useful volume capacity of the chosen launch vehicle	yes		yes	Rephrased when launch vehicle was selected first
MR-TR-04	Precision	The landing precision on the Moon shall be 2km	6m		yes	
MR-TR-05	Landing loads	The landing loads shall not exceed launch loads	yes		yes	
MR-TR-06	Return mission	The transportation system shall retrieve the astronauts from the Moon after 1 year	yes		yes	
SYS-ST	Structural system	The structural system shall ensure integrity of the habitat during its lifetime	yes		yes	
SYS-ST-01	Provide safety	The structural system shall protect the astronauts and the interior	yes		yes	
SYS-ST-01-01	Environment	The structural system shall protect the astronauts and interior from the Lunar environment	yes		yes	
SYS-ST-01-02	Load cases	The structural system shall withstand all load cases	yes		yes	

Code	Item	Requirement	Value	Margin	Com-pliant	Comments
SYS-ST-02	Docking	The structural system shall need to be docked autonomously after landing	yes		yes	Replaced ”assem-bling” by ”docking”
SYS-ST-02-01	Positioning	The modules with active docking shall have a positioning error of less than 10cm	yes		yes	Specific value added
SYS-ST-02-02	Tools needed	The structural system shall include a defined set of tools for assembly	yes		yes	
SYS-ST-02-03	Assembly manual	The structural system shall include an assembly manual	yes		yes	
SYS-EP	Environmen-tal Protection System	The habitat shall protect the astronauts and internal systems against the environmental conditions during the mission	yes		yes	
SYS-EP-01	Radiation protection	The radiation protection system shall ensure that the astronauts receive a radiation dose of less than 0.5Sv during one year	0.44Sv	±0.06Sv	yes	
SYS-EP-02	Meteorite protection	The meteorite protection system shall protect the habitat and the astronauts from micrometeorites and secondary impact debris	yes		yes	
SYS-EP-02-01	Projectile design velocity	The meteorite protection system shall be designed for an impact velocity of 45 km/s	45 km/s		yes	Specific value added
SYS-EP-02-02	Shielding reliability	Each module of the habitat shall have a meteorite shielding reliability of 0.998	0.998		yes	Specific value added
SYS-EP-03	Thermal protection	The environmental protection system shall ensure all subsystems stay within their operating temperature ranges while operating	yes		yes	
SYS-EP-04	Atmospheric Conditions	The habitat shall maintain human life supporting atmospheric conditions	yes		yes	
SYS-EP-04-01	Oxygen	The environmental protection system shall sustain an oxygen level of 20% ±1%	20%		yes	
SYS-EP-04-02	Pressure	The environmental protection system shall sustain an internal pressure of 1 atmosphere	1 atm		yes	
SYS-EP-04-03	Humidity	The environmental protection system shall sustain a humidity level between 40% and 60%	40%-60%		yes	
SYS-PW	Power System	The power system shall provide the power for the habitat and its subsystems at all time	yes		yes	

Code	Item	Requirement	Value	Margin	Compliant	Comments
SYS-PW-01	Generation	The power system shall have a means of generating electrical energy	yes		yes	
SYS-PW-01-01	Sustainability	The electrical energy shall be generated without influencing the lunar environment	yes		yes	
SYS-PW-01-02	Capacity	The minimal power generation during the lunar day shall be 37.74kW	105.36 kW		yes	New values derived from system needs
SYS-PW-01-03	Peak performance	The power generation system shall be able to support peak demands of 90.09 kW	105.36 kW		yes	New values derived from system needs
SYS-PW-02	Storage	The power system shall have a means of storing electrical energy	yes		yes	
SYS-PW-02-01	Capacity	The energy storage shall have a capacity of at least 7800kWh	9361.54 kWh		yes	New values derived from system needs
SYS-PW-02-02	Peak performance	The fuel cells shall be able to support peak demands of 90.09kW	91kW		yes	New values derived from system needs
SYS-BA	Bioastronautic System	The habitat shall support and sustain human life	yes		yes	
SYS-BA-01	Medical support	The bioastronautic system shall feature medical supplies	yes		yes	Medical kits are provided
SYS-BA-01-01	Medical care supplies	The habitat shall include at least the same set of medical care supplies available at the ISS	yes		yes	
SYS-BA-01-02	Medical equipment	The habitat shall provide at least the same medical equipment as available on the ISS	yes		yes	
SYS-BA-02	Physical Excercise	The bioastronautic system will adverse physiological effects due to microgravity	yes		yes	Exercise equipment provided
SYS-BA-03	Sleeping facilities	The bioastronautic system shall enable the astronauts to sleep	yes		yes	
SYS-BA-04	Nutrition	The bioastronautic system shall provide nutrition to the astronauts	yes		yes	

Code	Item	Requirement	Value	Margin	Com-pliant	Comments
SYS-BA-04-01	Nutrient re-quirements	During the assembly of the lunar habitat, planned menus must meet the nutritional requirements in JSC 28038, Nutritional Requirements for International Space Station Missions Up To 360 Days After assembly complete, the menus shall meet nutritional requirements	yes		yes	
SYS-BA-04-02	Taste	A preflight testing of all individual food items shall be made available to each crewmember to obtain their recommendation for development of an initial inflight menu	yes		yes	
SYS-BA-05	Mental well-being	The bioastronautic system shall ensure mental well-being of the astronauts	yes		yes	
SYS-BA-05-01	Living environment	The habitat shall have a suitable living environment	yes		yes	
SYS-BA-05-02	Entertain-ment	An entertainment system shall be available to the astronauts	yes		yes	
SYS-BA-05-03	Leisure	The astronauts shall have leisure activities	yes		yes	
SYS-BA-06	Sanitary system	The habitat shall house sufficient sanitary systems	yes		yes	
SYS-BA-07	Waste processing	The habitat shall include a waste processing system	yes		yes	
SYS-COM	Communica-tion System	The habitat shall have a communications system	yes		yes	
SYS-COM-01	Up and Down Link	The communication system shall enable an uplink and downlink to Earth	yes		yes	
SYS-COM-01-01	Uplink speed	The communication system shall provide an uplink bitrate of at least 3 Mbps	20 Mbps		yes	
SYS-COM-01-02	Downlink speed	The communication system shall provide a downlink bitrate of at least 100 Mbps	622 Mbps		yes	
SYS-COM-02	Contact time	The habitat shall have continuous contact with Earth	yes		yes	
SYS-COM-03	Local com-munication	The habitat shall have a lunar communication system for lunar exploration	yes		yes	Added during system research

Code	Item	Requirement	Value	Margin	Compliant	Comments
SYS-COM-03-01	Astronauts within the exploration area will be able to have continuous communication with the habitat	yes		yes	Added during system research	

15
LEAP Missions

Following the end of this DSE project, LEAP will be officially initiated. In section 15.1, the chosen location for this mission is presented and elaborated on. The designed lunar habitat will be part of a complete lunar village, of which the layout will be shown in the same section. Following this, the years between the DSE and the operational phase of the lunar habitat will be discussed. The development logic and flow is described in section 15.2, accompanied by a flow diagram and a Gantt chart. Hereafter, the production plan is briefly described in section 15.3. Finally, the verification and validation techniques to be performed during the LEAP mission are proposed in section 15.4.

15.1. Lunar Village Integration

The habitat as described in the previous chapters is intended as the first part of a bigger lunar village. During its operational lifetime, several modules will be added to expand the village and add value to the mission. Important for this expansion is the village location, of which the selection process is described here. The type of extensions which are proposed will also be discussed, as well as the final layout of the lunar village.

Lunar Village Location

For the lunar village, of which the habitat is the first part, three locations were considered: the Apollo 11 landing site, the Apollo 17 landing site and the lunar south pole. These locations were assessed based on six criteria: the scaling of available maps, the illumination time percentage per year, the cycle characteristics of the illuminations, the communication time percentage per year, the qualitative useful area and dust content. These criteria are used in a trade-off in Table 15.1.

Table 15.1: Trade-off of considered locations. Green: Excellent; exceeds requirements. Blue: Good; meets requirements. Yellow: Correctable deficiencies. Red: Unacceptable.

