CDF STUDY REPORT
MOON VILLAGE
Conceptual Design of a Lunar Habitat

concurrent design facility

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CDF Study Report
Moon Village
Conceptual Design of a Lunar Habitat

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FRONT COVER

Study Logo showing Stylised Lunar Habitats
with Earth in the background

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1 INTRODUCTION

1.1 Background

As part of the ESA Moon Village initiative, a moon village concept study was carried out by architecture, interior design, engineering and urban planning firm Skidmore, Owings & Merrill (SOM), in collaboration with ESA and the Department of Aeronautics and Astronautics of the Massachusetts Institute of Technology (MIT).

Following on from this concept study, the CDF has been requested by DG-SM and funded by GSP, to look into the conceptual definition of a Habitat Module as a precursor of an open concept multi partner permanent human settlement on the lunar surface dubbed “Moon Village”.

The study was carried by an interdisciplinary team of experts from across ESA as well as participation from SOM and MIT in six sessions, starting with a kick-off on the 23rd January 2020 and ending with an internal final presentation on the 18th February 2020. Unusually for a CDF study, a number of interested European entities involved in e.g. – future solutions for advanced manufacturing, development of systems for sustainable space exploration, potential suppliers from space and non-space, social scientists (e.g. anthropologists, psychologists, ergonomists) from the ESA academic network were also invited to take part as observers, with the opportunity to exchange views and express comments as the design progressed.

Before the study started and in line with the SOM-ESA-MIT Moon Village concept, some parameters were set, such as:

- Definition of the mission:
  - Crew Size
  - Mission Duration
  - Habitat Location
  - General Habitat Structure and Configuration.

1.2 Objective

The CDF was tasked with some primary objectives and some secondary objectives were to be studied if time permitted.

The primary objectives were:

- Review the boundary conditions of the performed SOM-ESA-MIT Moon Village concept study
- Identify requirements of the habitat module with regards to lunar environment
- Deliver habitat functional design features
- Define habitat interior design features
- Standardise interfaces.

The secondary objectives were:

- To define a rough Concept of Operations and ROM running costs for the habitation module
- Propose a baseline for launch and delivery to the lunar surface
- Assess potential In-Situ Resource Utilisation (ISRU).

1.3 Scope

The final output of the study is an "open to the public" report, with the intention of inspiring the ESA technology programmes to foster European capabilities supporting the development of technologies for the long term exploration of the lunar and Martian surfaces.

1.4 Document Structure

The layout of this report of the study results can be seen in the Table of Contents. The Executive Summary chapter provides an overview of the study; details of each domain addressed in the study are contained in specific chapters. This version of the report (1.1 dated September 2020) reflects some late comments and is releasable to the public.

2 EXECUTIVE SUMMARY

2.1 Study Flow

As part of the ESA Moon Village initiative, a moon village concept study was carried out by architecture, interior design, engineering and urban planning firm Skidmore, Owings & Merrill (SOM), in collaboration with ESA and the Department of Aeronautics and Astronautics of the Massachusetts Institute of Technology (MIT).

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The primary objectives were:

- Review the boundary conditions of the performed SOM-ESA-MIT Moon Village concept study
- Identify requirements of the habitat module with regards to lunar environment
- Deliver habitat functional design features
- Define habitat interior design features
- Standardise interfaces.

The secondary objectives were:

- To define a rough Concept of Operations and ROM running costs for the habitation module
- Propose a baseline for launch and delivery to the lunar surface
- Assess potential In-Situ Resource Utilisation (ISRU).

2.2 Requirements and Design Drivers

The mission requirements and design drivers for the Moon Village Habitat study are listed below:

- Habitat features, functionalities and design:
  - Ability to accommodate a crew of 4 people and support Mission Duration up to 500 consecutive days (revised to 300 days based on the crew radiation exposure assessment) for a given crew.
  - When deployed on the moon surface, the Habitat and respective support systems shall be able to provide functions for Crew Habitation as well as support to Science and Surface operations, including crew access to and from the lunar surface.

- Sufficient radiation protection to ensure exposure is within maximum allowable exposure levels for the crew over the mission duration, accounting for both periods of nominal and solar event external radiation levels.
- A 10-year lifetime after deployment on the moon surface.
- The location is to provide access to resources, optimal illumination conditions and scientific interest.

- Launch, transfer and Delivery:
  - Compatibility with current state-of-the-art launcher capabilities.
  - The Habitat and required support components shall be transferred into an appropriate Lunar Orbit, and then from lunar orbit to the moon surface.
  - Surface transfer to the final location and deployment on the moon surface.

2.3 Mission

The Habitat is the central component, allowing a crew to live and work on the surface of the Moon. Nevertheless, several additional components were deemed necessary, which can potentially be reused (landers, tugs, cranes) or expanded (power and thermal control infrastructures) with the deployment of additional Habitats and increase in the crew size.

The launch baseline is based on the Space Launch System (SLS) Block 2, which was at the time of writing the best performing launcher, also with regards to its suitability for human exploration missions. In the baseline case, separate launches are employed to launch the Habitat, Tug (which is used to insert the Habitat and in some cases also the Lander into Lunar Orbit), as well as the lander (that delivers the Habitat from Lunar orbit to the lunar surface).

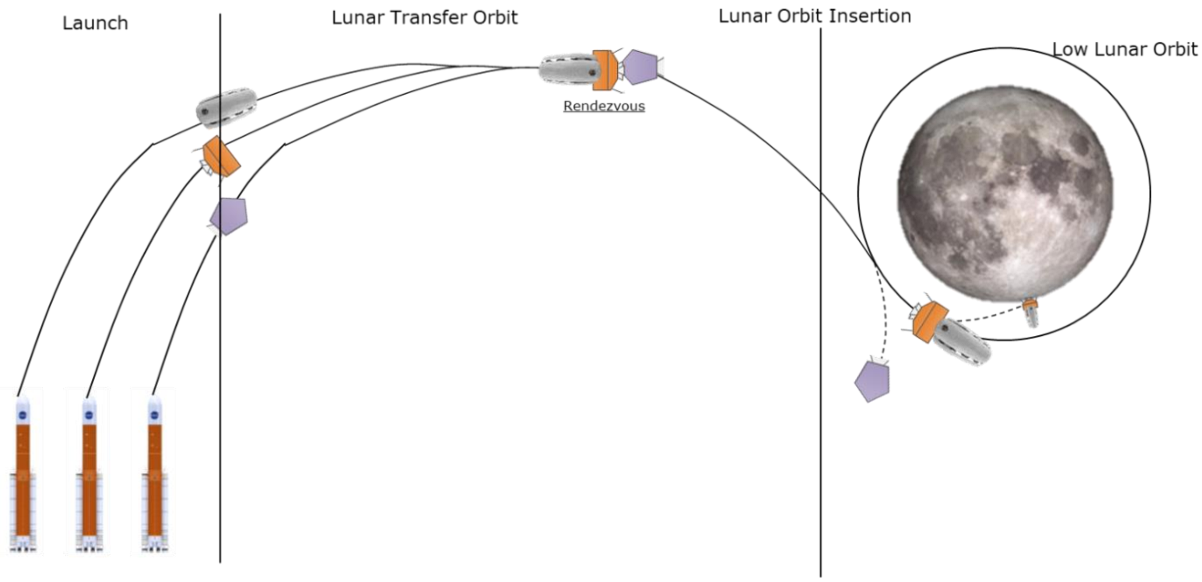


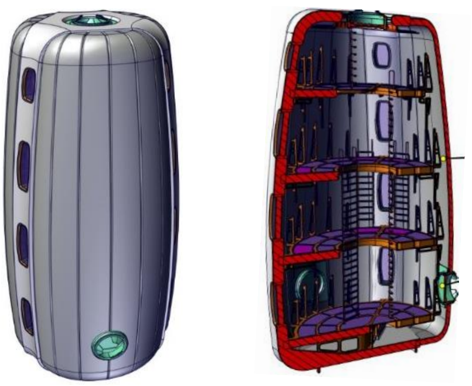
Figure 2-1: Representation of the launch, transfer and rendezvous of the Habitat, Tug and Lander, and subsequent delivery of the Habitat to the lunar surface

The Habitat itself is to be built, tested and launched from Earth, either with all or a significant part of its internal equipment. Having deployable components, it shall be launched in its stowed condition. For transfer, a variable extent of manoeuvres are to be performed, depending on the assumed launcher capabilities and strategy. Upon reaching lunar orbit, the Habitat and any additional components are to be transferred to the lunar surface, at which point a transfer to the building site is needed and deployment occurs. The crew arrives in a separate launch, performs any pending deployment activities and occupies the habitat (including adding the required shielding materials, which assumes the use of loose or sintered lunar regolith in or around the habitat walls), at which point the nominal mission starts. Crews rotate as per mission duration, and potentially support the delivery of further habitats.

During the study, the Habitat design evolved as a result of the assessment of its design features, as well as recommendations issued for further studies. Some of the most relevant impacting habitat design at System level were:

- The overall structural design is kept from the reference design, although with recommendations for further studies to be performed.
- Different subsystems designed and sized according to the identified needs. Some modifications/recommendations to be taken into account.
- Recommendation for shelter in lower level of the Habitat for increased radiation protection, including the crew accommodations and equipment for basic survival needs.
- All support and temperature sensitive equipment to be also in lower level (for less demanding temperature control during transfer, serviceability during solar storm, lower Centre of Mass for launcher limits).
- Deployable Shell layer composition proposed, taking into account MMOD protection, thermal qualities, gas permeability and heritage.
- Window layering and materials proposed taking heritage into account.
- Additional external elements such as a Power Plant, Radiators and Airlock proposed.

The outcome of this analysis resulted in the following Baseline for the Habitat design (Table 2-1).

Habitat – System baseline summary	
Configuration stowed	

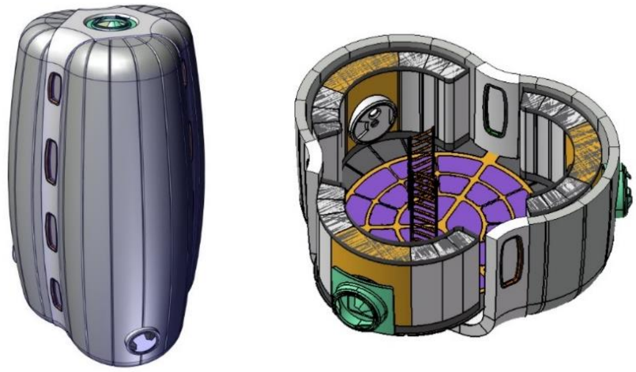
Configuration deployed		
Mass	Dry Mass (w/ margin)	58.227 kg
	Wet Mass	65.433 kg
Dimensions	Stowed	~8 m (diameter) x 15.5m (height)
	Deployed	~10.5 m (diameter) x 15.5 m (height)
Instruments and Crew Accommodation	Galley, Crew Quarters, Waste Collection, Hygiene facilities, Restraints and Mobility Aids, Medical suite and supplies	
Mechanisms	Deployable Hinged floors, Interfacing Hatches (x4), Restraining Clamp Bands (for transfer)	
Power	1 kW Intrinsic Power Generation System with 15 m² structure mounted solar panels and batteries; 59 kW surface-deployed Fission Reactor	
Environment Control and Life Support	Regenerative closed loop systems for air and water, Food production and Preparation, Waste collection and Handling, Consumable fluids (water, oxygen, nitrogen) and storage	
Radiation Protection	Nominal and Solar Event radiation protection, through use of locally sourced regolith placed on deployable walls (protection across the Habitat) and water storage on first-level floor (for shelter improved shielding)	
Thermal	Multilayer Insulation (MLI) for transfer (external blanket) and usage (integrated in the deployable shell), Heaters	
Structures	Primary Metallic Rigid "3-pillar" structure, Partially Deployable floors, Modular Interior Outfitting, Multi-layer Inflatable Shell	

Table 2-1: Habitat – System baseline summary

A launch Scenario with SLS Block 2 launches, assuming the launch of the entire Habitat, was not feasible due to the high mass of the Habitat, which in turn rendered launch of the assumed lander unfeasible. An option to deliver a half-sized habitat (or launch the habitat in 2 parts) could be in line with the assumed capability. *Table 2-2* presents this scenario, along with the launches for all the remaining components or resupply.

Mission	1. Habitat Delivery			2. Support Components Delivery			3. Resupply	
Launch No.	1.1	1.2	1.3	2.1	2.2	2.3	3.1	3.2
Launcher	SLS Block 2	SLS Block 2	SLS Block 2	SLS Block 2	SLS Block 2	SLS Block 2	SLS Block 2 (or lower)	SLS Block 2 (or lower)
Component	- Half Habitat	-Tug	-Lander	Tug, with: - Surface Radiators - Mobile Crane	Tug, with: - Power Station -ECLS Redundancies -Airlock	-Lander	Tug, with: - ECLS Fluids - Crew Supplies	-Lander
Payload Mass (10^3 kg)	28.6	N/A	N/A	18.4	19.4	N/A	8.9	N/A
Launch Mass (10^3 kg)	28.6	42.7	45.8	33.3 (14.9 Wet Tug, 18.4Payload)	34.3 (14.9 Wet Tug, 19.4 Payload)	45.8	8.9	45.8 (potentially lower)
Number of launches per deployed Habitat (no lander/tug reusability)	- 1x if half-scaled Habitat - 2x if Full Habitat split into 2	- 1x if half-scaled Habitat - 2x if Full Habitat	- 1x if half-scaled Habitat - 2x if Full Habitat	1x	1x	2x	1x / 500 days	1x / 500 days
Number of launches per deployed Habitat (lander and tug reusability, no refuel launches assumed)	- 1x if half-scaled Habitat - 2x if Full Habitat split into 2	1x for either half-scaled or full Habitat	1x for either half-scaled or full Habitat	1x	1x	0x	1x / 500 days	0x

Table 2-2: Baseline scenario SLS B2

Alternatively, an option to launch with SpaceX Starship was also studied, potentially managing to launch the Full Habitat and support equipment to the moon surface with a single launch (excluding Starship refuelling flights, and assuming some optimisation on the entire payload mass is still achievable).

The habitat design, as proposed, was not conceived for launch in two parts and assembly at the surface. The delivery option which includes future launcher capability (i.e. Starship) for the full habitat is the preferred option for future investigations, should this launcher become available.

Starship Launch Option	
Habitat Total Wet Mass	68173
Power Station	6713
Thermal Surface Radiators	4721
Airlock Module	9000
Mobile Crane	13000
Launch Adapter (allocation)	1000
Launch Mass (kg)	**102607**
Assumed StarShip Performance (kg)	**100000**
Potential Launch Performance Margin	-2607

Table 2-3: Launch Mass (all mission components) – Starship option

2.4 Technical Conclusions and Options

The proposed habitat mass by SOM is too high for SLS launched lander; an investigation is proposed to study 2-floor modules of maximum 28.6T Mass (possibly to be integrated on lunar surface), or alternatively use the Starship launcher to launch the 4-floor habitat. SOM's habitat size, however, is compatible with SLS type fairing. Investigations involving planned future launcher capability (i.e. Starship) for the full habitat is the preferred option.

Launch mass and logistics can be optimised by e.g. launching crew accommodation and life support facilities (such as redundant ones, or nitrogen/oxygen/water tanks) would need to be launched separately due to its high total mass; this would have a cascading effect since launching a high mass is also increasing the mass/size of other building blocks such as moveable crane.