Criterion > Option	Map scale	Illumination time [%]	Illumination cycle type	Communication time [%]	Qualitative useful area	Qualitative dust content
Apollo 11	1:25,000 *green*	50 *blue*	14 day lapses *blue*	100 *green*	Large *green*	High *yellow*
Apollo 17	1:1000,000 *blue*	50 *blue*	14 day lapses *blue*	100 *green*	Medium *blue*	Medium *blue*
South pole	1:5000,000 *yellow*	70 *green*	summer-winter *yellow*	50 *yellow*	Small *yellow*	Unknown *yellow*

The most detailed maps can be found for the Apollo 11 landing site, whilst the maps available for the lunar south pole are significantly less detailed[1]. It is preferred to have detailed maps, to be able to determine what the optimal location would be for the habitat. A misjudgement of the situation will have a significant negative result for the mission. An advantage of the lunar south pole is that the location is illuminated for a longer amount of time than the Apollo 11 and 17 landing site. This longer illumination together with the near constant inclination of the Sun with respect to the horizon is beneficial for the thermal and power system [61]. However, the fact the the lunar south pole experiences a high level of illumination summer and a rather low level in winter counters this positive effect with respect to the

[1] http://www.lpi.usra.edu/resources/mapcatalog/usgs/ [Cited:16-06-2017]

Apollo 11 and 17 landing site [61]. Another negative asset to the lunar south pole is that it only has a direct line of sight with Earth half the lunar day time. This will make the communication system a lot more difficult than that of the two Apollo sites [61]. The useful surface area of the location is important for the expansion of the habitat into the lunar village with all the required resupplies. It is impossible to determine an exact number for this area, however, it is visible in the maps available that the Apollo 11 provides the largest semi-flat useful area, then the Apollo 17, which has mountains surrounding it and lastly the south pole which is on a mountain[1].

The final criterion considered is the qualitative dust content. The dust clouds formed during landing have shown that the Apollo 11 landing site has more dust than the Apollo 17. This has some implications for the distance between the habitat and the landing site for resupply missions. How this compares to the amount of dust on the lunar south pole is unknown. Based on these criteria the Apollo 11 landing site compared to the other options would be the most optimal location for the habitat. Therefore it was decided to plan for the habitat to be in the vicinity of the Apollo 11 landing site.

Auxiliary Modules

As explained in chapter 1, the Moon is a point of interest to gain experience in living on another celestial body. Therefore, this report discusses the detailed design of a lunar habitat which will be the starting point for gaining this experience. In addition to the habitat, the village shall also facilitate research on the Moon.

This Moon research shall consist of studying the effect of reduced gravity on Earth organisms, studying the soil properties in the vicinity of the habitat and exploring the option of mining lunar resources. One of these lunar resources would be the lunar ice that is expected to be present deep in the lunar soil. If the presence of ice could be confirmed, the option of mining and utilising it for generating rocket fuel should be studied. To enable this research, the habitat will be extended during its ten years life time with multiple auxiliary modules to form the lunar village. At this point it is proposed to consider to at least extend the habitat with a science lab, a greenhouse, a garage and an extra storage space.

In general, all auxiliary modules should comply to the following specification unless extensive research suggests otherwise:

- Provide a reliable and safe working environment.
- Able to dock to habitat.
- Transportable by MLR.
- Able to withstand launch loads and transportable by existing launch vehicle(s).
- Fully operational before 2040.

Science Lab

The science lab extension shall provide a work space for the astronauts to perform research on the Moon. Carefully selected research, that is proposed by the parties funding the lunar mission will be undertaken in this facility. This with the exception of long-term plant growth and photosynthesis research in lunar gravity, which is to be done in the lunar greenhouse. Below, some specifications are defined for the science lab in particular:

- Facilitate typical indoor research to be expected on the Moon.
- Environmental conditions equal to inside habitat.
- Workable volume of at least 70m^3[2].

Greenhouse

As mentioned in the previous section, the greenhouse extension shall enable long-term research of the influence of lunar gravity on plant growth and their photosynthesis. Plants are of special interest for future manned missions to other celestial bodies, since they can provide food and generate O_2 via photosynthesis by using CO_2, which is one of the principal contaminants of the habitat atmosphere.

The research done in the lunar greenhouse shall aid the development of a bioregenerative life support system for future manned space missions[62]. Below some specifications are defined for the greenhouse in particular:

[2]http://www.russianspaceweb.com/iss_fgb2.html [Cited: 17-06-2017]

- Facilitate bioregenerative life support system supporting research.
- Environmental conditions equal to inside habitat.
- Growing area of at least 32m^2 [62].

Garage
To study soil properties, explore the option of mining lunar resources or test equipment on the Moon, the astronauts will have to venture outside of the lunar village. Specially equipped rovers will help them cover large distances. These rovers will have their own protection system that will protect their systems and the astronauts inside from the lunar environment. However, while these rovers are maintained, or certain instruments are switched, the protection system will not be able to protect as effectively. Therefore, to prevent system damage in these cases, a garage will be added to the habitat. Furthermore, the habitat and its extensions will be powered by Power Trucks. ThesePower Rangers will also require checks and maintenance, which are preferably performed in the garage. Below, some specifications are defined for the garage in particular:

- Protect rover systems against micrometeorites, radiation and lunar dust.
- Direct docking of the SEV to habitat inside the Garage. (see section 11.2)
- Continuous housing for the MLR, SEV and LSR. (see section 11.2)
- Incidental room for at least one Power Ranger at a time.

Storage
In the design of the habitat, a large storage space is included in one of the inflatable modules. While the lunar village is extended with auxiliary modules, the amount of storage required will also increase. Therefore, a storage module shall be added to accommodate all the extra storage required. Below, some specifications are defined for the storage in particular:

- Environmental conditions equal to inside habitat.
- Storage volume of at least 70m^3[3].

Preliminary Village Layout

Figure 15.1 features a preliminary layout of the lunar village with the habitat modules depicted in grey and the auxiliary modules in white. The layout is based on the daily routine of the astronauts and possible escape routes in case of an emergency. During the first few years of its operational life, the habitat will function on its own without extensions. During these years, the habitat will have four airlock exits in total(depicted in Figure 15.1 with a small white circle near the edge of the module) to the lunar outside. Two of these will be in the hard-shell module (The Shell) and one exit in each inflatable module (The Hive and The Nest).

It is preferred to have two airlock exits in the hard-shell module, because each airlock exit can only harbour two astronauts at a time. In case of an emergency it is safer, if the crew of four astronauts can exit the habitat simultaneously in proximity of each other. In Figure 15.1 the garage is connected to one of the original airlock exits. In view of safety, this is even better, because the astronauts exiting at the garage can immediately get into the SEV, pick up the astronauts at the other airlock exit and drive to safety.
The other auxiliary modules mentioned in Table 15.1 are connected to the inflatable module which is mostly reserved for storage. They were intentionally not connected to the sleeping module, as the modules might produce sounds that could disturb the astronauts while sleeping. Furthermore it is not favourable to have to move through the sleeping quarters every time the astronauts have to go to work. To avoid creating one long hallway with airlock exits to the outside only at the ends, which is ill-favoured with respect to safety, another hard-shell module shall be added. This hard-shell module (aux center) will be a connecting module for the science lab, the greenhouse and the extra storage module, which will each have their own airlock exit. The habitat will be powered by Power Rangers. These Power Rangers will be positioned close to the habitat, but in such a way that the shadow of the village modules will not

[3]https://www.space.com/10992-space-station-extra-storage-room.html [Cited: 17-06-2017]

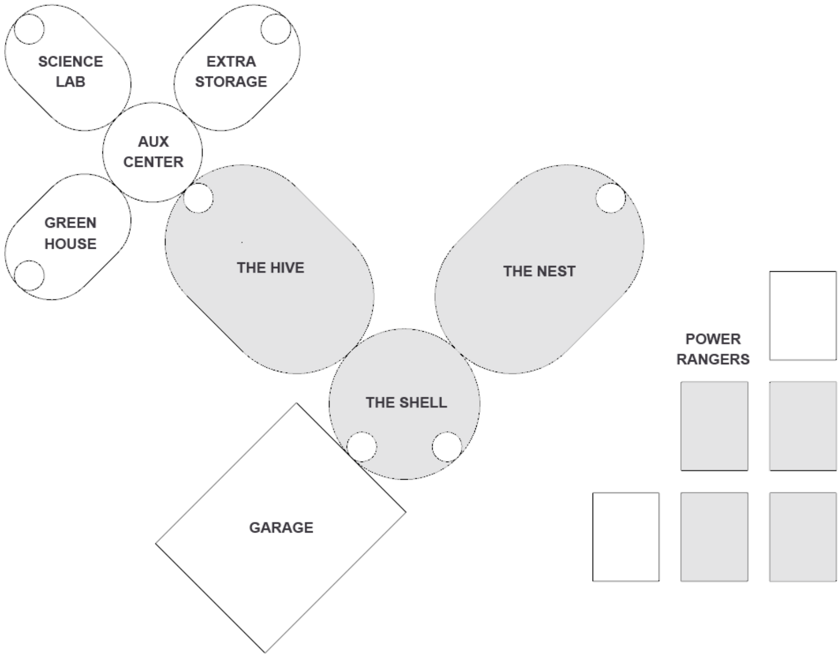


Figure 15.1: Preliminary layout lunar village. Gray parts are existing habitat design. White parts are auxiliary modules to be added. Ground trajectory Sun parallel to vertical axis.

hit the solar panels mounted on the roofs of the Power Rangers during the lunar illumination time. With each extension of the village, Power Rangers will be added to keep providing sufficient power for the village.

15.2. Project Design & Development Logic

Since the lunar village extension has now been discussed, the complete project design and development logic can be shown and elaborated on. To ensure the success of the LEAP mission, every step in the development process has to be defined in an appropriate way. To be able to perform the mission at all, an organisation has to be set up, which will ensure political, legal and financial support. This organisation will connect all stakeholders for the mission, and provide a platform through which all communication will go.

After the LEAP organisation is set up, the actual design process can start. This detailed design will include the inflatable modules, the hard-shell module, the landers, the rovers, the Power Rangers and the rest of the lunar village. The mission logistics will also be designed in detail during this phase. Concurrently to this, a lunar satellite orbiter will be designed, which will collect specific data about the Moon and map the location in more detail. Also, additional research will be conducted on, among others, the inflatable material, the radiation protection, the precision landing, and the regolith sintering.

Following the phase of detailed design, the actual manufacturing of parts, subsystems and rovers can start. Throughout this phase, the manufactured parts will be tested and verified. The process through which this will happen will be further elaborated in section 15.4. As explained in that section, an Earth version of the lunar habitat will be built to perform verification. In the meantime, a scout mission will be prepared, which will locate the most favourable location in the vicinity of the Apollo 11 location, and place beacons for the precision landing manoeuvres. Following this scouting mission, an infrastructure mission will be prepared which will set up the infrastructure around the location and place the Power Rangers. Concurrently to this, the first crew of astronauts will be selected and trained. The selection and training of later crews will continue for the next decade.

Halfway 2030, the final manufacturing of the lunar habitat will start. This includes performing all non-destructive and non-altering tests and analyses. The first module which will be launched is The Shell, which is planned to happen in March 2034. Once The Shell is stationary at its desired location, the two inflatable modules can be launched. After their landing on the lunar surface, they will be transported to The Shell. There, they will be docked on the hard-shell module, after which they will be inflated. This last step will happen in December 2034, after which the lunar habitat will be operational for the following ten years. A flow diagram of these activities has been constructed in Figures 15.2 and 15.3. In the top right, every activity/milestone contains the amount of months it roughly takes. Next to the flow diagram, a Gantt chart is created based on the development strategy. This Gantt chart brings the expected time slots into perspective, and shows the general time line of the project. The Gantt chart can be seen in Figure 15.4 and Figure 15.5.

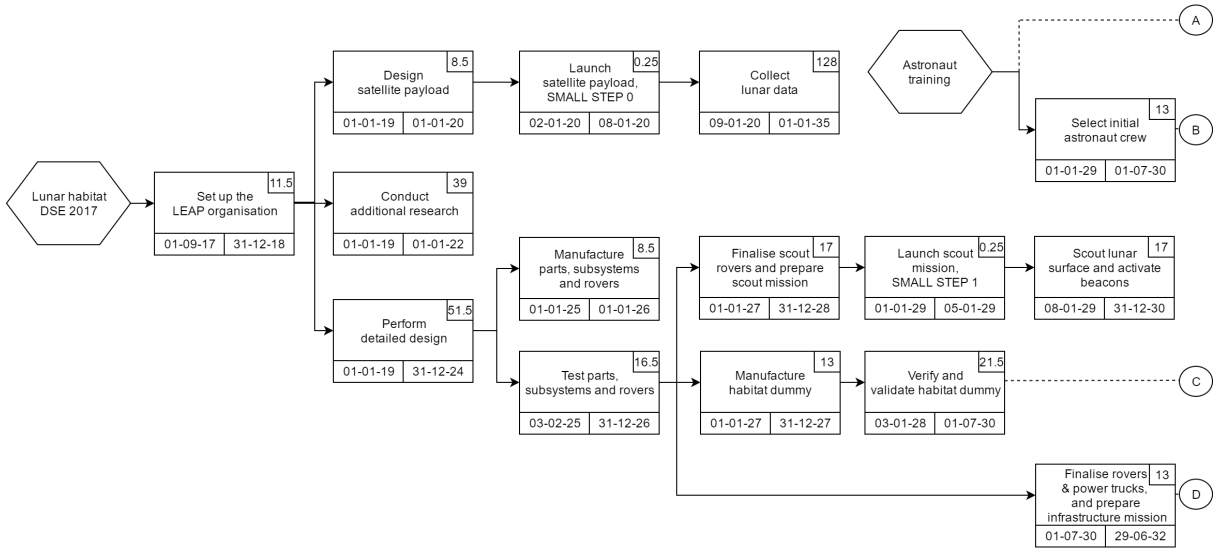


Figure 15.2: Stage 1 of LEAP's mission timeline.

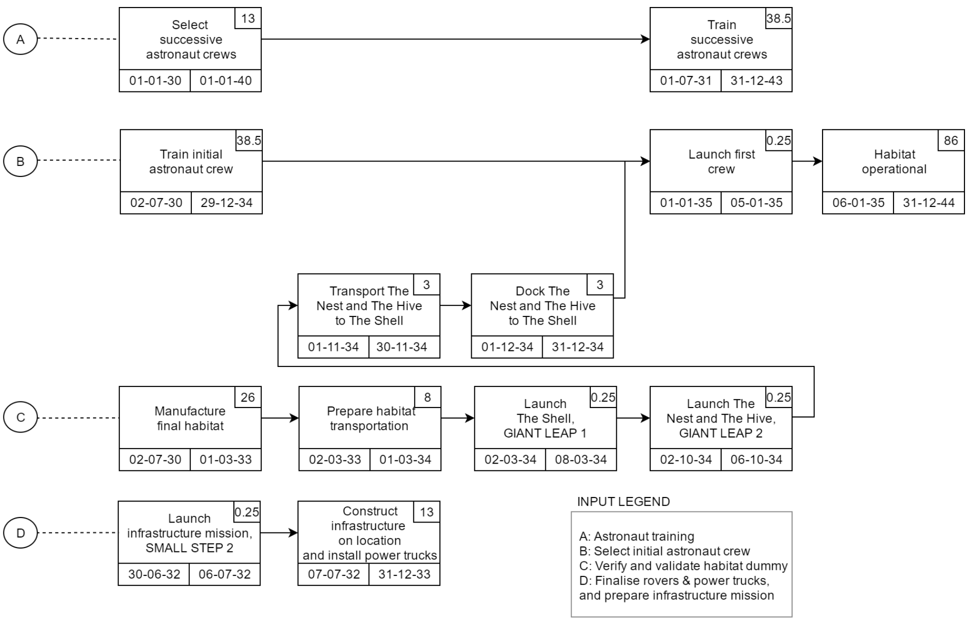


Figure 15.3: Stage 2 of LEAP's mission timeline.