Feasibility of RDV (possibly with uncooperative target) in LTO is to be investigated. As this is un-crewed, the RDV could be performed at an optimal time (for example at apogee) or timed over several orbits.

Other modules such as the high mass lander, orbit service module for the habitat, and long lifetime lunar polar power station (e.g. fission reactor) and radiator plants, lunar crane, were not part of the scope of this study, and could be designed in the future.

Inflatable structures are found to be insufficient for radiation protection; a larger contribution is expected from internal equipment. Protection against solar particle events will need to come from additional layers of e.g. water or regolith, as well as optimization of internal equipment placing (e.g. move beds to lower floor), or moving the ground floor below the lunar surface.

There is a low European heritage for using inflatable technologies in space; several advancements will need to be done (bladder design, coatings, shell, MMOD, estimation of leak rate etc.); as well as investigation into alternative methods such as 3D printing for complementing the inflatable structures for MMOD and Radiation protection.

Crew accommodation facilities, as well as life support facilities, as well as possible updates (e.g. launcher adapter fitting, precise inflatable stowing, as well as estimation of spare parts, is proposed to be designed in a follow-up study.

3 SYSTEMS

3.1 Methodology

The current study looks into the SOM-ESA-MIT Moon Village concept study and aims at identifying requirements of the habitat module, the habitat functional design and interior design features, but also at defining a rough Concept of Operations.

To achieve this, the following process steps were undertaken at systems level:

- The SOM habitat scenario and associated mission requirements and assumptions are reviewed establishing mission requirements and system design drivers and assumptions.
- The challenges and needs with respect to a concept of operations are established covering launch, transfer, any rendezvous, landing, re-fuelling.
- Key building blocks or components for realising the launch and landing of the final habitat design are identified.
- Possible mission campaign architectures are identified addressing launcher, transfer spacecraft (tug) and lander combinations.
- Sizing case assessments are made to determine the real limits of current capabilities and so identify delta's between capability and final habitat concept.
- The habitat concept as presented by SOM is assessed by each domain and when possible performed a bottom up assessment of the specific subsystem, to assess the validity of the boundary conditions and assumptions taken.
- A notional baseline design and potential Launch and Delivery strategies are defined from the component/building block options.

3.2 Background

3.2.1 ESA Moon Village Concept

The Moon Village is a concept introduced by Jan Woerner, Director General of the European Space Agency, promoting international collaboration among industry, academia, and professionals, towards a common purpose. It represents an extension of this paradigm of deep space activities, after the unparalleled level of cooperation achieved by the International Space Station.

This collective mission embodies a truly global project, mobilizing expertise and contributions from a wide range of disciplines, experiences and nations. It represents a long-term global vision with a multiplicity of opportunities for not only scientific and technological activities, but also activities based on resources exploitation or tourism.

3.2.2 Previous Work

Efforts have been made in the past, both at ESA, but also in agencies and industry worldwide, pursuing the identification of the needs and feasibility of establishing a permanent human presence on the Moon. These range for investigations to the habitation modules themselves (which vary in purpose, type and technology), but also to all other required mission components, or more generally, the building blocks needed to make such a vision feasible.

Reference RD[2] categorises and provides a comprehensive summary of these building-block groups, which range from Launchers and Transportation (Figure 3-2), Science and Research, Habitation (Figure 3-3), Resources, Electrical Power, Operations, Life Support, Robotics and Communications, and then identifies the key ones.

Figure 3-1: Images of lunar habitat options proposed by consortium set up by ESA in GSP Study 4200022835 (Graphics: Foster + Partners)

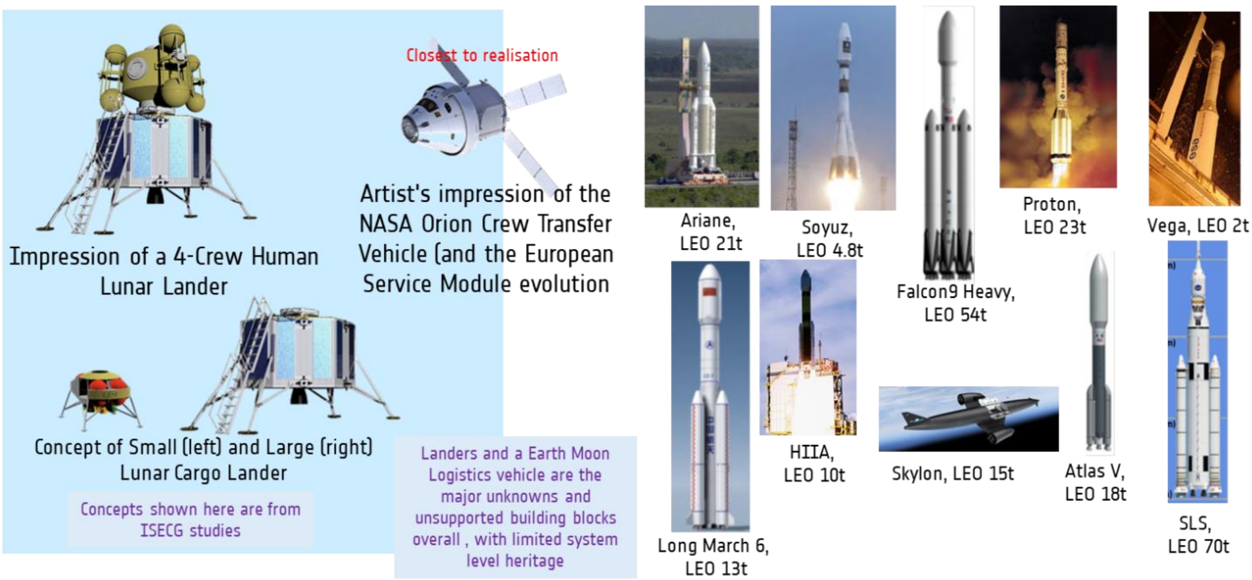


Figure 3-2: Transportation options [Ref RD[2]]

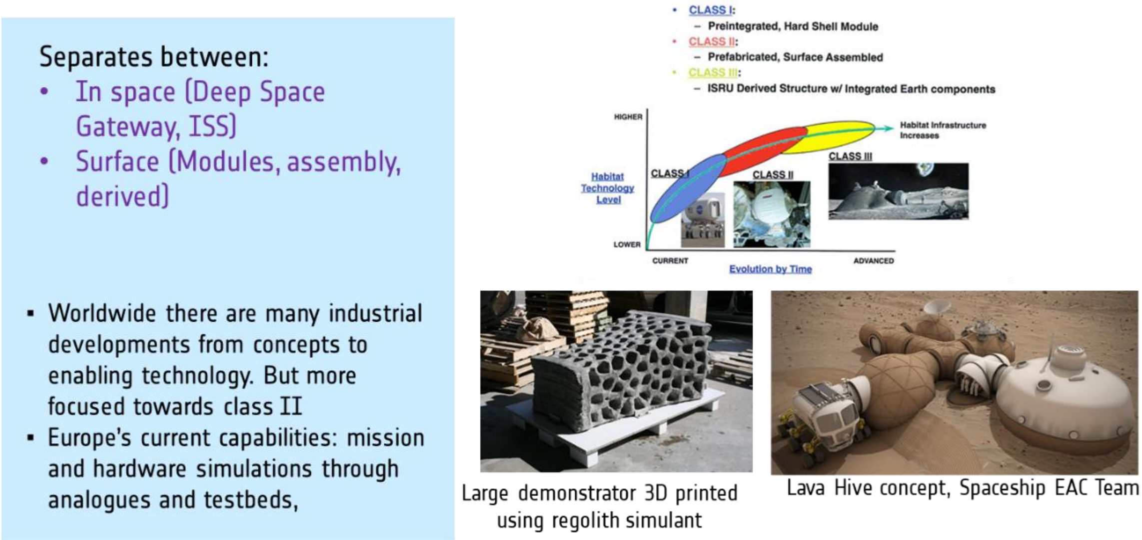


Figure 3-3: Habitat State of the Art and Definitions [Ref RD[2]]

3.2.3 ESA-SOM-MIT Proposed Habitat Design

3.2.3.1 Description

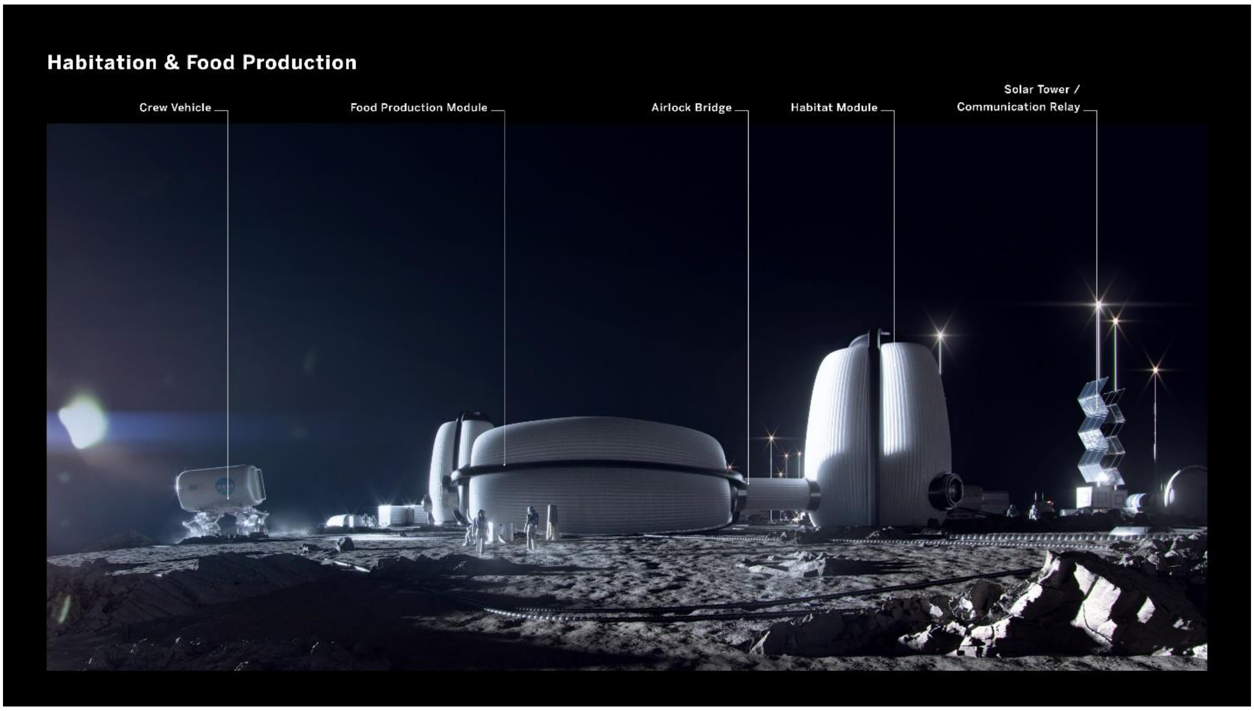
The habitat design assessed in this study was carried out by architecture, interior design, engineering and urban planning firm Skidmore, Owings & Merrill (SOM), in collaboration with ESA and faculty at the Department of Aeronautics and Astronautics of the Massachusetts Institute of Technology (MIT).

The study targeted the following topics:

- Investigating concepts for the first permanent human settlement on the lunar surface.
- Demonstrating the potential of an international private-public partnership to advance human space exploration through cross-disciplinary cooperation.
- Holistic approach to the planning of a lunar development, centring on the need for habitation systems, designed as adaptive space environments to enable versatile surface operations.

Figure 3-4: Renderings of potential lunar village (credits: SOM)

The following subsections summarise the assumptions and main outcomes of the previous study done by SOM, but more complete information can be found in reference RD[1].



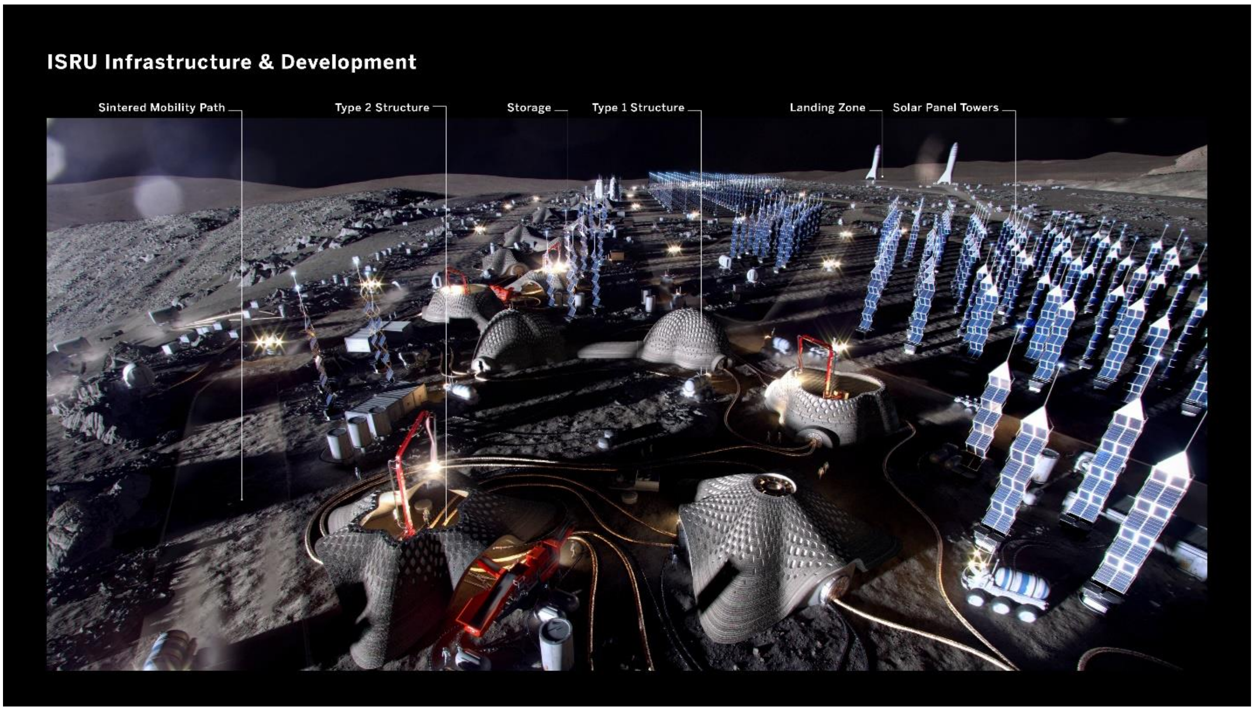


Figure 3-5: Various rendrings of potential moon village

3.2.3.2 Assumptions

The Main assumptions followed in the ESA-SOM-MIT study were as follows:

Assumptions	
1	Singular Architecture, to enable surface exploration and resource exploration
2	South pole, Near Shackleton crater rim and Permanently Shadowed Region (PSR) Cargo and logistics delivered to the moon
3	Annual resupply after first crew arrives
4	Continued delivery of modules, cargo and equipment for expansion
5	Commercial cargo vehicles capable of putting habitat in LEO will be available Habitat safety measures will be met for prolonged human exploration mission
6	International Agency and commercial partnerships will do the first stage of delivering a habitat to the moon
7	ISRU capabilities will be available for reinforcement and construction

3.2.3.3 The Habitat

The proposed habitat is comprised of a vertical Rigid Central frame, which, in combination with an inflatable deployable multilayer shell, encompass a pressurized volume of 698.29 m3 when deployed.