Task Name	Duration	Start	Finish	Predecessors	Resource Names
Set up the LEAP Organisation	347 days	Fri 1-9-17	Mon 31-12-18		Orange
Design Satellite payload	262 days	Tue 1-1-19	Wed 1-1-20	1	Blue
Launch Satellite, SMALL STEP 0	5 days	Thu 2-1-20	Wed 8-1-20	2	Red
Collect lunar data	3908 days	Thu 9-1-20	Mon 1-1-35	3	Red
Conduct additional research	784 days	Tue 1-1-19	Fri 31-12-21	1	Blue
Perform detailed design	1566 days	Tue 1-1-19	Tue 31-12-24	1	Blue
Manufacture parts, subsystems and rovers	262 days	Wed 1-1-25	Thu 1-1-26	6	Blue
Test parts, subsystems and rovers	499 days	Mon 3-2-25	Thu 31-12-26		Blue
Manufacture habitat dummy	261 days	Fri 1-1-27	Fri 31-12-27	8	Blue
Verify and validate habitat dummy	651 days	Mon 3-1-28	Mon 1-7-30	9	Blue
Finalise scout rovers and prepare scout mission	521 days	Fri 1-1-27	Sun 31-12-28	8	Blue
Launch Scout mission, SMALL STEP 1	5 days	Mon 1-1-29	Fri 5-1-29	11	Red
Scout lunar surface and activate beacons	517 days	Mon 8-1-29	Tue 31-12-30	12	Red
Finalise infrastructure rovers and power trucks, and prepare infrastructure mission	522 days	Mon 1-7-30	Tue 29-6-32	8	Blue
Launch infrastructure mission, SMALL STEP 2	5 days	Wed 30-6-32	Tue 6-7-32	14	Red
Construct infrastructure on location and install power trucks	583 days	Wed 7-7-32	Fri 29-9-34	15	Red
Select initial astronaut crew	391 days	Mon 1-1-29	Mon 1-7-30		Green
Train initial astronaut crew	1174 days	Tue 2-7-30	Sun 31-12-34	17	Green
Launch first crew	5 days	Mon 1-1-35	Fri 5-1-35	18;26	Green
Select successive astronaut crews	2609 days	Tue 1-1-30	Sun 1-1-40		Green
Train successive astronaut crews	3263 days	Tue 1-7-31	Thu 31-12-43		Green
Manufacture final habitat	696 days	Tue 2-7-30	Tue 1-3-33	10	Blue
Prepare habitat transportation	261 days	Wed 2-3-33	Wed 1-3-34	22	Blue
Launch The Shell, GIANT LEAP 1	5 days	Thu 2-3-34	Wed 8-3-34	23	Red
Launch The Nest and The Hive, GIANT LEAP 2	5 days	Mon 2-10-34	Fri 6-10-34	16;24;23	Red
Transport The Nest and The Hive to The Shell	22 days	Wed 1-11-34	Thu 30-11-34	25	Red
Dock The Nest and The Hive on The Shell	21 days	Fri 1-12-34	Sun 31-12-34	26	Red
Lunar habitat operational	2610 days	Mon 1-1-35	Sat 31-12-44		Red

Figure 15.4: Mission task schedule with description, duration and dates (Orange = organisational, Blue = Design & Manufacturing, Red = Mission Operations, Green = Manned Mission Related).

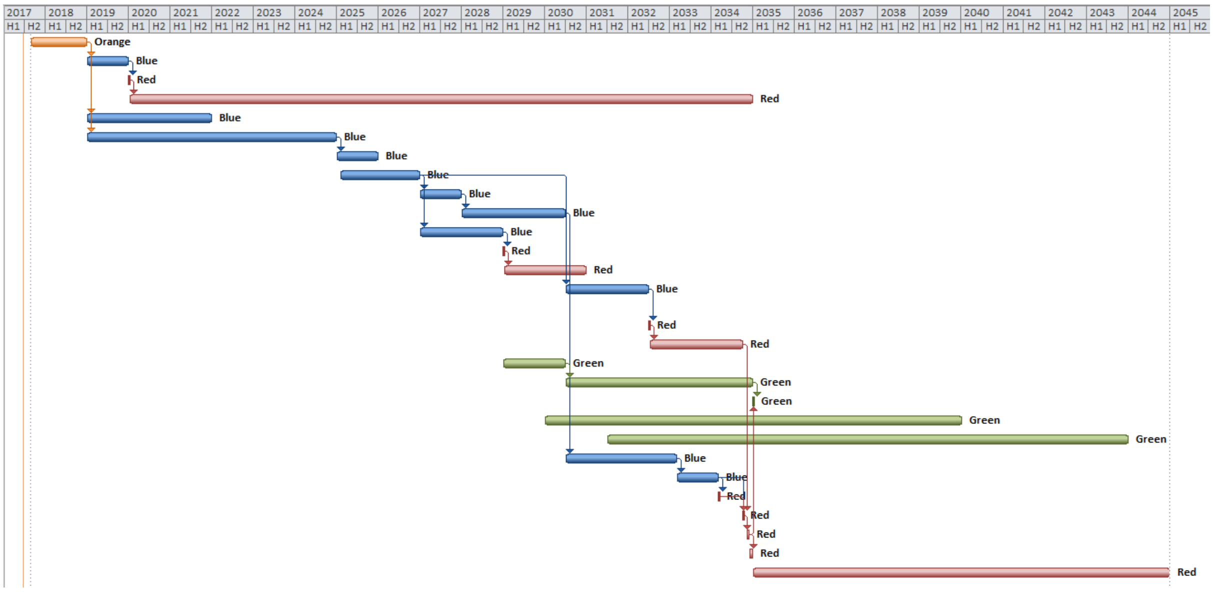


Figure 15.5: Visualisation of task relations and chronological order of Figure 15.4 (Orange = organisational, Blue = Design & Manufacturing, Red = Mission Operations, Green = Manned Mission Related).

15.3. Production Plan

Figure 15.6 shows a very general production plan for the production of the final modules. The process starts off with the manufacturing of the structural components. These will then be assembled together with the airlocks to create the structure of the modules. Further, the inflatable structure has to be folded into the shape of transportation. The remaining systems will be integrated in the hard-shell module and in the busses of the inflatable shell. Lastly, the modules will be connected to their respective lander, stored in the launcher and sent towards the Moon. The production plan needs to be specified with respect to delivery and assembly schedule in order to ensure a smooth production phase.