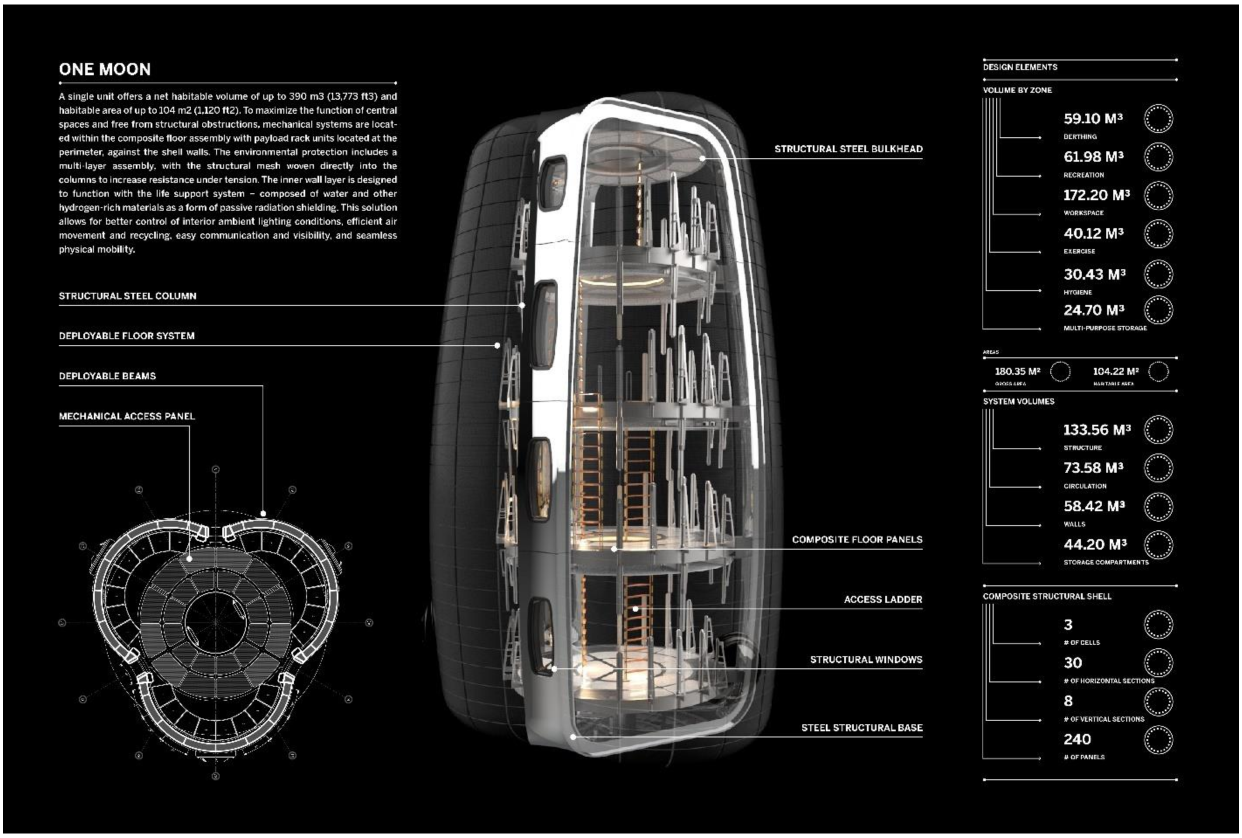


Figure 3-6: Habitat in x-ray view

The centre secondary structure includes 4 habitable levels, with a combination of fixed and deployable floors, which also provide support to the inner modules, providing several functions to the crew, including crew quarters, areas for food preparation, workstations, laboratories, support to EVA's and Command Control, among others.

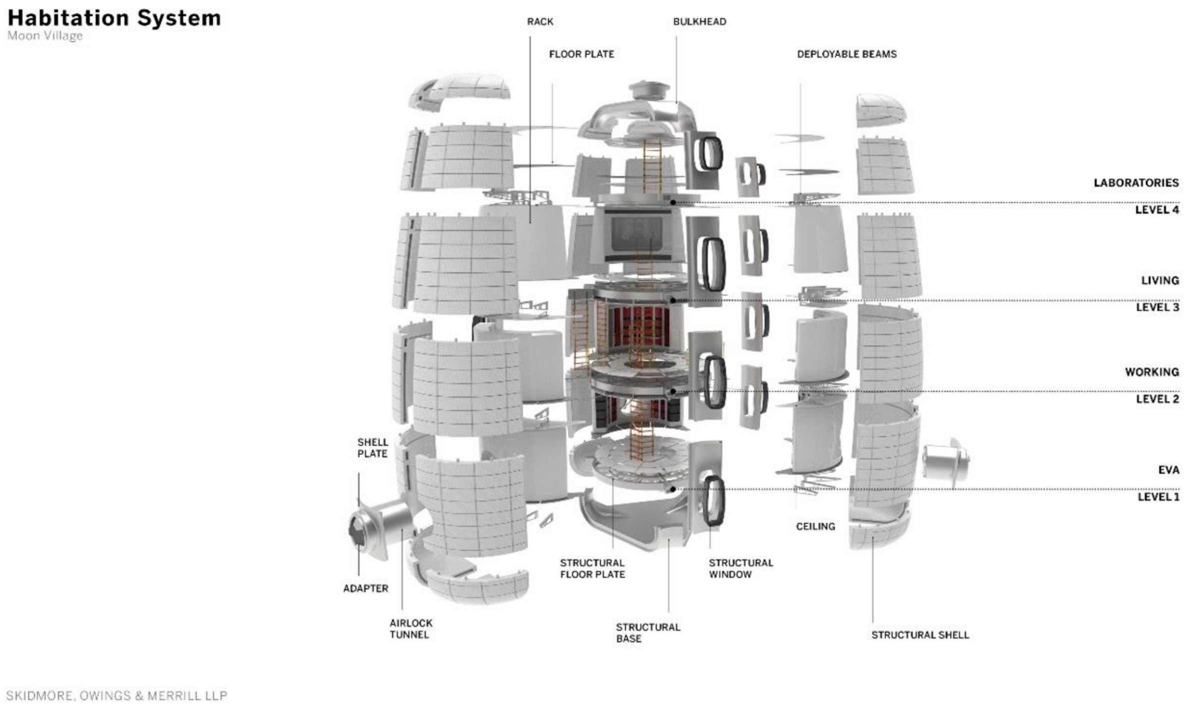


Figure 3-7: Exploded view of the proposed Habitat

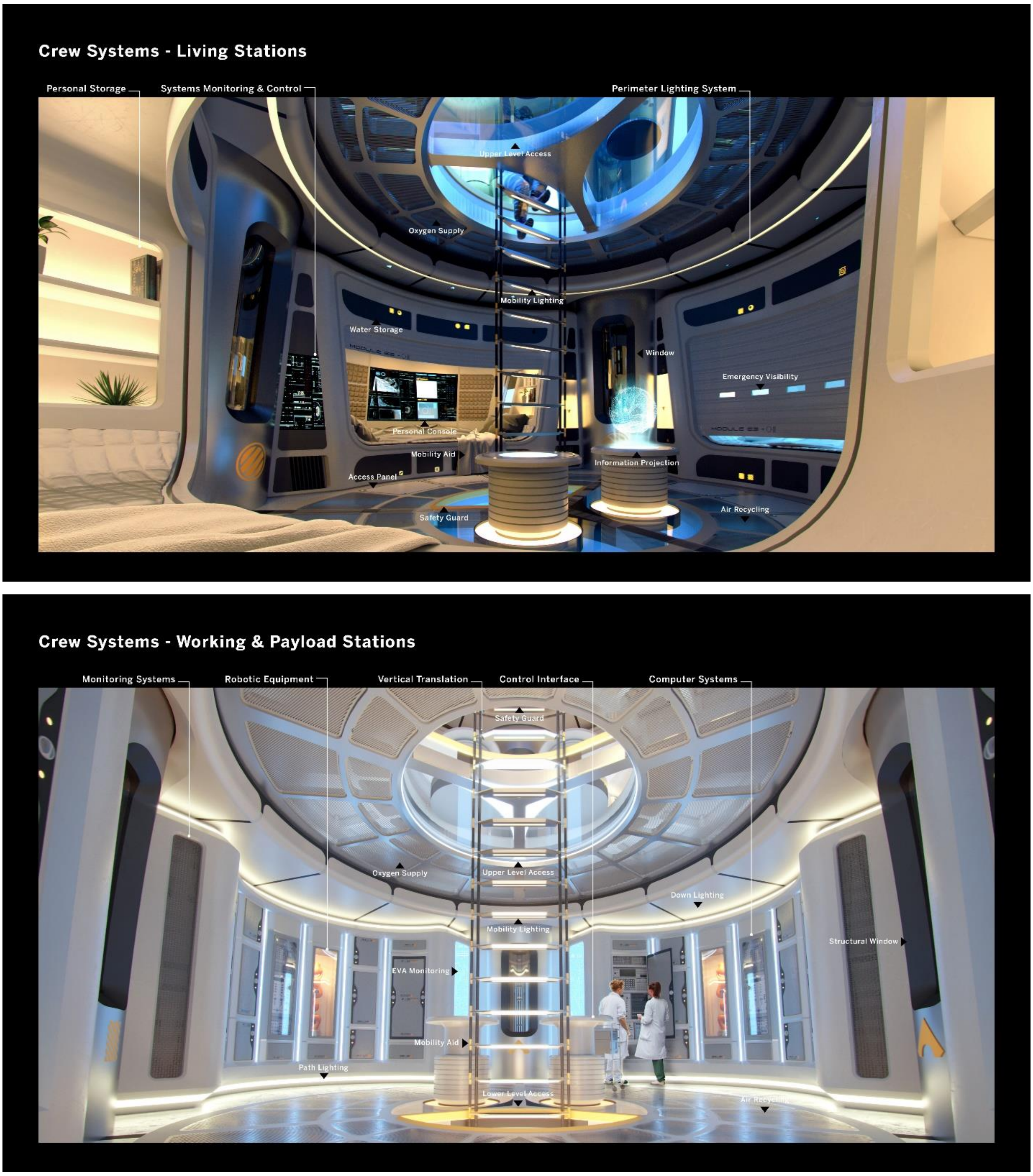


Figure 3-8: Renderings of the proposed Habitat interior

The main specifications of the Habitat, as proposed at the start of the study, are listed below:

- Crew size: 4 persons
- Mission duration: 500 days
- Location: South pole, Shackleton crater (near PSR)

- Habitat:
  - Class: Class II (module only)
    Class III (with ISRU structure)
  - Mass: 47,960 kg (wet)
  - Height: ~15.5 m (4 levels)
  - Diameter: ~10.5m (deployed)
  - Volume: 698.29 m^3 (Pressurised)
    388.53 m^3 (Habitable)
  - Area: 180.35 m^2 (Gross)
    104.22 m^2 (Net)

3.2.3.4 Comparison with current habitation modules

With the purpose of understanding if the initial Habitat mass estimate was in line with similar space structures, a comparative study was done to assess the trends with regards to mass and volume of space structures, including satellites, but most significantly habitation modules (such as the ones used in the International Space Station - ISS).

When comparing the Wet Mass per cubic meter of pressurized volume, the estimated mass for the Moon Village proposed habitat is below the trend for Space Stations and ISS Modules. It is lighter per cubic meter of pressurised volume when compared to these structures, being thus more comparable to the mass to pressurised volume ratio achieved on the Bigelow Expandable Activity Module (BEAM), an experimental expandable space station module developed by Bigelow Aerospace, that at the time of writing of this report was deployed in the ISS:

- Moon Village SOM proposed habitat (deployed): 68 kg/m^3
- Destiny/Columbus (outfitted): 257/229 kg/m^3
- Destiny/Columbus (no payload): 170/139 kg/m^3
- Kibo PM: 102 kg/m^3
- BEAM (inflatable): 88 kg/m^3

3.3 System Requirements and Design Drivers

The Mission Requirements driving the Habitat assessment performed within this study are listed in Table 3-1. They are originally defined to provide some sense of usage and lifetime. Inspiration is taken from ISS utilisation and the following trade-offs.

Mission Requirements		
Req. ID	**Statement**	**Rationale**
MIS-010	The Habitat shall be able to accommodate a crew of 4 people.	MPCV current crew capacity.
MIS-020	The Habitat shall support Mission Duration up to 300 (revised from 500) consecutive days for a given crew.	Extended ISS expeditions and Radiation Chapter 3.2, and 3.6.2
MIS-030	The Habitat shall have a 10-year lifetime (TBC) after	Multiple expeditions

Mission Requirements		
Req. ID	**Statement**	**Rationale**
	deployed in the moon surface.	
MIS-040	The Habitat shall be deployed on a site combining easy access to resources, benign illumination conditions and scientific interest.	Systems Section 3.2.3.3 ISECG GER
MIS-050	The Habitat shall provide sufficient radiation protection to have an internal radiation environment compatible with maximum allowable exposure levels for the crew over the mission duration, accounting for both periods of nominal and solar event external radiation levels.	Radiation chapter
MIS-060	When deployed on the moon surface, the Habitat and respective support systems shall be able to provide functions for Crew Habitation (including, Life Support, Crew Quarters, Hygiene, Food preparation, storage), as well as support to Science and Surface operations.	Astronaut survival and operational capability.
MIS-070	The Habitat and required support components shall be launched in stowed condition from earth.	Systems Section 3.6.5
MIS-080	The Habitat and required support components should be compatible with current state-of-the-art launcher capabilities.	Section 3.5 and 3.6
MIS-090	The Habitat and required support components shall be transferred into an appropriate Lunar Orbit.	Section 3.4
MIS-100	The Habitat and required support components shall be transferred from lunar orbit to the moon surface.	Section 3.4
MIS-110	Access of the crew to the interior of the Habitat from the moon surface (and vice-versa) shall be possible.	Section 3.2.3
MIS-120	The habitat and support components shall be transferred to the final location and deployed on the moon surface.	Section 3.4.1

Table 3-1: Mission requirements

3.4 Mission System Architecture

3.4.1 Concept of Operations

The assumed concept of operations takes as reference the delivery of the very first units of the proposed Habitat, meaning that although short-term habitation solutions are assumed as already available at the moon surface in support to short duration crewed missions at the time of deployment, the proposed Habitat is assumed as the first long-term solution.

The Habitat is to be built, tested and launched from Earth, either with all or a significant part of its internal equipment. Having deployable components, it shall be launched in stowed condition. For transfer, a variable extent of manoeuvres are to be performed, depending on the assumed launcher capabilities and strategy. Upon reaching lunar orbit, the Habitat and any additional components are to be transferred to the lunar surface, at which point a transfer to the building site is needed and deployment occurs. The crew arrives in a separate launch, performs any pending deployment activities and occupies the habitat, at which point the nominal mission starts. Crews rotate as per mission duration, and potentially support the delivery of further habitats.

3.4.1.1 Mission Phases, Challenges and Needs

From the Concept of Operations, a set of mission phases were identified, and the challenges associated with each were assessed and specific needs numbered.