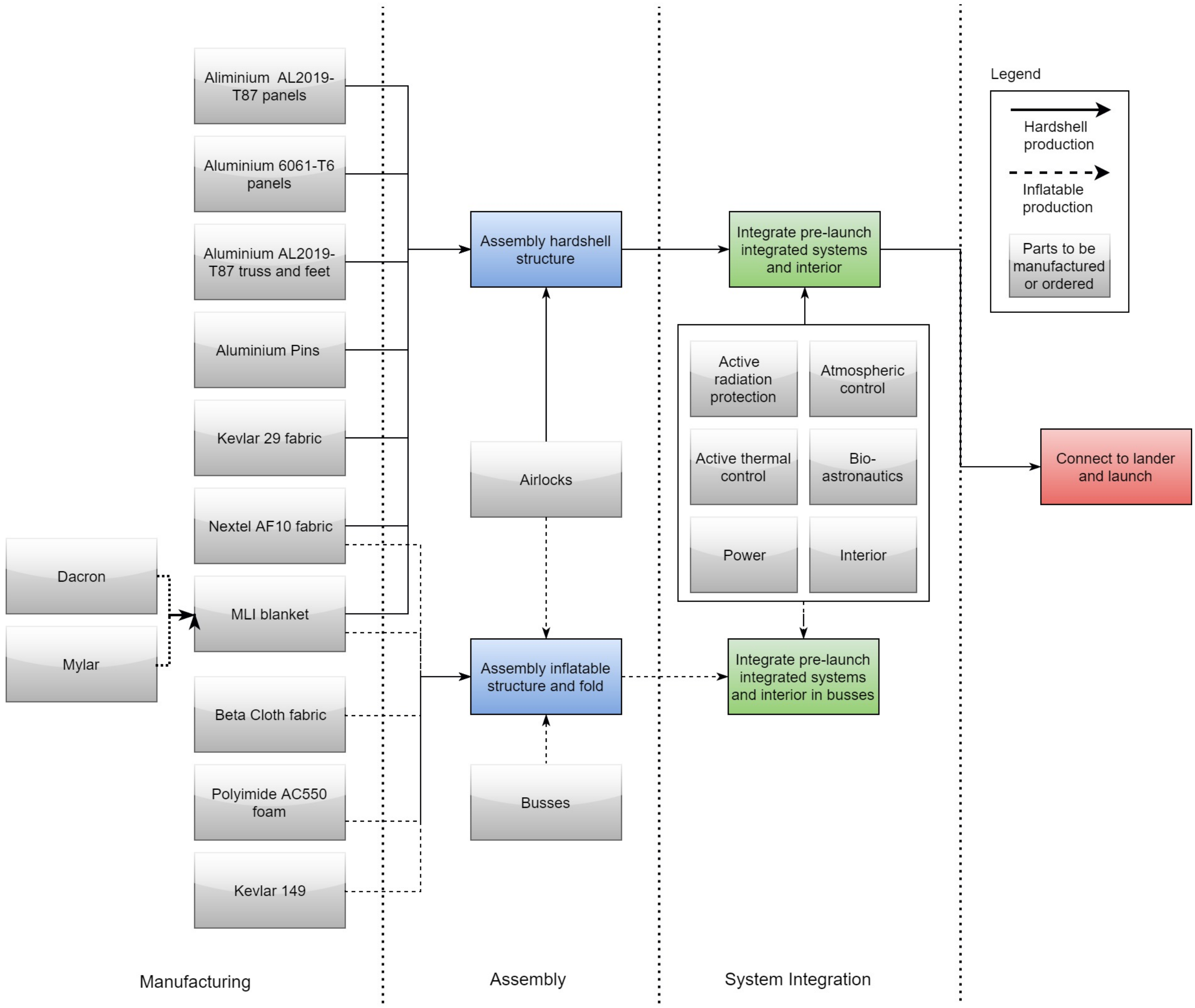


Figure 15.6: Rough production plan.

15.4. Mission Plan Verification and Validation

Once the design of the lunar habitat is completed, verified and validated, the mission can be started and the actual lunar habitat can be build, as described in section 15.2 and section 15.3. This process will also contain a full verification and validation, which will be elaborated in this section.

Mission Plan Verification

During the execution of LEAP, multiple aspects have to be verified continuously, to ensure the success of the mission.

Firstly, the transportation modes will have to be verified. This means that the launch vehicles, which will be used during the mission, have to be tested and proven to be working through demonstration. Furthermore, the final payload packages have to be inspected and made sure to have the correct dimensions, such that they fit inside the payload fairing of the launch vehicles. Additionally, the landing accuracy of the landers using the beacon system has to be verified. This is done through inspection, both by sensors on site and potentially using data from lunar orbiting satellites.

The materials with which the lunar habitat will be build, will also have to be verified. This means that every batch of parts or bulk material has to be analysed, and tests have to be performed on some elements of that batch to ensure the high quality. Specifically for the LEAP mission, caution should be exercised that all parts and materials are compatible with the lunar environment, which means that the tests have to be performed in as much as possible similar conditions to the lunar conditions. Various vacuum chambers and other test facilities which simulate these harsh lunar conditions can be used for this process.

Every subsystem which is built separately, should be inspected whether it conforms to the design (in terms of dimensions, mass etc.), but also be tested whether it performs its task in the desired way. This means that each subsystem has to be built a multitude of times, so that the actual subsystems which will be sent to the lunar surface will not be damaged or altered during testing. Once all subsystems are verified separately through means of unit tests, a dummy habitat will be built here on Earth. Through this integration of the subsystems, a system test can be performed: It can be verified whether all subsystems work appropriately together. This lunar habitat will be used throughout the entire duration of the mission, so that ground control can always simulate what the astronauts in the actual lunar habitat on the surface of the Moon will experience. This will be further elaborated in the mission plan validation.

Auxiliary systems for the lunar habitat will also have to be verified before usage on the lunar surface. Systems such as the automated rovers will be built on Earth, and tested under conditions simulated to be similar to the lunar surface. Like the habitat itself including the subsystems, these auxiliary systems will also have full working versions on the Earth, so that tests can be performed throughout the mission operation, and specific situations can be simulated for these auxiliary systems. This way it will be ensured that they will work effectively on the Moon, and can operate in the lunar environment.

Finally, to be able to inspect and test the lunar habitat once it is on the lunar surface, and afterwards continuously while it is operational, multiple sensors will be installed in and on the habitat. This way, the different subsystems can be monitored, as well as the whole habitat functioning like a system. Every system independently has to receive a safety confirmation, together with the complete lunar habitat. This means that a safe living environment can be ensured before the astronauts arrive on the lunar surface, which is a very important measure in the verification of the final safety of the habitat.

Mission Plan Validation

Next to verification, the product of the LEAP mission also needs to be validated. In contrary to the requirement validation, this validation is more about assessing what the product will do and whether the right product is built for the given requirements. Thus, the lunar habitat and its auxiliary systems will by subjected to various validation methods, to ensure that this design is the right way to perform the mission.

The habitat for example, has to be tested on full compatibility of all systems, both internal and external. This means that the various interfaces which will be present, have to be well integrated into the complete system and compatible with each other. This is done by testing the system end-to-end. In the case of the lunar habitat, this can be done by e.g. performing various tasks from beginning to end, and inspecting whether all systems respond in the desired way.

Furthermore, specific mission scenario tests have to be performed. For the lunar habitat, this means that the model which is built on Earth will be used to test and analyse normal operation and very specific situations which might occur. Stress tests are included in this process, to asses the robustness of the complete system. This way, the product can be validated whether it fulfils all requirements, and is able to execute the mission under various conditions and in different scenarios.

16
Market Analysis

This chapter focuses on the market regarding the lunar habitat. This includes an identification of all cost items in section 16.1 and an estimation of the costs in section 16.2. Afterwards, a brief analysis is presented in section 16.3 on the possibilities of funding project LEAP.

16.1. Cost Breakdown

A cost breakdown structure (CBS) is made to identify the expenses for all activities after the DSE. The CBS is split into 6 phases. Firstly, an organisation needs to be set up to organise the project and manage different parties partaking in the mission. Afterwards, the development phase will take place in which the design will be optimised. After that, the manufacturing process starts, which will revolve around building the parts for the habitat and assembling them. During the test phase, the produced parts and/or systems are tested and verified. The launch phase is a large and expensive phase for this project and can only be started once the final designs are finished. Furthermore, the costs for setting up the habitat are taken into account. Finally, once the habitat is operational, costs are made for continuous ground and lunar operations, maintenance, resupplies and scientific experiments.
The cost breakdown structure is presented in Figure 16.1

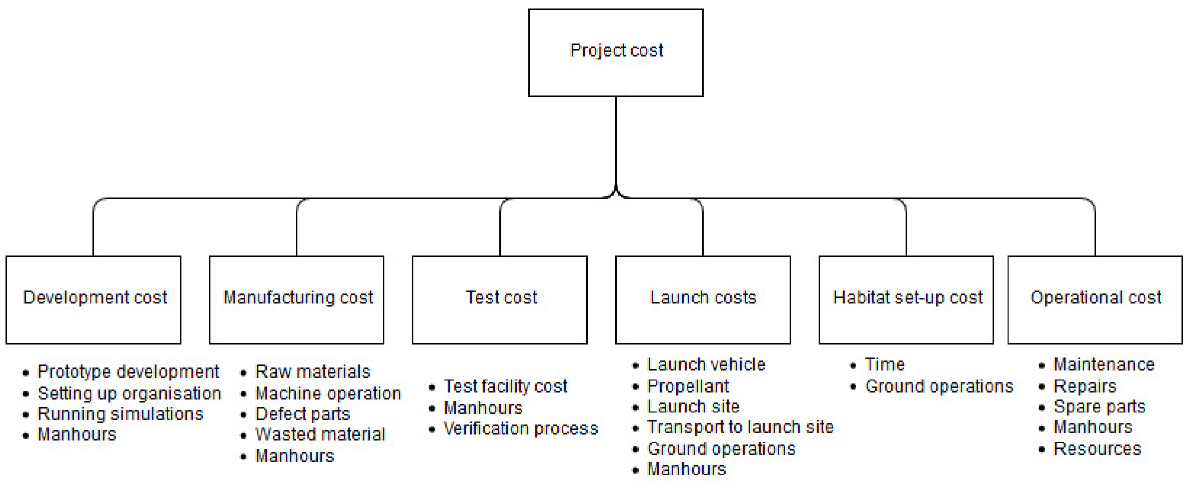


Figure 16.1: Cost breakdown structure of the lunar habitat.

One of the major costs are the development costs, mainly due to man hours. Time and money will be spent to run simulations for systems and subsystems as well as developing prototypes.
A smaller, yet not less important cost, is setting up an organisation. This involves looking for interested parties, bringing them together, making a financial, legal and operational framework and managing the overall continuation of the project.
Manufacturing costs are mainly derived from the amount of raw material that needs to be processed. Operating machines and employing people who are suitable for the job also add up to the costs. During any manufacturing process, material is being wasted, either as scrap or as a defect part. The whole manufacturing process is time intensive.
The costs of the test phase originate from testing facilities which must either be contracted or constructed. This whole verification process requires a lot of man-hours. The launch costs are one of, if not the largest costs of the project.

For the launch, the launch vehicle, propellant, launch site and transport to the launch site needs to be taken into account. Ground operations are a crucial part of a successful launch.
Setting up the lunar habitat will mostly require a lot of time. First, a suitable landing site needs to be located, after which the surface needs to be prepared for the habitat.
The final expense is the operational cost. This includes all the costs that are generated by keeping the habitat operational. Maintenance needs to be performed by astronauts to detect early malfunctions or damage. This can lead to repairs, which require spare parts. Resources, such as water, food and oxygen are also adding costs since the habitat needs a resupply at least once a year.

16.2. Cost Estimation

The cost estimation is split into two main parts, which is the development and deployment of the habitat, and the mission operations.

For the development and deployment of the habitat a cost of $35 billion was estimated in 2009[1]. This estimation heavily relies on expected developments of heavy-lift launch vehicles and crew capsules. These developments are both important for the mission to be technically feasible and for the cost to be reduced significantly due to the commercial market competing for contracts [63]. The cost of the habitat development and building for this estimate would add up to $17 billion. An amount of $14 billion is then reserved for developing landers and the $4 billion left would be used to account for launch costs to transport the habitat[1].
However, based on an inflatable concept by NASA, the cost for the development and setting up of the habitat is estimated at $12 billion. From this, $2 billion is reserved for launchers and $10 billion for development of the habitat itself[2] [63]. Although an inflatable concept is likely to reduce the payload mass and volume of the habitat, which reduces the payload capacity required, the cost reduction with respect to the previous estimation of $17 billion is quite incredible, if solely based on this difference. How this reduction is furthermore obtained is not specified. The launch costs are actually in line with previous estimations, however, the safety margin of $2 billion is omitted.

For the annual operating costs, $1 billion was estimated for the support services and equipment on the ground in 2009[1]. Next to that, it is determined in section 11.2 that two Falcon Heavy launches are required per year, one for crew and one solely for cargo. It is important to note that these resupply missions only take into account the cargo needs for the habitat. Whereas the habitat's annual operation costs are estimated at $1.44 billion today (assuming a constant yearly inflation of 2%), the total annual cost of the lunar village will likely increase when other modules are installed. For the development costs of the habitat and its lander, the original source has been used[1]. For the development costs of the rovers, the Martian explorer Curiosity ($2.5 billion in 2009) has been used as a reference [3]. With a total of four different rover types, this leads to an estimated development cost of $11.7 billion for the rovers today. In section 11.2 it is determined that one Angara A5V and three SLS Block 1B launches are required for the entire deployment of the habitat, leading to a total of $49.6 billion in non-recurring costs.
As a conclusion, for the ten-year operational lifetime of the habitat, the total costs are estimated at $64 billion today. This is the sum of the non-recurring costs and the annual operating cost multiplied by ten. Taking into account a constant inflation of 2% and a currency exchange of 0.91 €/$ this would result in a cost of €58.3 billion today[4][5].

16.3. Motivations for Funding Project LEAP

Due to the high cost of the lunar base, one needs to address the question of how it is going to be funded. In order to do so, a benefit analysis of the habitat which performed, from which the interested parties can be derived.

[1]https://www.csis.org/analysis/costs-international-lunar-base [Cited: 10-05-2017]
[2]https://www.good.is/articles/the-case-for-a-moon-base-mars-colony-nasa-project-horizon [Cited: 10-05-2017]
[3]https://www.nasa.gov/news/budget/index.html [Cited 26-6-2017]
[4]https://knoema.com/kyaewad/us-inflation-forecast-2015-2020-and-up-to-2060-data-and-charts [Cited: 11-05-2017]
[5]http://fxtop.com/en/historical-exchange-rates-graph-zoom.php?C1=EUR&C2=USD&A=1&DD1=01&MM1=01&YYYY1=1953&DD2=11&MM2=05&YYYY2=2017&LARGE=1&LANG=en&CJ=0&MM1Y=0 [Cited: 11-05-2017]

The benefits of a lunar base can be obtained first by looking at similar missions. In this case, the closest comparable project is the International Space Station. The main reason for the construction of the ISS was for research purposes. Throughout the lifetime of the station, more than 300 experiments have been carried out, excluding the insight that has been gained on human life in space. The station is mainly funded by the USA, with their investment consisting of a little more than 80%, while the remaining sum is provided by different countries. This is an option, but the chance of the US investing a great amount of money again is not likely. Nevertheless, an other group of international partners, lead by ESA, are already showing interest in developing a lunar village. To reach the amount of financing needed, a suitable option would be to set up an organisation like CERN. The main core of CERN consists of 22 countries, supported by others as well. The idea behind this concept is that the amount of financial contribution that every country provides is reflected in a number of requirements this country can set regarding the allocation and use of resources. This can be beneficial for the contributors, as a large contribution can enforce the development/production of certain components to take place somewhere, thereby increasing employment. This model can also be applied for the lunar habitat, using the resources of multiple countries, which are then contracted in different ways. This way, not only the companies of different countries can benefit, but also the gained knowledge regarding the developed technology is distributed over the countries, increasing the potential for technical collaboration in the future.
The involvement in this project of private companies in the space sector also cannot be underestimated. The last years they amassed a significant amount of knowledge and intellectual resources and they continue developing their activities because of to financial and visionary incentives. As a result, technologies are developed at a faster pace and costs are being reduced significantly.

Besides the aforementioned scientific benefits, an important motivation for deploying a lunar village is to investigate the opportunities of humans living on a different celestial body, as this is the next step for the technological development of mankind. At the moment, manned missions to Mars are already being planned, skipping the step of constructing a lunar outpost. Recently, this leap to Mars was being reconsidered though, because of the better accessibility of the Moon, which reduces transportation costs as well as mission complexity regarding safety. Stationing humans on the Moon first gives more insight in extraterrestrial habitats, which can prove valuable before investing greater amounts of resources into a martian outpost.

Additionally, the development of a lunar habitat also boosts the development of technologies that can be applied in everyday life. It already has been proven that technologies developed for the ISS are adopted in a wide variety of common products, which will most likely be the case for the lunar habitat as well. The development of the lunar village is not only beneficial to the development of 'common' technology, but it can also be a source of profit for private companies. With the current commercialization of spaceflight, a lunar village can be used as a springboard for further space exploration. Studies are carried out regarding the transformation of the ice situated at the poles of the Moon into fuel. Using this or other technologies, the lunar village can be used as a foundation for missions that go further into the solar system. The use of the village can reduce the cost of future launches, which could make it an interesting investment for private companies.

17
Sustainable Development Strategy

During current projects, the concept of sustainable development is taken into account in order to minimise the environmental, economic and social impact. The aim is to create a supportable design, which minimises its toll on the environment. In order to get a better overview of the sustainability approach, the mission is divided in different phases, which are described in section 17.1. Furthermore, section 17.2 discusses the sustainable economic strategy which has to be set up. Finally, the social impact has to be sustainable on an ethical level, considering the entire mission, as well as on a practical level inside the habitat. This is explained in section 17.3.

17.1. Environmental Sustainability

In order to obtain a clear overview of the environmental sustainability, the mission is divided into three phases: the ground phase, the space phase and the post-operational phase. The ground phase consists of the design of the lunar habitat and the set-up of the ground stations. Afterwards, the space phase includes the launch, the deployment and the operation of the lunar base. Once the habitat has completed its nominal operational lifetime of ten years, the options for dismantlement and further use of the facilities are analysed.