Mission Phases	Challenges	Needs
Testing	• Test inflation of the structure on Earth (and interfaces between inflatable and rigid structures), interfaces between modules • Packing to survive launch loads • Possible size and mass limitation from existing test centers -> possible need for digital twins (vibration, AIV handling) • Large lifetime & inflatables -> extensive leak testing, materials testing possibly with entire setup (with solar testing)	• N/A
Transportation to launch site	• N/A	• N/A
Launch	• Mass/volume	• Adapter to mount on launcher, separation mechanism
Transfer	• Habitat powering (e.g. thermal requirements) -> likely	• Possibly a small service module in case it needs to be powered, also to provide AOCS
Optional: dock with transport module / Space Servicing Module	• Capture of habitat in space (uncontrolled) • Habitat powering (e.g. thermal requirements) -> likely • Tug likely to be very large, even with EP (beyond current estimates for e.g. CLTV)	• Reflectors to aid the close proximity • 'Handle' to capture • Possibly a small service module in case it needs to be powered -> also to provide AOCS
Optional: dock with gateway	• Habitat powering (e.g. thermal requirements) -> likely	• Possibly extra port/airlock (possibly in small service module) • LIDAR/RDV sensors on opposite of the tug (e.g. on small service module)
Optional: dock	• Similar to docking with gateway	• Possible docking to Tug

with landing module	(except safety requirements)	
Landing	• Typical precision 500 m 3-sigma. Minimum distance to Moon village in terms of dust ejection TBC • Currently no building blocks foreseen high mass landings. Designs are available • COM issue during landing	• Possibly dust reduction material on landing site • Possibly beacons to improve landing accuracy to 10m (note that beacons need to be positioned in corners of a 20x20km square. Possibly even higher accuracy with visual markers on ground
Deployment from lander	• Deployment from a lander -> Inflatables. Large lifting machine • Skycrane type lander -> no need for deployment • Moving crane • Vertical alignment • Habitat continues to need power (solar panel/battery etc.) • Need to unwrap restraints	• Either large deployment mechanism, or skycrane, or Starship. But more likely a large moving crane • Crane also used to aid unwrapping restraints
Transfer to building site	• See deployment from lander -> now deployment from rover • Insecurity of soil hardness at high Moon altitude (formed by impact ejecta) • Habitat continues to need power (solar panel/battery etc.)	• Foundation at building site. Hard surface • Hole to place partially habitat in for radiation protection (option) • Power supply
Optional: assembly of architecture	• Module stacking ->Sequence definition • Crane to lift this could be very high mass • Habitat continues to need power (solar panel/battery etc.)	• Crane high enough to place upper module
Usage	• Heat dissipation (external,) • Loss of thermal-optical properties • Protection against SPE • Outfitting & commissioning • Inflation • Unoccupied phase • EVA's, maintenance, emergencies	• Possibly infrastructure with heat dissipation structures • Ground floor to be sheltered (i.e. regolith wall or entire floor beneath the surface) • Maybe need extra shielding material on other floors TBC ray-tracing • ISS type hatch needed for EVA but more 'standing human' shape -> in tunnels

Table 3-2: Challenges and needs

3.4.2 Mission Components

The Habitat is the central component, allowing a crew to live and work on the surface of the Moon. Nevertheless, several additional components were deemed necessary.

The definition of these additional components assumes the deployment of the first Habitat of its kind. The concept and number of units can thus be scaled accordingly, and other elements potentially reused (landers, tugs, cranes) or expanded (power infrastructures) with the deployment of additional Habitats and increase in the crew size.

These additional mission components are identified in the following subsections.

3.4.2.1 Habitat Service Module

During the transfer phase, support functions to the Habitat may be required. These include:

- Power generation and supply to the Habitat to allow operating survival heaters to maintain required minimum non-operational temperatures within the allowable range for sensitive equipment (namely Environmental Control and Life Support Systems),
- Manoeuvre and Attitude Control Habitat in specific launch scenarios to allow for rendezvous and docking with other mission elements.

These tasks can be performed by a dedicated Service Module launched with the Habitat or, alternatively, by a tug (assumed for the purpose of this study).

3.4.2.2 Airlock Module

The Habitat is currently assumed to not include an integrated Airlock in the design, depending on an additional module to enable ingress and egress of crews and equipment, as well as any Extra Vehicular Activity when on the moon surface. This function is to be performed by an external additional Airlock Module that connects to one of the docking interfaces on the side of the Habitat.

3.4.2.3 Launchers

Capability of the launch segment is considered a driver for this mission. The following available launchers (current capability or in late stages of development) were considered for the study, depending on the Mission Category:

- Ariane 5, Ariane 6, Proton, Soyuz, SLS Block 2, HIIB, Long March 5, Falcon Heavy

New launcher developments were also considered (including new and early developments):

- SpaceX Starship
- SLS-like upgrade to performance requirement.

Nevertheless, SLS Block 2 performance was taken as the baseline for this study (assuming habitat mass reduction or habitat split into components and on-site assembly is possible).

3.4.2.4 Lander

A purpose-built logistics lander is assumed to be required to transfer the Habitat and Cargo from lunar orbit to the lunar surface. Current developments both within the Agency (such as the Heracles EL3) but as well in commercial landers (PTS, SpaceIL, iSpace, Astrobotic, Blue Origin) target payload masses much lower than what is required for the studied mission.

Reusability plays a big role in the sustainability of the operations on the moon surface, especially if propellants can be produced in-situ. The baseline is a single use lander. It could be imagined a lander could also make use of local O2 from ISRU systems.

Preliminary sizing and underlying assumptions are discussed in Section 3.5.2 -

Lander Sizing.

3.4.2.5 Tug

A purpose-built space tug is assumed to be required to transfer the Habitat, Lander and Cargo from Lunar Transfer Orbit into Lunar Orbit. Current developments in very early stages of development include the multi-purpose Cis-Lunar Transfer Vehicle (CLTV), RD[3], but capability is significantly below what is required for the Moon Village habitat and components.
Preliminary sizing and underlying assumptions are discussed in section 3.5.3.

3.4.2.6 Mobile Crane

A Mobile Crane system is assumed to be required to extract the Habitat from the Lander system, move the Habitat to the deployment site, and deploy the Habitat (and potentially other support systems) on site.

Preliminary sizing and underlying assumptions are discussed in the Mechanisms Chapter 6.3.4.

3.4.2.7 Power Station

Several options for Power supply to the Habitat and other mission components supporting the Habitat as well as surface operations were studied. Options include the use of nuclear fission generators or solar arrays combined with batteries or regenerative fuel cells for energy storage. Preliminary sizing and underlying assumptions are discussed in the Power Chapter 7.2.

3.4.2.8 External Radiators

Due to the operational equipment and activities performed inside the Habitat, as well as the high variability in thermal conditions the Habitat has to sustain during its operational life, heat rejection needs drive the need for the use of significant area of external radiators that have to be deployed on the lunar surface, in the vicinity of the Habitat. Preliminary sizing and underlying assumptions are discussed in the Thermal Chapter 9.2.

3.4.3 Mission Categories and Options

For the deployment of the multiple Moon Village elements and components identified during the study and in Sections 3.3 - System Requirements and Design Drivers and

3.4.2 - Mission Components, several Mission Categories can be identified, each one presenting different requirements and constraints. This means that for these different categories, the choice of launcher and trajectory may be restricted to a limited subset of options, or potentially be optimised for other categories.

For the list of potential launchers presented in Section 3.4.2.3 - Launchers, several transfer options are possible, each having advantages and drawbacks in performance (launch mass), transfer duration and different delta-V split between the Launcher and Tug. The considered transfers were:

- Direct Insertion
- Low Earth Orbit (LEO) Parking
- Geostationary Transfer Orbit (GTO) Parking
- Bi-elliptic Transfer
- Weak Stability Boundary Transfer.

Figure 3-9 schematically summarises the transfer options considered.

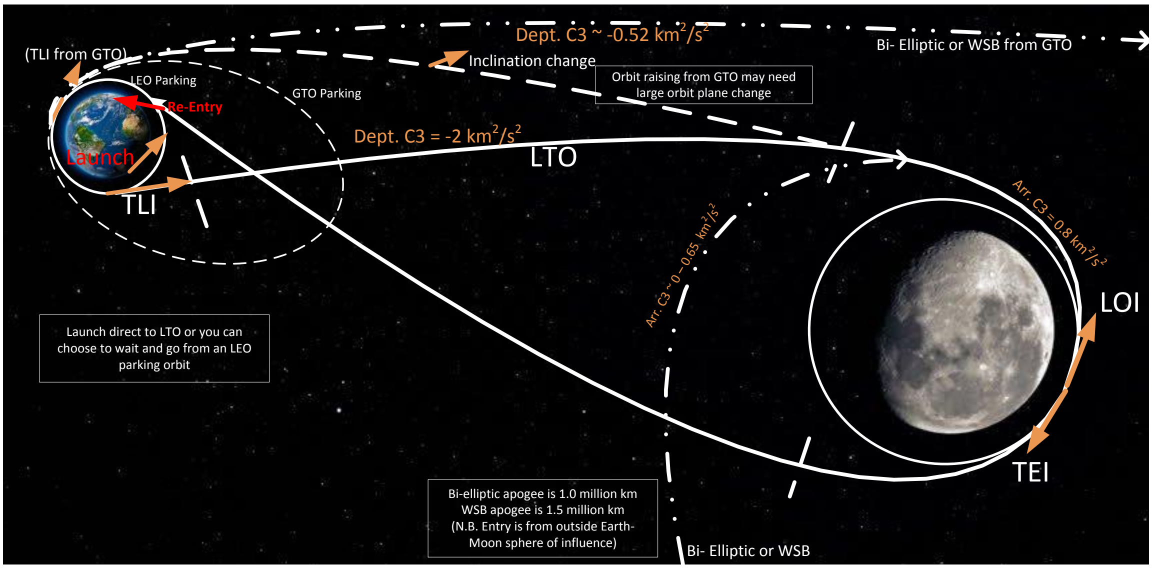


Figure 3-9: Transfer options considered for the study (source RD[2])

Multiple combinations of launchers and transfer options were studied. The suitability of these options was then assessed against the Mission Categories considered for this study, and mapped to each to each category.

The Mission Categories follow the same that were identified in Reference RD[2]. They divide into 4 categories, as follows:

- **Category 1 – Pre-cursor missions**: These have the main purpose of de-risking crucial technology (engines, payloads, GNC, ISRU). They are mostly driven by Cost, and can use a large array of transfer trajectories (LEO Parking, Bi-elliptic, Weak Stability Boundary), fitting especially the ones that minimise delta-V. Potential launch options could be based on Ariane 5 and 6, Falcon Heavy, Proton Soyuz launchers, with expected performances around 1,900 – 13,500 kg to Low Lunar Orbit (LLO), and transfer times from 5 - 87 days or more.

- **Category 2 – Large infrastructure missions**: These have the main purpose of Delivery of Large Infrastructure (Habitats, Rovers, Landers). They are mostly driven by launcher performance, and trajectory can be either optimised for cost or mass (Direct Insertion, Weak Stability Boundary). Potential launch options could be based on SLS and Ariane 5 and 6 launchers, with expected performances around 13,000 – 45,000 kg in LLO, and transfer times from 5 - 87 days or more.
- **Category 3 – Resupply missions**: These have the main purpose of regularly delivering consumables and equipment. They are mostly driven by cost, with transfer options to be selected depending on specific needs of the cargo (LEO Parking, GTO, Weak Stability Boundary). Potential launch options could be based on SLS, Proton or Ariane 5 and 6 launchers, with expected performances around 4,000 – 13,000 kg in LLO, and transfer times from 2.5 - 87 days or more.
- **Category 4 – Crewed missions**: These have the main purpose of transporting the Habitat Crew, the Crew module, and potentially a Lander and/or Tug. They are mostly driven by transfer duration, with transfer options limited to lower duration transfers, direct injection into LTO (or through LEO parking). Potential launch options are nowadays limited SLS with the ORION + ESM (European Service Module), with expected performances around 45,000 kg in LLO, and transfer times from 2.5 - 5 days.

Reference performance figures for each Mission Category with potential suitable launchers and transfer options are presented in Figure 3-10 against transfer duration. Note that the list of options in not extensive and that the mass performance estimates are assumed to include also the dry mass of any required transfer vehicles.

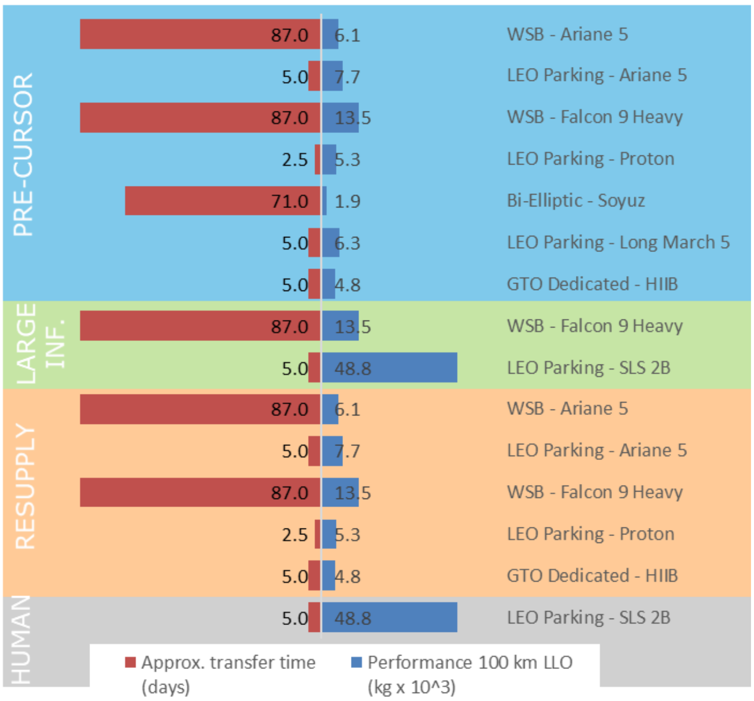


Figure 3-10: Typical Launchers and Mass Performance estimate into 100 km circular Low Lunar Orbit, as considered per mission category

3.5 System Assumptions and Trade-Offs

3.5.1 Assumptions

The assumptions taken as inputs for the Moon Village study Habitat assessment were as follows:

	Assumptions
1	Crew size: 4 pax.
2	Mission duration: 500 days (revised to 300 days based on crew radiation exposure assessment)
3	Cargo lander (E.g., EL3) with a payload capacity of 1700 kg (TBC), potentially refuelled on the surface will be available.
4	Some scenarios (TBC) require rendezvous in lunar orbit with cargo lander (specification TBD)
5	Reduced Habitat mass or Habitat can be split into up to 2 parts (and 2 launches)
6	The gateway exists.
7	Involvement of humans will be assumed to be an available capability.
8	Early pre-cursor missions have demonstrated and implemented ISRU, hence Phase 2 of ISECG lunar exploration scenario is achieved.
9	The following surface capabilities exist at the time of first SOM habitat launch:
9.1	Existing class 1 habitats already present, e.g. service habitat, astronauts module(s) (Columbus like…) to act as initial support to crew to 'un-pack' class II habitats
9.2	Limited stay of up to two weeks during construction class II habitats, with crew able to return to the gateway
9.3	Rovers
9.4	Robotics/telepresence from gateway

3.5.2 Lander Sizing

The preliminary sizing of the Lander component of the Moon Village mission took the following assumptions:

	Assumptions
1	Lander is launched from Earth without payload but fully fueled (fuel and oxidizer)
2	Lander scenario, as opposed to an alternative Sky-Crane scenario (no losses assumed for canted thrusters as would be the case for the alternative)
3	Proximity operations, Attitude Control and Hovering delta-V not taken into account at this stage (although margins are considered appropriate to cover for these)
4	Descent/ Ascent from/to a 100x100 km LLO
5	Delta-V for descent from LLO to the lunar surface taken as 1880 m/s. Reference: RD[7]
6	Delta-V for ascent from the lunar surface to LLO taken as 1865 m/s. Reference: RD[7]
7	ISP of the Cryogenic Propulsion system was assumed to be 450 s

8	ISP of the Bi-Propellant Propulsion system was assumed to be 320 s
9	For the calculation of the partial refuel of propellant in the surface (in the applicable usage scenarios), the LOX/(LOX+LH2) mass ratio was taken as 6/7ths (approximately 0.86)

For this, several propulsion technologies were first traded-off together with reusability scenarios, to evaluate the impact of those in a Lander system, which also has to be launched as part of the mission to deploy the Habitat and other elements in the lunar surface. The payload mass driving the sizing of the lander for this trade-off is the Habitat mass.