Ground Phase

The ground element consists of two main elements: production, and ground operations of the lunar base. Starting the production from the material selection, the sustainability has to be taken into account as the production and its location are driving factors for sustainability. The amount of energy needed and waste produced during the process are factors which are of major importance in the material process selection. Some materials used are assessed in section 4.5, which addresses the manufacturing aspect of the habitat.

The ground operations need to be considered in order to increase the mission's sustainability. As the design of the ground stations is out of the scope of the project it will not be addressed fully. A straightforward option is to make use of existing ground stations or to make use of sustainable materials if new facilities need to be constructed. Furthermore, during operation, the use of renewable energy increases the sustainability of the ground station.

Space Phase

The space phase of the mission consists of two main elements: initiation, and the operation of the lunar habitat. The initiation phase includes the launch of the habitat components and comprises the largest role in the development of sustainability up to this stage. The launch vehicle is generally not reusable and requires a high amount of fuel, thus the selection of the launch vehicle and the amount of required launches are mainly derived from the mass and size of the lunar habitat components. Therefore, the mass and size should be always be kept at a minimum. Other elements regarding the feasibility of the launch cannot be altered as it depends on the company providing the launch vehicle.

The lunar habitat should have a minimum impact on the lunar ambient, in particular regarding the waste production. Thus, all systems connected to waste production have to be designed according to this. Furthermore, the launch and deployment needs to be reliable not to lose an investment, as this will not only lead to a loss in pecuniary investment but also a loss of materials. Also, if the deployment fails, non-operational systems are introduced to the lunar environment.

Finally, as the habitat is the first part to be deployed as part of a bigger mission to create a lunar village, it shall be designed such that the least amount of changes have to be undergone when extended, to ensure the habitat's sustainability. This minimises material waste but more importantly, no extra materials have to be sent into space for alterations.

Post Operational Phase

Once the mission has fulfilled its designed lifetime, three actions can be taken: maintaining the lunar habitat operational, reusing the materials of the lunar base for other purposes or simply abandoning it.

The first two are the preferred options regarding sustainability and thus will be analysed. The first option is possible if the habitat is still reliable enough for it not to require continuous maintenance. The second option is a good alternative for the materials which have already been invested in, by having sent them to the Moon earlier on. Materials and structures could re-purposed in new assemblies or act as spare parts for existing structures, thus leading to waste reduction and cost savings. In order for this to be possible, the dismantling of the structures should be taken into account during the design of the habitat. The optimal solution is a combination of the two previously explained options, using the habitat until it is no longer possible to operate it and finally recycling the materials for other purposes. This way, the highest sustainability is obtained.

For the inflatable structure, it would be necessary to collaborate with Bigelow Aerospace, since they are in possession of the patents of the systems necessary for the structure. This company is very much into sustainable space living and exploration[1]. This might boost the modularity, reusability and recyclability of the inflatable structure. However, hard-shell type structures have been deployed in space longer, which shows that these type of structures can be operated reliably in space for double the intended mission time.

17.2. Economic Sustainability in Space

A project is economically sustainable if the usage of resources is balanced over a long term. To achieve this and to minimise the risk of running out of funds over time, thorough planning is required. Economic sustainability within this project can only be achieved with a wide range of sponsors over the entire mission lifetime, which can be ensured in two ways.

One aspect of preserving the access to sufficient funds is by maintaining the satisfaction of existing sponsors. A crucial measure to achieve this, is by fulfilling the agreements within the contract in a satisfactory manner. Furthermore, staying in touch with the sponsor, giving regular updates of the progress, and taking their input into account can help to maintain or extend sponsor contracts. Thus, having a good reputation with respect to sponsor satisfaction is crucial for being economically sustainable.

Promotion by the LEAP organisation can attract the attention of potential sponsors and thus cannot be neglected. As the project progresses, more funds are needed and thus, the recruitment of more sponsors will be a major task of the organisation to keep the project financially healthy.

17.3. Social Sustainability of the Lunar Mission

When a habitat is built on the Moon, a next phase starts for mankind, which is the settlement on other celestial bodies. For social sustainability, it is of vital importance that the start of this phase is well prepared. Space as we know it is not inhabited by any known life forms. Therefore, the impact humanity will have on celestial bodies is tremendous. Any traces left on these clean slates are there for an eternity.

The preparation of this phase consists of setting moral and ethical standards concerning living and working in space. With respect to working in space, a start has already been made to set these standards. For one, the codes of conduct that already exist for terrestrial businesses can relatively easily be adjusted to apply for extra-terrestrial businesses and settlements[2]. Furthermore, guidelines have been modelled

[1]bigelowaerospace.com/news/learnmore.php?story=spacecom_speech [Cited: 19-05-2017]
[2]www.spacefuture.com/archive/lunar_ethics_and_space_commercialization.shtml [Cited:20-05-2017]

for ethical commercialisation of outer space, on which standards for celestial bodies like the Moon can be based. These guidelines can be summarised in three terms:[2]

- **Space Preservation:** means that space is deemed valuable in itself, independent of any benefits that it (can) generate.
- **Space Conservation:** entails that the Universe's resources are to be cared for and protected for all and that exploitation benefiting the few ought to be avoided.
- **Space Stewardship:** demands that we are held accountable for managing space resources and that in all actions the effect on others, the environment and the future have to be considered.

The Agreement Governing the Activities of States on the Moon and Other Celestial Bodies together with the Outer Space Treaty were set up to regulate exploitation and exploration of space and its bodies and to ensure a fair distribution of space resources for the "common heritage of mankind"[2] [64]. This vague statement, however, can become a big hurdle for commercial development and set back technical development for years. To prevent this from happening, a global organisation will have to be set up that will collect royalty-like payments over the top revenues of space businesses and concern itself with the distribution of these payments[2].

With respect to living in space, ethical and moral standards have to be set to define the extent to which Earth is responsible for these settlers. As of now, Earth is responsible for how the settlers are affected by Earth and their own actions[2]. This responsibility is relatively easily lived up to when the settlers are in the vicinity of Earth, for instance in free orbit around the Earth or on the near side of the Moon. However, as mankind is planning to distance itself more from Earth, it will become more and more difficult for Earth to keep an eye on these settlers. With the possibility of people never returning from space and children being born on other celestial bodies in the future, it is essential that the ethical question is answered: at what stage are people no longer Earth's responsibility and are people to be allowed to escape Earth's responsibility under any circumstances?
Another problem that arises is that everyone has the human right to have a nationality. What will happen when children are born on the Moon? Is the human right taken away from them? Or will some new kind of nationality arise?
On a more practical level, social sustainability is very important within the habitat itself. Living in close quarters with three other astronauts for a year can accumulate a lot of stress and friction within a group. To reduce this friction, private space, group bonding and occupation are of great importance. If the lunar habitat were to grow into a larger community it is vital that this community is kept healthy by ensuring equity, diversity and social cohesion aside from the quality of life, governance and maturity [65].

18
Concluding Remarks

The goal of this DSE was to evaluate whether designing, manufacturing, deploying and operating a habitat hosting four astronauts for one year as part of a lunar village would be feasible and if so, to present a conceptual design for such a habitat.
It can be concluded that such a mission, named Lunar Exploration Access Point (or LEAP), is technologically, economically and scientifically feasible and sustainable while meeting the given customer requirements. Therefore it was assumed that the actual presence on the Moon would take place as of 2030 and that the habitat would be operational in 2035. As a location, a landing site in the vicinity of the Apollo 11 mission in the Mare Tranquillitatis is considered.

The LEAP habitat consists of a hard-shell module (The Shell) to which two inflatable soft-shell modules (The Hive and The Nest) are attached. The Shell is an aluminium structure shielded by aluminium plates and a combination of aramid and ceramic fabrics and synthetic layers to protect against micrometeorites and radiation and to act as thermal insulation. Also, a magnetic field is created around the hard-shell module to actively but temporarily protect against radiation.
The design of the two soft-shell modules is inspired by technology developed by Bigelow Aerospace and consists of a combination of silica, ceramic, aramid and synthetic fabrics and a synthetic foam spacer.