Two types of propulsion systems were considered for the Lander:

- Cryogenic: LOX/LH2
- Bi-propellant.

Three usage scenarios were considered for the Lander sizing:

- Full refuel from launch/orbit:
  - In this Scenario, it is assumed that the lander is launched with the required amount of propellant to deliver the Habitat to the surface and to return without payload to LLO, for potential refuel and reuse.
- Full refuel in orbit + partial refuel (LOX) on ground (ISRU):
  - In this Scenario, it is assumed that the lander is launched with the required amount of propellant to deliver the Habitat to the surface and the amount of fuel (LH2) to return without payload to LLO, for potential refuel and reuse. It is assumed however that a refuel of Oxidizer (LOX) is needed at the moon surface for the return trip to orbit. It is assumed that the liquid oxygen can be locally sourced.
- Full refuel both in orbit + full refuel on ground:
  - In this Scenario, it is assumed that the lander is launched with the required amount of propellant to deliver the Habitat to the surface. It can either then be discarded (single-use) or refueled at the lunar surface with both the fuel and oxidizer.

The mass of the lander was broken into 3 components:

1. The propellant mass, calculated using Tsiolkovsky equation.

$$\Delta V = I_{sp}\, g_0 \ln\left(m_0/m_f\right)$$

2. The structural support mass, which is the component of dry mass that is dependent on payload mass, was derived from historical data and reference to other landers (available at RD[3]). The parametric model philosophy was originally developed for RD[4].

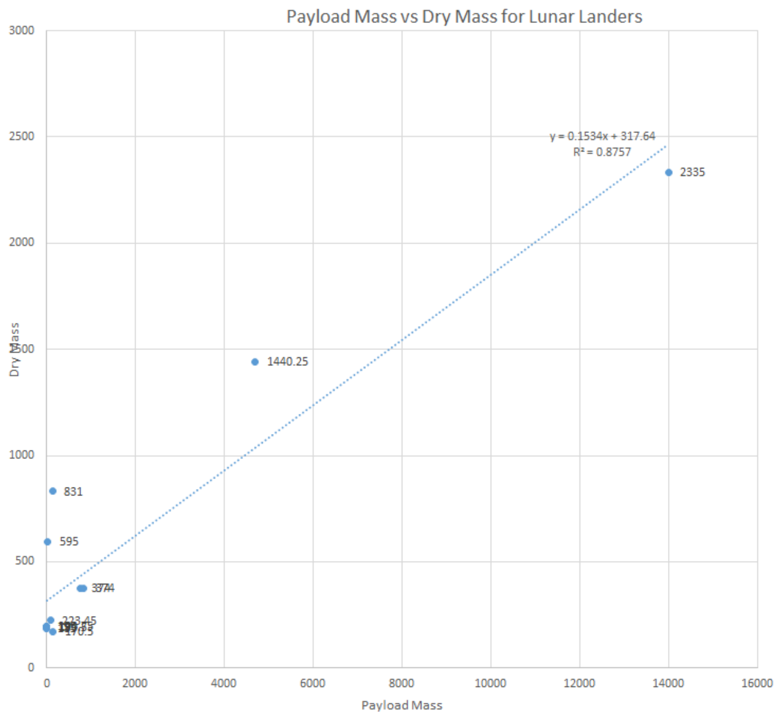


Figure 3-11: Structural Support Mass vs. Payload Mass for several currently proposed Lander designs and fitting used in the model

Fitting the data to a linear model resulted in the following approximation. It shall however be noted that this component of the dry mass does not exactly scale linearly especially for lower payload masses, where structural mass becomes more significant. As such, it is expected that this model shall under predict lander dry mass for lighter payload designs, while being potentially conservative for higher payloads. Additionally, due to the lack of data for very high mass landers (currently nonexistent), the model may have limitations, hence a full bottom-up sizing and assessment for a lander system of this magnitude is recommended at later stages.

$$m\_struc = 0.1534\, m\_pay + 317.6 \text{ kg}$$

3. Propulsion system dry mass: component of dry mass that is proportional to propellant mass.

$$m\_propdry = 0.15\, m\_propellant$$

Thus, the dry mass and wet masses of the Lander were calculated as (respectively),

$$m\_\text{dry} = m\_struc + m\_propdry$$

$$m\_\text{wet} = m\_dry + m\_propellant$$

With these assumptions, the Wet Mass of the Lander has been computed for all usage scenarios across a range of masses compatible with the mission. It can be seen that among all usage scenarios, the Bi-Propellant solutions are the worst performing options. As expected, the wet mass was also higher for the cases in which the lander is to carry the propellant (or part of it) for the ascent during the descent. Figure 3-12 shows these trends.

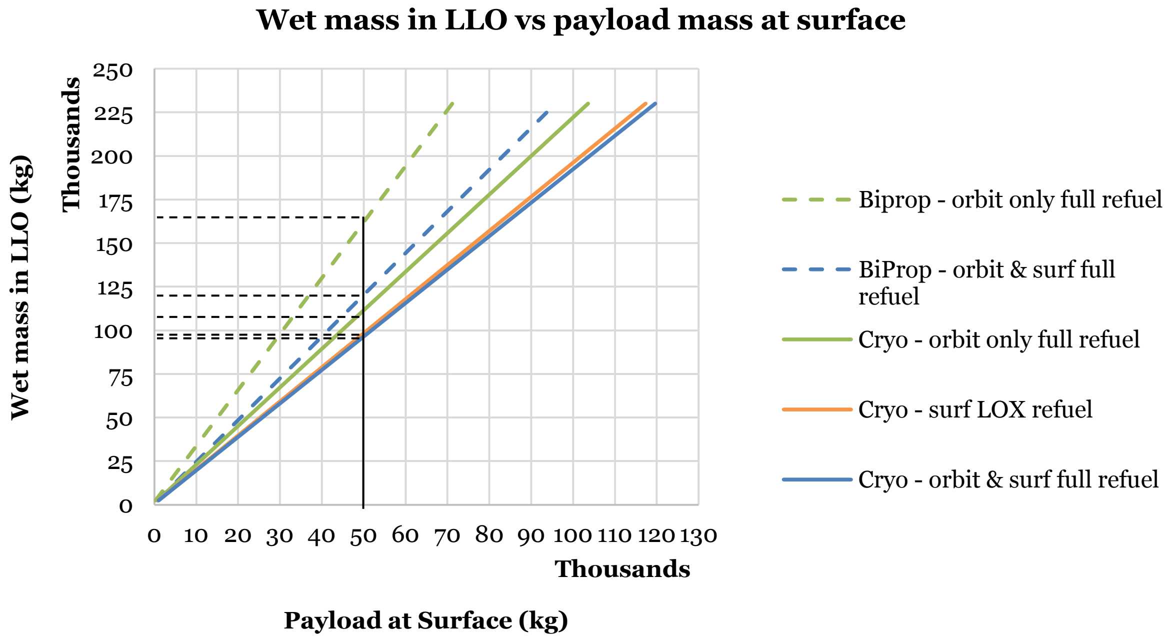


Figure 3-12: Wet Mass of a the Lander + Payload in LLO prior to descent to the surface vs. Payload Mass for the different usage scenarios assumed

Extracting the results for the approx. 48 metric ton initial Habitat mass delivery to the moon surface from LLO, the wet mass for the Lander (including payload) and its respective launch mass from Earth (including propellant but excluding payload, as it is assumed rendezvous with the habitat occurs on-orbit/during transfer) are as follows:

- Full refuel in orbit:
  - Cryogenic: LOX/LH2
    - Wet mass LLO (w/ payload): ~105 ton
    - Launch mass Lander (fueled): ~58 ton
  - Bi-propellant
    - Wet mass LLO (w/ payload): ~153 ton
    - Launch mass Lander (fueled): ~105 ton
- Full refuel in orbit + partial refuel (LOX) on the moon surface
  - Cryogenic: LOX/LH2
    - Wet mass LLO (w/ payload): ~93 ton
    - Launch mass Lander (fueled): ~46 ton
- Full refuel both in orbit + full refuel on the moon surface (or single-use lander if not refueled on the moon surface)
  - Cryogenic: LOX/LH2
    - Wet mass LLO (w/ payload): ~90 ton
    - Launch mass Lander (fueled): ~44 ton

- Bi-propellant
  - Wet mass LLO (w/ payload): ~123 ton
  - Launch mass Lander (fueled): ~75 ton

Also as noted before, the results show that for all cases the Lander propelled by a cryogenic system would have an overall lower mass due to the higher ISP. However, due to propellant losses resulting from cryogenic boil-off (which was not taken into consideration in the estimates above), the usage of the lander would have to be limited to a relatively short period after refuelling. While this could be compatible with shorter transfers to the moon, it would not allow for earlier launch of the lander in respect to other components and on-orbit loitering. Therefore, the more conservative Bi-Propellant propulsion system was taken for the rest of the study.

Regarding reusability, and since this study focused on the delivery of the first habitat to the lunar surface, this was not assumed to be a driving/enabling factor. It is assumed therefore that the lander carries only the amount of propellant needed for descent, which implies that it shall be considered as single-use, until eventually in-situ refuelling is available, potentially opening the possibility to reuse those very first Landers on the moon surface.

3.5.3 Tug Sizing

The preliminary sizing of the Tug component of the Moon Village mission took the following assumptions:

	Assumptions
1	The Tug is launched from Earth without payload but fully fueled (fuel and oxidizer)
2	Proximity operations and Attitude Control on delta-V not taken into account at this stage
3	The Tug is launched directly into Lunar Transfer Orbit (LTO)
4	Delta-V taken as 974.4 m/s, assumes insertion into 100km x 100 km LLO from LTO (arrival C3 of 0.8 km^2/s^2). Reference: RD[6]
5	Structural/subsystem sizing extrapolated from CLTV CDF study RD[3]
6	ISP of the Cryogenic Propulsion system was assumed to be 450 s
7	ISP of the Bi-Propellant Propulsion system was assumed to be 320 s
8	(Option – not taken in consideration in the sizing) The Tug is to perform the support functions required by the habitat during transfer, such as Attitude and Orbit Control, survival Power Supply and TT&C.

A Tug is used for the Moon Village missions that enables the insertion of all mission components (except the Crewed Vehicles and, in some cases, the Lander).

As seen before, the Lander may in fact have a larger wet mass at launch when compared to the Habitat. Therefore, no specific assumption is made here on whether the Tug is sized for the Habitat, the Lander, or any other mission element or combination of elements. Instead, the analysis is done considering a tugged payload mass range. The options for sizing for different elements or combinations of elements is dealt with in Section 3.5.4 - Launch Scenarios, together with the trade-off performed for selecting the most appropriate launch scenarios.

For the Tug Dry Mass estimates, structural and subsystem sizing trends were extrapolated from the CLTV study RD[3]. In this Phase 0 study, a multipurpose vehicle is designed for, among other missions, tugging payload from Lunar Transfer Orbit (LTO) to Near Rectilinear Halo Orbit (NRHO). The design allowed for a 5.1 ton maximum cargo capability (including a cargo module), achieving a mass at separation from Launcher (assumed to be an Ariane 6.4+) of 10.5 tons.

The mass of the Tug was broken into 4 components:

1. The propellant mass, calculated using Tsiolkovsky equation.

   $$\Delta V = I_{sp}\, g_0 \ln \left(m_0/m_f\right)$$

2. The structural support mass, which is the component of dry mass that is dependent on payload mass. This was assumed to be 0.19 of the payload mass, as was the case for the CLTV study.

3. The propulsion system dry mass, a component of dry mass that is assumed proportional to propellant mass. This was assumed to be 0.19 of the propellant mass, as was the case for the CLTV study.

4. Avionics/other subsystems mass which are assumed to not scale with the dimensions of the Tug were introduced in the model as a fixed value of 877 kg, also derived from the conclusions of the CLTV study

Results are presented in Figure 3-13. As expected, a cryogenic subsystem, having a higher ISP, is able to provide the required delta-V for a given payload mass with a lower launch wet mass. Both options are considered for the study.

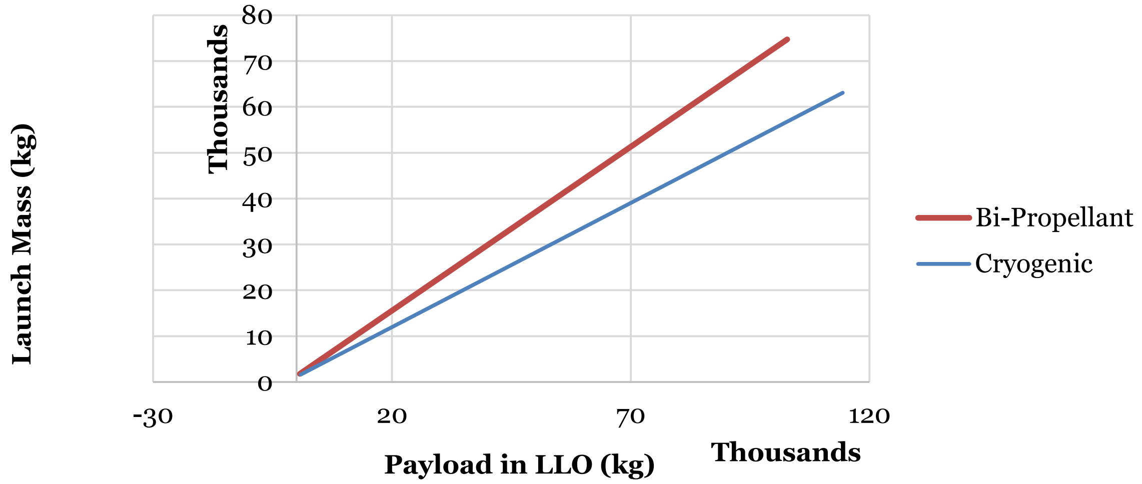


Figure 3-13: Tug Launch Mass (with fuel, no payload) vs payload mass in LLO

3.5.4 Launch Scenarios

Most of the Mission Components require being launched into lunar vicinity (in the case of this study, assumed to be Low Lunar Orbit). The Habitat was the component assumed to be driving the required launch capability, as both the sizing of the Lander and Tug derive from the Habitat Mass. Therefore, these 3 components are assumed in the following scenarios. All remaining components are assumed not to drive the launcher and scenario selection.

Although an in depth assessment of the launch options was not in the scope of this study, it was deemed important to evaluate what were the option to launch the studied components (especially the Habitat), as this would help set boundaries and interfaces for the design of the Habitat. Therefore, a performance trade-off between several potential launch scenarios was done, not only to establish the boundaries of the design space assuming current and future potential launch capabilities, but as well to choose the most appropriate split of required support elements per launch, driving the overall concept of operation for the delivery of the first Habitat

The launch scenarios were split into 2 categories:

1. **SLS Scenarios**: These scenarios assume the use of launch capability currently under advanced state of development. The baseline is therefore based in the Space Launch System Block 2, which was at the time of writing the best performing launcher, also in with regards to its suitability to human exploration missions, able to launch up to 45 000 kg to lunar vicinity, assuming a TLI with C3 = -0.99 km^2/s^2, as per RD[8] (a slightly higher performance was taken, 45.75 tons, as a departure C3 = - 2 km^2/s^2 was taken). For these Scenarios, the available mass that would be available for the Habitat is derived from the launcher performance, taking into account the need to also launch the lander and tug elements, and assuming several options for the number of launchers and combination of elements.
2. **New Launcher for Full Habitat**: In this scenario, the reference mass of the Habitat as provided in the beginning of the study was considered (47.960 kg, as per Section 3.2.3.3 - The Habitat). Then, the Tug and Lander are scaled accordingly, and the required performance for a novel Launcher is derived (one possibility is the SpaceX Starship, although with a different launch profile, see Section 3.7.1.1). The best performing scenario from the SLS Scenarios was chosen as the one to be assessed for this novel launcher assessment.

These launch scenarios assumed the launch, landing of an habitat only. Further logistic, and crew launches are required.

3.5.4.1 SLS Scenarios

Four Scenarios were established, assuming multiple launches for different combinations of the 3 components required and preliminarily sized in the previous Sections: the Habitat, the Tug, used to insert the Habitat (and in some cases also the Lander) into Low Lunar Orbit, as well as the lander that delivers the Habitat from LLO to the lunar surface.

These scenarios assume the current performance of the SLS Block 2 into Lunar Transfer Orbit (LTO), assuming the launcher's Exploration Upper Stage (EUS) performs the Trans Lunar Injection (TLI), as per the SLS Mission Planner's guide, RD[8], considering a departure C3 = - 2 km^2/s^2

The sizing of the Lander and the Tug are performed as described in Sections 3.4.2.4 - Lander and 3.4.2.5 - Tug, respectively.

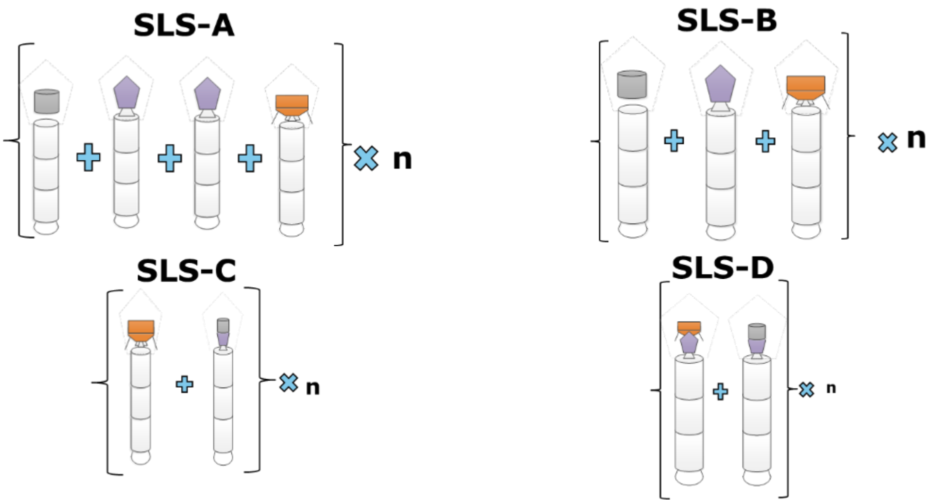


Figure 3-14: SLS Scenarios

Scenarios A and B assume rendezvous in Lunar Transfer Orbit will be feasible. Further discussions on the challenges, risk and feasibility of performing this manoeuvre is done in the GNC Chapter 8.1.

- **Scenario SLS-A** considers 4 launches to LTO, with the maximum SLS Block 2 LTO performance (45.750 kg). The launches are 2 for Bi-propellant Tugs, 1 for a Lander and 1 for the Habitat, all separately launched into LTO. Each Tug then performs a rendezvous with Lander and Habitat, respectively, which get inserted into a 100 km circular LLO. After separating from the Tugs, the Lander performs a rendezvous with Habitat, which is then delivered to the lunar surface. This Scenario allows for a maximum habitat mass of 28.9 metric tons to be launched, since the lander mass is the driver for this scenario.

	Tug (x2)	Lander (x1)	Habitat (x1)
Scenario	Bi-propellant Tugs (x2), Lander and Habitat separately launched into LTO. Rendezvous of each Tug with Lander and Habitat, respectively, insertion into LLO and delivery of Habitat to surface.		
Launcher	SLS B2 (or lower)	SLS B2	SLS B2 (or lower)
Propulsion	Bi-propellant	Bi-propellant	-
ISP (s)	320	320	-
Purpose	Inject habitat+lander assembly in LLO	Land habitat	-
Manoeuvre	LTO to LLO insertion	LLO-DOI-Surface	-
Delta-V (m/s)	974.4	1967.7	-
Wet Mass - Launch config. (10^3 kg)	**35.3**	**45.8**	**28.9**
Wet Mass - incl. Payload (10^3 kg)	81.1	74.7	-
Dry Mass (10^3 kg)	13.7	11.0	-
Payload (10^3 kg)	45.8	28.9	-

Table 3-3: Launch performance for Scenario A

- **Scenario SLS-B.1** targets the optimization of the number of launches by assuming a single bi-propellant Tug is capable of inserting both the lander and the habitat into LLO. The launches are 1 for a Bi-propellant Tug, 1 for a Lander and 1 for the Habitat, all separately launched into LTO. The Tug then performs a rendezvous with both the Lander and the Habitat, which have previously rendezvoused, then inserting the assembled component into a 100 km circular LLO. After separating from the Tug, the Lander delivers the Habitat to the lunar surface. This Scenario results however in a reduction of the maximum habitat mass to 23.1 metric tons, since the tug mass is the driver for this scenario.

	Tug (x1)	Lander (x1)	Habitat (x1)
Scenario	Bi-propellant Tug, Lander and Habitat separately launched into LTO. Rendezvous of single Tug with both Lander and Habitat, insertion into LLO and delivery of Habitat to surface.		
Launcher	SLS B2	SLS B2 (or lower)	SLS B2 (or lower)
Propulsion	Bi-propellant	Bi-propellant	-
ISP (s)	320	320	-
Purpose	Inject habitat+lander assembly in LLO	Land habitat	-
Manoeuvre	LTO to LLO insertion	LLO-DOI-Surface	-
Delta-V (m/s, incl. 5% margin) *	974.4	1967.7	-
Wet Mass - Launch config. (10^3 kg)	**45.8**	**36.7**	**23.1**
Wet Mass - incl. Payload (10^3 kg)	105.5	59.8	-
Dry Mass (10^3 kg)	17.6	8.8	-
Payload (10^3 kg)	59.8	23.1	-

Table 3-4: Launch performance for Scenario B.1

- **Scenario SLS-B.2** Looks at the same launches and concept of Scenario SLS-B.1, but assuming a Tug propelled by a more efficient cryogenic propulsion system. This modification results in an increase of the maximum habitat mass to 28.9 metric tons (the same as Scenario SLS-A), since the tug mass is no longer the driver for this scenario, but rather the lander becomes the heaviest component.

	Tug (x1)	Lander (x1)	Habitat (x1)
Scenario	Cryogenic Tug, Lander and Habitat separately launched into LTO. Rendezvous of single Tug with both Lander and Habitat, insertion into LLO and delivery of Habitat to surface.		
Launcher	SLS B2 (or lower)	SLS B2	SLS B2 (or lower)
Propulsion	Cryogenic	Bi-propellant	-
ISP (s)	450		-
Purpose	Inject habitat+lander assembly in LLO	Land habitat	-
Manoeuvre	LTO to LLO insertion	LLO-DOI-Surface	-
Delta-V (m/s, incl. 5% margin) *	974.4	1967.7	-
Wet Mass - Launch config. (10^3 kg)	**42.7**	**45.8**	**28.9**
Wet Mass - incl. Payload (10^3 kg)	117.4	74.7	-
Dry Mass (10^3 kg)	19.5	11.0	-
Payload (10^3 kg)	74.7	28.9	-

Table 3-5: Launch performance for Scenario B.2

On the other hand, **Scenarios C and D** look at options avoiding the dependency on the LTO rendezvous manoeuvre, requiring only a rendezvous in Low Lunar Orbit.

- **Scenario SLS-C** assumed the Habitat is assembled in a Bi-propellant Tug and launched assembled, and a lander with capability for LLO insertion is launched on a second launch, both sent into LTO. This Scenario allows for a maximum habitat mass of 18.2 metric tons to be launched, significantly reduced from the previous scenarios since the lander mass is the driver for this scenario.

	Tug + Habitat (x1)	Lander (x1)	(Habitat)
Scenario	The Habitat is launched with a Bi-propellant Tug, and lander with capability for LLO insertion are sent into LTO. Both are inserted into LLO (the tug inserts the Habitat, the Lander inserts itself), rendezvous of the lander and the habitat and delivery of Habitat to surface.		
Launcher	SLS B2 (with Hab)	SLS B2	-
Propulsion	Bi-propellant	Bi-propellant	-
ISP (s)	320	320	-
Purpose	Inject habitat or lander in LLO	Land habitat	-
Manoeuvre	LTO to LLO insertion	LLO-DOI-Surface	-
Delta-V (m/s)	974.4	1967.7	-
Wet Mass - Launch config. (10^3 kg)	**33.1**	**45.8**	**(18.2)**
Wet Mass - incl. Payload (10^3 kg)	33.1	51.8	-
Dry Mass (10^3 kg)	6.0	9.4	-
Payload (10^3 kg)	18.2	18.2	-

Table 3-6: Launch performance for Scenario C

- **Scenario SLS-D** assumed that 2 Cryogenic Tugs, launched with the Lander and the Habitat into LTO. This Scenario allows for a maximum habitat mass of 17.9 metric tons to be launched, also significantly reduced from scenarios A to B.2, and lower than Scenario C since the lander mass is reduced due to sharing of the launch mass with the respective Tug.

	Tug + Habitat or Lander (x2)	(Lander)	(Habitat)
Scenario	Cryogenic Tugs (x2), each one launched already with the Lander and the Habitat into LTO. Insertion into LLO, rendez-vous of the lander and the habitat and delivery of Habitat to surface.		
Launcher	SLS B2 (with Lander or Hab)	-	-
Propulsion	Cryogenic	Bi-propellant	-
ISP (s)	450	320	-
Purpose	Inject habitat or lander in LLO	Land habitat	-
Manoeuvre	LTO to LLO insertion	LLO-DOI-Surface	-

Delta-V (m/s, incl. 5% margin) *	974.4	1967.7	-
Wet Mass - Launch config. (10^3 kg)	**45.8 (w/Lander) / 29.0 (w/ Hab)**	**28.6**	**17.9**
Wet Mass - incl. Payload (10^3 kg)	45.8 (w/Lander) / 29.0 (w/ Hab)	46.6	-
Dry Mass (10^3 kg)	8.0 (Lander version) / 5.4 (Hab version)	7.0	-
Payload (10^3 kg)	30.4 (Lander version) / 21.4 (Hab version)	17.9	-

Table 3-7: Launch performance for Scenario D

In summary, the maximum Habitat (or Habitat component) mass assuming LTO rendezvous that can potentially be achieved with the SLSB2 is 28.9 metric tons, with either 4 launches - Scenario A, or 3 launches - Scenario B.2. If LLO rendezvous only is assumed the maximum mass that can be achieved is 18.2 metric tons (Scenario C).

	Habitat (or component) mass (10^3 kg)
Scenario A – SLS B2 capability, Bi-prop Tugs	28.9
Scenario B.1 – SLS B2 capability, Bi-prop Tug	23.1
Scenario B.2 – SLS B2 capability, Cryo Tug	**28.9**
Scenario C – SLS B2 capability, Bi-prop Tug	18.2
Scenario D – SLS B2 capability, Cryo Tug	17.9

Table 3-8: Summary of maximum Habitat Launch Mass for each Launch Scenario

3.5.4.2 New Launcher for Full Habitat

One other assessment that was done was to check what would be the performance of a potential launcher designed to comply with the reference habitat launch mass of 47.960kg.

This scenario considers 3 launches to LTO, and is derived from SLS Scenario B.2, as this was the best performing option. As for the previous cases, the lander is launched with propellant for descent only.

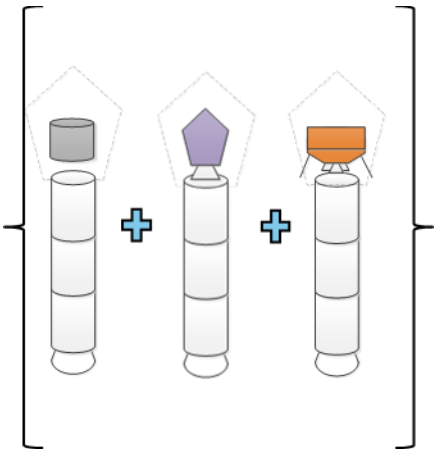


Figure 3-15: SLS Scenario B.2

This scenario concludes that for a Habitat mass of 48 metric tons would require 75.3 metric tons capability to LTO, assuming LTO rendezvous. When compared with the previous scenarios, SLS B2 is as expected not capable of launching the required mass into LTO.

	Tug (x1)	Lander (x1)	Habitat (x1)
Scenario	Cryogenic Tug, Lander and Habitat separately launched into LTO. Rendezvous of single Tug with both Lander and Habitat, insertion into LLO and delivery of Habitat to surface.		
Launcher	New Development (TBD)	New Development (TBD)	New Development (TBD)
Propulsion	Cryogenic	Bi-propellant	-
ISP (s)	450		-
Purpose	Inject habitat+lander assembly in LLO	Land habitat	-
Manoeuvre	LTO to LLO insertion	LLO-DOI-Surface	-
Delta-V (m/s, incl. 5% margin)	974.4	1967.7	-
Wet Mass - Launch config. (10^3 kg)	**69.8**	**75.3**	**48.0**
Wet Mass - incl. Payload (10^3 kg)	193.1	123.3	-
Dry Mass (10^3 kg)	31.6	17.9	-
Payload (10^3 kg)	123.3	48.0	-

Table 3-9: Launch performance requirement for hypothetical purpose-built launcher

3.5.5 Habitat Radiation Shielding Trade-Off

Radiation protection to the crew over long inhabitation periods is an important function driving the Habitat design. The shielding needs are defined from the expected radiation environment, as discussed in the Radiation Chapter 10.2. Hence, the radiation conditions anywhere inside the Habitat are to be kept within the limits of exposure for the crew, assuming a normal expected radiation environment. On the other hand, the Habitat shall also provide a “shelter” area with increased protection, for when solar events occur. This section is assumed to be the ground floor of the Habitat, as it is easier to place the required amount of material closer to the surface.