The mission would require a final check of the environmental conditions by gathering more recent data of the location, a mission leg named SMALL STEP 0. It is assumed that this would happen from lunar orbit by adding the required instruments to a scheduled lunar mission. After that has been done, the first payload with scouts and beaconing rovers, SMALL STEP 1, will be sent with an Angara A5V. The rovers will scout a four by four km operational area for obstacles and soil properties, after which they will position themselves as beacons on the corners of the area to facilitate precision landing. The next launch, SMALL STEP 2, consists of sending a multifunctional lunar rover (MLR), a lunar sintering rover (LSR), a space exploration vehicle (SEV) and three Power Rangers. The MLR will smoothen the surface for landing, transportation and module placement by removing debris where possible. It will also be used to transport the soft-shell modules, resupplies and landers. The LSR will sinter the ground where the modules will be placed to harden it and reduce dust. The SEV is a pressurised rover that will be used for lunar exploration. The Power Rangers are mobile units that will provide and store power. The launch vehicle selected for this part is the SLS Block 1B, which is also the preferred launch vehicle for the 2 following launches.
The launch of The Shell, in mission step GIANT LEAP 1, comes next, also containing two more power rangers. GIANT LEAP 2 is the final deployment launch containing The Hive and The Nest.
The first manned mission, MANKIND 1, comes after the modules have been connected, after which the habitat can start its operational phase. The astronauts are envisioned to perform the transportation to the Moon and back with a SpaceX Dragon 2 capsule. Furthermore, the habitat and mentioned auxiliary vehicles will require a yearly resupply mission, next to the yearly astronaut transportation, which brings the total amount of launches to 23.

The LEAP habitat is estimated to cost €58.3 billion, including development and manufacturing costs of the modules and the required rovers, the launches as described above and the operational costs of the habitat including overhead costs. Items not included are future modules that will be part of the lunar village and the launches needed for their deployment.
Economically and organisationally, the rise of private companies in space exploration makes the industry more efficient due to higher financial incentive of reducing time and costs.

Current developments in technology regarding space transportation, communication, inflatable struc-

tures and in-situ use of resources also have a significant impact on the feasibility of future space missions as the LEAP. This also means that the design of the LEAP habitat is subjected to the expected results of those developments. The minimal TRL level was set at TRL 3, which means that the conceptual design requires monitoring of the technological advancements in those fields.
For the future design of the LEAP habitat and the LEAP mission, it is recommended that the following aspects need further research:

Landers and Payload Configuration
Previous landers and their configuration only apply to smaller payloads and assume the payload to stay on top of the descent and ascent module. For large payloads that do not require to take of from the lunar surface after landing, the lander configuration from before is not an option. Different two-stage lander configurations have been proposed but not designed. Recommendations are 1) include methods of local lunar transportation, 2) further minimise elevation of the useful payload over the lunar surface for practical access to the payload, 3) design a lander configuration that practically unloads multiple large and heavy payloads (such as multiple rovers).

Inflatable Modules
Inflatable modules are in an early state of development. In order to effectively use them for future LEAP mission modules, it is recommended to research including interior design of the modules in the automated expansion phase to minimise astronaut involvement.

Module Transportation and Docking
Docking of modules in a low-gravity environment with a non-circular cross section is an operation that has not been done before. Although it is assumed that it is technologically possible and a preliminary concept has been given since it is a crucial part of the entire mission it is required to investigate how exactly the autonomous docking technology and procedure would look like.

Active Radiation Protection
The use of a magnetic shield to protect for radiation also is a practice that has not been done before. After the analysis of passive radiation protection it was concluded that in the most extreme cases of a Solar Particle Event (SPE), feasible passive protection was not sufficient to host the astronauts for one year. This will also be the case for other long duration human missions out of the protective magnetic field of the Earth. In order to provide sufficient protection, the behaviour of radiated particles around a magnetic field should be researched, especially the effect of secondary radiation.

Optical Communication
For the communication system in this design, an existing optical system setup was assumed for the sake of having precise characteristics. Optimally, a system tailored to the data transfer needs should be designed in the future that enables higher data transfer for a lower weight and power budget.

In-situ Resource Applications
The use of lunar resources, especially regolith, can be a weight saver for future missions. However, the estimated performances of the technology at the time of the LEAP mission are not yet sufficient for more practical applications next to foundation sintering. Recommendations include 1) optimising solar sintering, in order to avoid transporting sintering polymers, 2) reducing the time over volume ratio and 3) increase sintering rover autonomy.

The LEAP mission will not only comprise a habitat, but numerous scientific and operational modules in order to maximise the scientific output and increase operationality of the village. Future village lay-out will include a laboratory to increase work space and the number of experiments that can be conducted. Also, a greenhouse is recommended to research the growth of organic materials in a such a scale that it is possible to produce food for a crew of four and extra oxygen. Furthermore, a garage is needed to store rovers in order to facilitate maintainability and reduce damage from the lunar environment. To employ the aforementioned modules, an extra hard-shell connection module is needed, as well as a second storage module.

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A
Section Contributions

This appendix includes information about the contributions of the members of group 26 to the project and this final report. Table A.1 show the tasks each team member had within the organisation of the project. It also shows the back up function they were assigned in case another team member was unable to perform their own task. Fortunately, it was not necessary to make use of this back up system. Table A.2 shows for each section who wrote it (and is therefore responsible), who checked the content for inconsistencies and references, who checked the spelling and grammar, who layouted the section in case of tables and images and who made some final adjustements to the section.

Table A.1: Function distribution over the team members.

Initials	Name	Team function	Back up
KD	Karel Dhoore	Quality Assurance	Quality Assurance
KM	Karen Moesker	Secretary	Editor
LT	Liv Mare Toonen	Controller	Archivist
MM	Michiel Mendonck	Systems Engineer	Systems Engineer
MW	Maaike de Wit	Editor	Technical Manager
SK	Sander Kistemaker	Archivist	Secretary
SL	Sonny Lie	Systems Engineer	Systems Engineer
VG	Victor Gutgesell	Project Manager	Controller
VJ	Viktor Jordanov	Quality Assurance	Quality Assurance
VP	Victor Poorte	Technical Manager	Project Manager

Table A.2: Report sections and their contributors.

Section number	Written	Content checked	Spell checked
Frontmatter			
Cover	VG	–	–
Preface	MW	MM	LT
Summary	VP	VJ	MM
List of Symbols	SL	MW	MW
Introduction			
1	KM	VJ	VG
Executive Summary			
2	VG	KD, LT	VP
Environmental Protection System Design			
3.1.1	SK	LT	VG
3.1.2	LT,MM	SL	VG
3.1.3	SL	VJ	VG
3.2.1	KM	LT	VG
3.2.2.	LT	SL	VG
3.2.3	SL	VJ	VG
3.3.1	MW	VJ	VG
3.3.2	LT	SL	VG
3.3.3	MW	VJ	VG
Structural Design			

Section number	Written	Content checked	Spell checked
4.1	VJ	SL	VP
4.2	VG	SL	VP
4.3	VG	SL	VP
4.4	KM	SL	VP
4.5	KM	SL	VP
Bioastronautics System Design			
5.1	SL	KD	SK
5.2	SL	KD	SK
5.3	MW	KD	SK
Communication and Data Handling System Design			
6	KD	VJ	MM
Interior Design			
7.1	KM	SL	MM
7.2	VJ	SL	MM
7.3	VJ	SL	MM
Power System design			
8	MM	SL	SK
Subsystem Summary			
9	KM	KD	MM
Transportation and Lander Configuration			
10	VP	SL	MM
Functional Analysis and Logistics Approach			
11.1	SK	KD	MM
11.2	MW	KD	MM
11.3	MW	KD	MM
Technical Risk Assessment			
12.1	VP	LT	LT
12.2	LT	VP	VP
Technical Verification and Validation			
13	KD	KD	MM
Requirements Compliance Matrix			
14	SL	KD	MM
LEAP Mission			
15.1	MW	KD	SL
15.2	VJ	KD	SL
15.3	KM	KD	SL
15.4	VJ	KD	SL
Market Analysis			
16.1	SL	KD	MM
16.2	SK	KD	MM
16.3	KD	KD	MM
Sustainable Development Strategy			
17.1	VP	KD	SK
17.2	KM	KD	SK
17.3	MW	KD	SK
Concluding Remarks			
18	KD	SL	MM
CAD Modelling			
MM, VP, VG, VJ			
Layouting			
MW, KM			