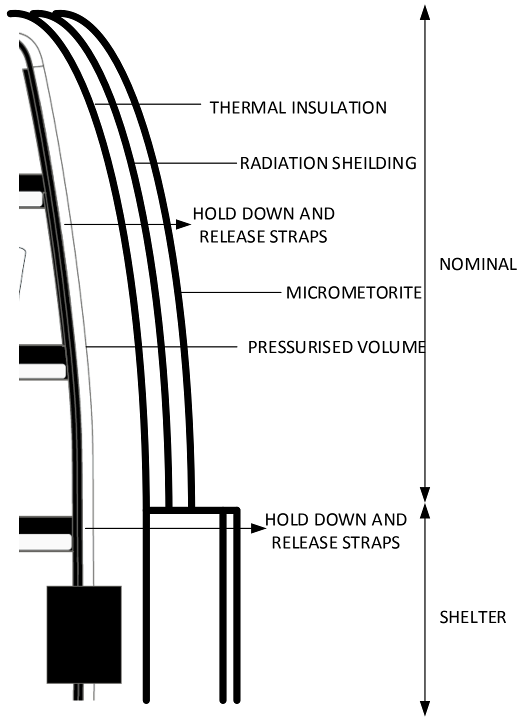


Figure 3-16: Schematic of the Habitat design highlighting the regions where shielding compatible with Nominal and Shelter shielding are to be applied

The amount of material required around the entire habitat is as follows, for each condition/section of the Habitat:

- Radiation shield normal conditions: 9 g/cm²
- Radiation shield solar events: 25 g/cm²

The shielding implies therefore the existence of a significant amount of material on the outer walls of the Habitat. While part of this mass will already be present, either due to the density of deployable walls or the structure of the Habitat itself, most of the radiation protection has to be provided by additional material whose function would be limited to radiation shielding. Launching this amount of additional "filler" shielding material is an option, but it is a very costly one as the required shielding mass is in the hundreds of metric tons, much above the Habitat launch mass. Therefore, the use of materials available in-situ, such as regolith and, potentially, water, as fillers to provide the required radiation shielding was baselined.

Specifically in what concerns the lunar regolith, there are several potential ways to use this material for shielding. The first one would be just using directly the loose lunar regolith, either piled around the habitat or placed in preexistent pockets in the outer walls of the habitat. The second option would be to sinter the regolith into external walls, constructed in situ through regolith sintering, a technique currently under development at ESA.

Due to the variety of ways these materials can be used, a trade-off and sizing exercise was performed for a deployable shield and needed fillers (when applicable). Table 3-10 summarizes the advantages and shortcomings of the different solutions considered.

Option #1: Foam + Regolith	Option #2: Foam + Water	Option #3: Foam + regolith wall
Pockets of space after deployment are filled either with loose regolith or sintered blocks	Pockets of space after deployment are filled with water	No spacers, and wall for radiation is constructed in-situ. Possibly for shelter.
No need to launch fillers other than 10% area of foam in the outer shell for deployment	Effective barrier for radiation. Launch of water (could be) needed. Water-tightness for pockets would have to be ensured	No filler launches. However, construction infrastructure need

Table 3-10: Trade-off on shielding

From this trade-off, the baselined solution was to provide the shielding with loose regolith, placed in pockets in the deployable shell.

3.6 Baseline Solution

3.6.1 Habitat Overview

After the overall assessment of the Habitat and identification of each subsystem's design and respective assessment and/or resizing, several modifications or recommendations for design were derived, and are presented in the different Chapters of this report.

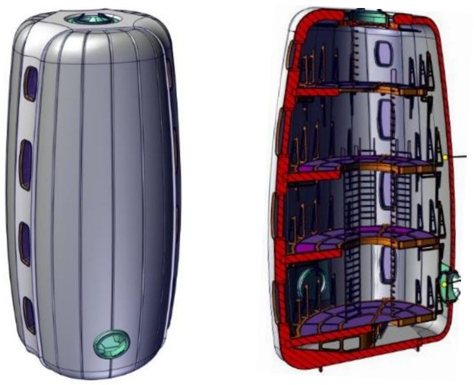
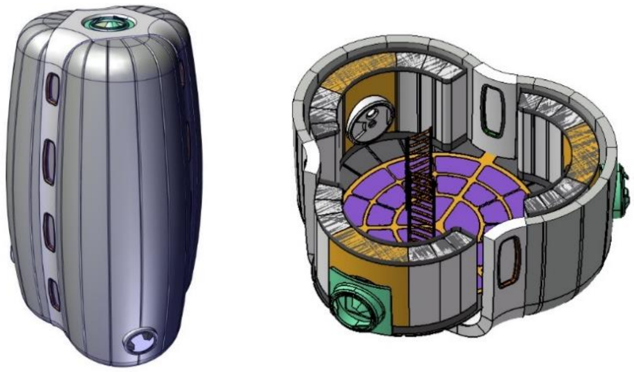
In this and the following subsections, a summary of the design solutions proposed for each subsystem is presented, along with revised Mass and Power Budgets. A launch Baseline is also proposed for this revised design.

Besides the assessment and sizing per subsystem, some overall recommendations/ design changes were also assumed or recommended in this design reassessment. A non-extensive list of the work performed or recommendations issued impacting habitat design at System level were:

- The overall structural design is kept from the reference design, although with recommendations for further studies to be performed.
- Different subsystems designed and sized according to the identified needs. Some modifications/recommendations to be taken into account.
- Recommendation for shelter in lower level of the Habitat for increased radiation protection, including the crew accommodations and equipment for basic survival needs.
- All support and temperature sensitive equipment to be also in lower level (for less demanding temperature control during transfer, serviceability during solar storm, lower Centre of Mass for launcher limits).
- Deployable Shell layer composition proposed, taking into account MMOD protection, thermal qualities, gas permeability and heritage.
- Window layering and materials proposed taking heritage into account.

- Additional external elements such as a Power Plant, Radiators and Airlock proposed.

3.6.2 Habitat Design Summary

Habitat – System baseline summary		
Configuration stowed		
Configuration deployed		
Mass	Dry Mass (w/ margin)	58.227 kg
	Wet Mass	65.433 kg
Dimensions	Stowed	~8 m (diameter) x 15.5m (height)
	Deployed	~10.5 m (diameter) x 15.5 m (height)
Instruments and Crew Accommodation	Galley, Crew Quarters, Waste Collection, Hygiene facilities, Restraints and Mobility Aids, Medical suite and supplies	
Mechanisms	Deployable Hinged floors, Interfacing Hatches (x4), Restraining Clamp Bands (for transfer)	
Power	1 kW Intrinsic Power Generation System with 15 m^2 structure mounted solar panels and batteries; 59 kW surface-deployed Fission Reactor	
Environment Control and Life Support	Regenerative closed loop systems for air and water, Food production and Preparation, Waste collection and Handling, Consumable fluids (water, oxygen, nitrogen) and storage	
Radiation Protection	Nominal and Solar Event radiation protection, through use of locally sourced regolith placed on deployable walls (protection across the Habitat) and water storage on first-level floor (for shelter improved shielding)	

Thermal	Multilayer Insulation (MLI) for transfer (external blanket) and usage (integrated in the deployable shell), Heaters
Structures	Primary Metallic Rigid "3-pillar" structure, Partially Deployable floors, Modular Interior Outfitting, Multi-layer Inflatable Shell

Table 3-11: Habitat baseline summary

3.6.3 Mass Budget

A bottom-up evaluation of the Habitat mass was performed in this study, based on the subsystems sizing, deriving from the requirements and needs identified.

The Mass budget for this mission's main component, the Habitat, is decomposed as follows:

- **Habitat Wet Mass**: divided into 2 major components, the sum of which shall represent the full Habitat mass when fully deployed and assembled in the surface (not including radiation shielding additional material):
  - **Total Dry Habitat Mass**: includes the minimum set of components of the Habitat that can be launched pre-assembled or requiring limited assembly. This is the Habitat structure, deployable shell, as well as most internally assembled subsystems, crew quarters and life support subsystems.
    - **Redundant ECLS Subsystems Dry Mass**: The redundant ECLS subsystems, although included in the Total Dry Habitat mass, can be taken out to split the mass of these redundancies into a separate launch.
  - **Resupply Mass (ECLS & Crew Supplies):** This includes the mass for all required ECLS fluids (Water, Oxygen, Nitrogen) and respective containers (tanks), as well as other crew supplies that need to be shipped both in the beginning of the mission, as well as periodically (every 500 days).

The revised mass budget of the Habitat is presented in Table 3-12.

			Mass (kg)
COM	Communications		-
ECLS	Life Support Systems		10584
CREW	Crew Accommodation		2532
MEC	Mechanisms		991
PWR	Power		1197
RAD	Radiation		-
STR	Structures		30239
TC	Thermal Control		3600
SYE	Systems Engineering		-
-	Harness	2.6%	1230
	Total Dry Mass Hab		**50420**

Dry Mass Hab		**50420**
System Margin	20%	10084

	Mass (kg)
Dry Mass incl. System Margin	**60505**
Resupply (ECLS)	5716
Resupply (Crew Supplies)	1952
Total Wet Mass	**68173**

Table 3-12: Habitat Mass Budget

Accounted for on the Structures mass budget entry are also:

- Deployable shell: this component was the target of a full reassessment, with the definition of the layering for the different functions it performs, as discussed in the Materials and Processes Chapter. The mass for this component was estimated at 7195 kg, resulting from an area density of 1.71 g/cm² for an approximate area of 420 m² of deployable surface.
- Windows: The 12 windows included in the Habitat Design sum up to an area of approximately 10.5 m². The reference used was the 4-pane fused silica and borosilicate glass window used in the Cupola in the International Space Station (RD[10]), which is composed of the following elements: 1 x 9.3 mm inner scratch pane+ 2 x 25 mm pressure pane + 1 x 11.4 mm debris pane, resulting in 155.7 kg/m² of glazed surface. This resulted in a total window mass for the Habitat of 1641 kg + 20% design maturity margin.

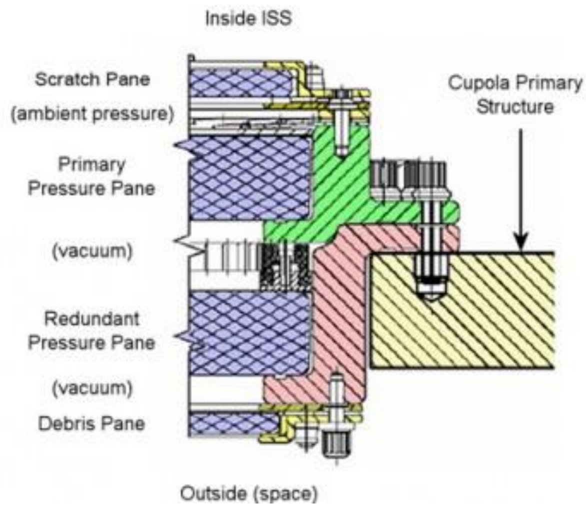


Figure 3-17: Schematic representation of the layer composition of the Cupola windows on board the ISS, and also assumed for the mass estimate of the Habitat design (source: RD[9])

Other elements that are not an integral part of the habitat, accommodated on the main structure, but that are anyway needed either to provide primary functions of the Habitat (Power generation, Thermal Control) or support to surface activities (Habitat deployment, EVA support) were also analysed and sized. These elements and the sizing underlying assumptions are discussed in the respective chapters.

- **Habitat Support Systems**: external systems deployed on the surface, directly connected to the Habitat, supporting its functions (namely power generation and thermal control), divided into:

- **Power Station**

A 59 kW Nuclear Fission Power Plant is the baselined option. Chapter 7.4.2 details the assumptions for this component.

		Mass (kg)
PWR	Power	6712.89
	Dry Mass Pwr_plant	**6712.89**

- **Thermal Surface Radiators**

A high radiator area required and low radiator inclination from the horizontal position led to baselining a solution in which radiators are deployed in the lunar surface. Chapter 9.2.2 details the assumptions for this component.

		Mass (kg)
TC	Thermal Control	5370
	Dry Mass Surf_Rad	**5370.00**

- **Airlock Module**

The additional Airlock Module needed to interface with the Habitat, the characteristics of the *Quest Joint Airlock* system were taken as a mass allocation. This is effectively a critical life support facility allowing the crew to re-acclimatise, and a similar system would likely be necessary for the habitat. The publically available system properties are:

- Length: 5.5 m
- Diameter: 4 m
- Mass: > 9000 kg

		Mass (kg)
SYE	Systems Engineering	9000.00
	Dry Mass Airlock	**9000.00**

- **Mobile Crane**

Details on the sizing assumptions taken for the Mobile Crane component are further detailed in Chapter 6.2.4.

		Mass (kg)
MEC	Mechanisms	13000.00
	Dry Mass Mob_Crane	**13000.00**

The Transfer vehicles (Tug and Lander) were considered as well. Their mass depends on the Launch Scenario, and they were sized as per Sections 3.5.3 – Tug Sizing and 3.5.2 - Lander Sizing, respectively.

3.6.4 Power Budget

The Power Budget takes into account the power demand for 3 basic System Modes that are assumed to include the sizing cases for Power across the different Mission Phases:

1. **Transfer** (Transfer)

During the transfer to the Moon, no habitat subsystems are assumed to be operating. However, survival heaters are needed to keep non-operational temperatures for equipment inside the habitat within allowable ranges. During this phase, without the support from the Power Station that is to support the Habitat in the surface, the required survival heater power is to be supplied by a local source or an habitat intrinsic power system. This can be achieved through a combination of Solar Arrays placed in the Habitat itself, a Habitat Service Module/Tug, or alternative options such as radioisotope heat sources, as discussed in Chapter 7.6.1.

2. **Nominal Operations during the Lunar Day** (Nom_Ops_Day)

   During Lunar Day Nominal Operations, a constant power supply is required not only to operate Life Support systems, but also other crew-related activities inside the Habitat. Additionally, it shall be noted that lighting, laptops and a 5 kW allocation for Science and Surface Operations are also assumed in the SYE power budget entry. The required power is to be supplied by an externally deployed Power Plant (several options discussed in the Power Chapter 7.4)

3. **Nominal Operations during the Night** (Nom_Ops_Night)

   During Lunar Night Nominal Operations, the same assumptions as for Day Operations were taken, except for a higher Habitat Lighting and Heater Power requirements.

The Budget is presented in Table 3-13 and includes a 20% system level margin.

Power Budget		**MODES**		
		Transfer	Nom_Ops_Day	Nom_Ops_Night
CREW (INS)		0	491*	491*
MEC		0	0	0
PWR		120	400	400
STR		0	0	0
SYE		0	6900	6950
TC		7200	0	2000
ECLS		0	40000	40000
Total		7320	47791	49841
System Margin	20%	1464	9558	9968
Total incl. Margin		**8784**	**57349**	**59809**

Table 3-13: Power Budget

*Due to an identified potential evolution in the power requirements for crew-related equipment and activities, an increase in Crew Accommodation power demand to ~2 kW when compared to the budgeted value is foreseen, as per the reported figures in the Crew Accommodation Chapter, despite this was not accounted for as the baseline in the Power Budget presented in the Systems and in the Power Chapters. This would have an approximate impact of a 1.8 kW increase in the Habitat's Power budget in the Nominal Operations modes, both during the day and night periods, and its impact in the subsystems sizing, such as the Power Plant and Thermal subsystem, shall be evaluated in later stages of development, despite a limited impact is expected.

3.6.5 Launch Baseline

As mentioned in Section 3.5.4 - Launch Scenarios, several possible scenarios were envisioned for such a mission.

The Scenarios that were assumed as baseline to this study were the SLS Scenarios (Section 3.5.4.1 - SLS Scenarios), due to their earlier assumed readiness for this mission.

From the Launch Scenarios presented before assuming the Space Launch System (SLS), Scenario B.2 was the best performing one for launching the Habitat and required Tug and Lander, and is therefore taken as baseline in this study to assess the launch feasibility of the revised Habitat design. The concept for the launch of the habitat, tug and lander in 3 different launches are depicted in Figure 3-18.

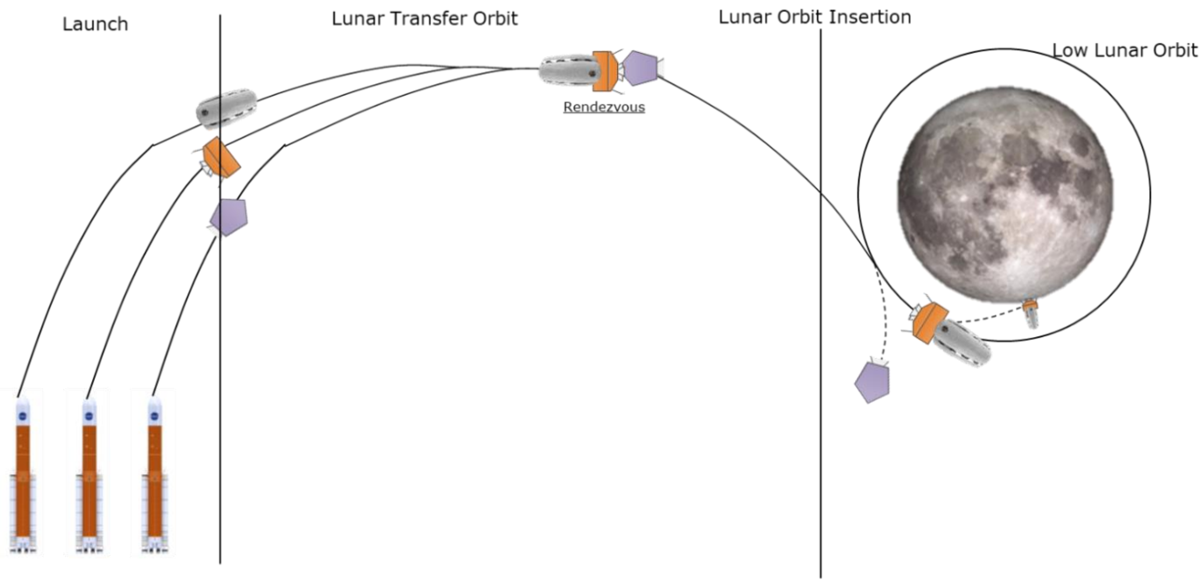


Figure 3-18: Representation of the launch, transfer and rendezvous of the Habitat, Tug and Lander, and subsequent delivery of the Habitat to the lunar surface

For this Scenario, the performance is now assessed against the assumed Habitat Baseline design achieved after the reassessment done in this study, along with the supporting mission elements.

Concerning the launch mass of the Habitat, in order to adapt the required launch mass to the Launch Scenarios previously considered, two options were considered. The first one considers the Habitat is launched complete and Wet, with all consumables and fluids required by the ECLS systems. Scaled down versions of the Habitat are shown as well. The case for the Half Habitat can also be interpreted as habitat module, from a Habitat split and delivered in two modules, to be assembled at the moon surface. However, this latter case is not considered viable with the current design of the Habitat, which was not conceived to be split and assembled; a re-design of the habitat would be necessary in that case.

Launch Option 1 - Wet, Full Habitat	
Total Wet Mass (kg)	**68173**
Launch Adapter (allocation)	1000
Launch Mass (Full and Wet) (kg)	**69173**
Full Habitat	69173
3/4 Habitat	52129
1/2 Habitat	35086
Launch Capability (SLS B.2 Scenario) (kg)	**28900**
Capability - Required (Full Hab)	-40273
Capability - Required (3/4 Hab)	-23229
Capability - Required (1/2 Hab)	-6186

Table 3-14: Wet Habitat Launch Mass – Option 1

As can be observed, this first option does not provide a feasible solution of launching even a half-habitat, with required launch mass being above the launch capability assumed.

A second option concerns the launch of the Habitat in dry condition and without the redundant ECLS systems. Besides the additional ECLS redundant systems, crew supplies and ECLS fluids (along with the respective tanks) are also to be supplied in a later launch.

Launch Option 2 - Dry, no ECLS redundancies	
Dry Mass incl. System Margin (kg)	**60505**
Redundant Systems (ECLS)	-3698
Fluid tanks (ECLS)	-1574
Launch Adapter (allocation)	1000
Launch Mass (Dry, No ECLS red.) (kg)	**56233**
Full Habitat	56233
3/4 Habitat	42424
1/2 Habitat	28616
Launch Capability (SLS B.2 Scenario) (kg)	**28900**
Capability - Required (Full Hab)	-27333
Capability - Required (3/4 Hab)	-13524
Capability - Required (1/2 Hab)	284

Table 3-15: Dry Habitat Launch Mass – Option 2

From the presented results, this second option could enable the delivery of the ½-scaled Habitat with an SLS Block 2. Three (3) launches would be required for the Habitat Delivery (or up to six if a full Habitat is to be delivered in halves). Delivery of support components could require as a minimum two (2) launches assuming a cargo tug is launched with the components (based on the performance of Scenario SLS C, as presented in Section 3.5.4.1 - SLS Scenarios), or up to 4 launches if reusability of a previously launched lander is not achieved. Lastly, a resupply mission consisting of at least one (1) launch, carrying fluids and crew supplies would also be required at the start of the mission and every 500 days. As an option, since the performance needs for this resupply mission are significantly lower than required for the missions to deliver the Habitat and main Support Components, the Resupply launch and delivery could be made with lighter launchers, potentially including already a smaller logistics lander that would deliver the cargo to the surface.

Table 3-16 summarises the assumptions for delivery of the Habitat and required components.

Mission	**1. Habitat Delivery**			**2. Support Components Delivery**			**3. Resupply**	
Launch Number	1.1	1.2	1.3	2.1	2.2	2.3	3.1	3.2
Reference Scenario	SLS B.2	SLS B.2	SLS B.2	SLS C	SLS C	SLS C	SLS C (or optimized solution)	SLS C (or optimized solution)
Launcher	SLS Block 2	SLS Block 2	SLS Block 2	SLS Block 2	SLS Block 2	SLS Block 2	SLS Block 2 (or lower)	SLS Block 2 (or lower)
Status	Mandatory	Mandatory	Mandatory	Mandatory	Mandatory	Optional (depending on reusability)	Mandatory	Optional (depending on reusability or potential combination of lander + tug + cargo into 1 launch)
Component	- Half Habitat	-Tug	-Lander	Tug, with: - Surface Radiators - Mobile Crane	Tug, with: - Power Station -ECLS Redundancies -Airlock	-Lander	Tug, with: - ECLS Fluids - Crew Supplies	-Lander
Payload Mass (10^3 kg)	28.6	N/A	N/A	18.4	19.4	N/A	8.9	N/A
Launch Mass (10^3 kg)	28.6	42.7	45.8	33.3 (14.9 Wet Tug, 18.4Payload)	34.3 (14.9 Wet Tug, 19.4 Payload)	45.8	8.9	45.8 (potentially lower)
Number of launches per deployed Habitat (no lander/tug reusability)	- 1x if half-scaled Habitat - 2x if Full Habitat split into 2	- 1x if half-scaled Habitat - 2x if Full Habitat	- 1x if half-scaled Habitat - 2x if Full Habitat	1x	1x	2x	1x / 500 days	1x / 500 days
Number of launches per deployed Habitat (lander and tug reusability, no refuel launches assumed)	- 1x if half-scaled Habitat - 2x if Full Habitat split into 2	1x for either half-scaled or full Habitat	1x for either half-scaled or full Habitat	1x	1x	0x	1x / 500 days	0x

Table 3-16: Baseline scenario SLS B2

3.7 System Options

3.7.1.1 Launch Option: Starship Scenario

As an option, and due to recent developments in the Commercial Launcher market, SpaceX's Starship was also considered as a potential scenario.

With the assumptions of:

1. Performance of 100 metric tons to the moon surface RD[12]
2. Full Launcher Vehicle lands at the moon surface
3. Launcher capable of deploying (hoisting) the Habitat and cargo to the surface

This Scenario assumes SpaceX Starship will be available at the mission timeline, and that a 100 metric tons capability to the Moon Surface is achieved, as per the Starship Users Guide (Revision 1.0, March 2020) RD[12]. The launch could take habitats directly to the surface, thus assuming the Starship inserts itself with the payload into lunar orbit, lands and hoists the cargo onto the lunar surface, with a crane system is assumed to be available and included in the Starship dry mass.

Assumptions	
1	Payload to Lunar Surface (assuming in orbit refuels) is at least 100 metric tons RD[12]
2	Payload to LEO Orbit is at least 100 metric tons RD[12]
3	Dry Starship Mass assumed at 120 metric tons (TBD)
4	Propellant Refuel Mass up to 1200 metric tons each (TBD)
5	Delta-V assumed as 4 km/s from LEO to LLO, 2 km/s from LLO to the Lunar Surface (and visa-versa), and 800 m/s from LLO to Earth Return Orbit
6	ISP assumed at 310 s

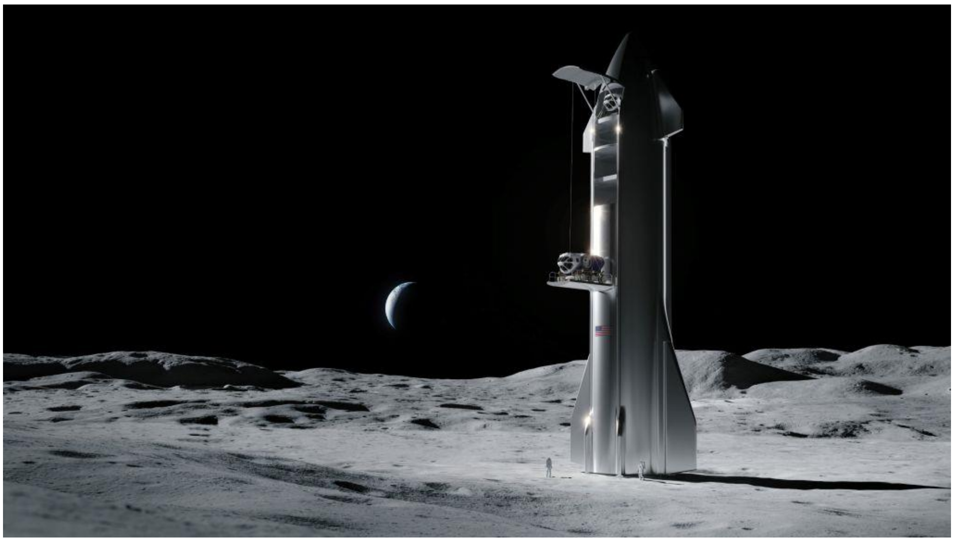


Figure 3-19: Image of Starship scenario (Source: SpaceX RD[12])

For this scenario, it is assumed as well that the Starship is refuelled with propellant twice (one in LEO, one in LLO before descent). These 2 refuels might require several refuelling tank launches themselves. It is however important to note that there is very little performance and operations information available at the time this assessment was performed, which is why several assumptions had to be made to perform a rough check of how a habitat delivery with the Spacex's Starship could be achieved.

	Mass (10^3 kg)
Payload in LEO	100
Starship dry	120
Refuel mass	1200
Mass in LEO after refueling	1420
Mass in LLO	381
Mass in LLO after refueling	1420
mass on surface	735
dry mass lander	120
payload mass on Moon	**100**
Return trip	
Mass in LLO (after starship lifts off again)	329
Mass in Earth Return Orbit	253

Table 3-17: Rough Launch performance check for the Starship Scenario (based on available information - RD[12]- and rough assumptions)

With the self-landing capability at the moon surface, this could imply that the use of Tugs and dedicated Lander would not be needed. Additionally, a 100 metric ton capability could become sufficient for the delivery of a full habitat with all supplies for the first 500 day mission, along with all surface equipment that support the Habitat and its deployment considered in this study in a single launch (excluding any required refuelling launches for the Starship, to be determined). However, a negative margin was still determined.

Starship Launch Option	
Habitat Total Wet Mass	68173
Power Station	6713
Thermal Surface Radiators	4721
Airlock Module	9000
Mobile Crane	13000
Launch Adapter (allocation)	1000
Launch Mass (kg)	**102607**
Assumed StarShip Performance (kg)	**100000**
Potential Launch Performance Margin	-2607

Table 3-18: Launch Mass (all mission components) – Starship option

Assuming this performance and assumptions are met, further work shall be needed to make this option feasible, including (but not limited to) the following:

- Accommodation of Habitat and other support equipment in the payload fairing (potentially requiring complex structures for compliance with launcher loads)
- Launcher capability to provide the required resources (thermal environment and power for survival heaters)
- Precise landing capability and impact of large structure landing in the stability of the local surface in potentially unprepared landing pads, and potential issue with dust ejection during landing and return launches.

3.8 Technology Needs

Included in this table are:

- Technologies to be (further) developed
- Technologies identified as coming from outside ESA member states
- Technologies available within European non-space sector(s)

Technology Needs						
*****	**Equipment Name & Text Reference**	**Technology**	**Supplier (Country)**	**TRL**	**Funded by**	**Additional Information**
*	High Mass Cargo Lander	30 to 60+ metric tons payload capability to the moon	None	New development		No known development targeting such high mass
	Logistics Lander	4 to 8+ metric tons payload capability	Several Commercial (Europe, USA) and Institutional (ESA) efforts	Low		Institutional example is the ESA's EL3 (although targeting lower payload capability). Commercial examples include development under the Commercial Lunar Payload Services NASA programmer, as well as iSpace, PTS Alina and ILL in Europe (for much lower payloads.
*	Tug	30 to 60+ metric tons payload capability		Low		No known development for required payload capability. MPCV ESM is under production and CLTV in pre-Phase A, both with lower payload mass.
*	SLS		NASA (USA)	High	USA	SLS first launch target 2021
	100+ ton payload launcher			Low		SpaceX Starship in early phases of development

* Tick if technology is baselined

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4 CREW ACCOMMODATION

4.1 Challenges and Needs Within the Concept of Operations

For the crew accommodation subsystem, the following challenges and needs have been identified within the concept of operations:

4.1.1 Testing

To reduce human error, increase productivity, and enhance safety and comfort of the crew, human factors of all technology interfaces and equipment in the crew accommodation subsystem have to be tested and the crew has to be familiar with the environment that they will be using during their stay on the lunar surface.

4.1.2 Inflation and Deployment of the Habitat

If the crew is already on site when the habitat inflation and deployment takes place, and depending on the duration of this process, alternative accommodation for the crew might be necessary.

In the context of a lunar human exploration mission, one or more pressurised rovers might be on site for surface mobility and exploration. If such (a) pressurised rover(s) is/are available, it/they could be used to accommodate the crew for the period of time during which the habitat is deployed.

4.1.3 Outfitting and Commissioning of the Habitat

Depending on the duration and complexity of the habitat outfitting and commissioning operations, alternative accommodation for the crew might be necessary.

As already mentioned, (a) pressurised rover(s) might be used to temporarily accommodate the crew. However, if considered early enough in the planning of the crew accommodation layout, a minimum deployment configuration could be implemented that allows for accommodation of the crew in the habitat even during outfitting and commissioning operations.

4.1.4 Minimum Accommodation Functionalities

The crew habitat should be designed to support a crew of four, staying on the lunar surface for up to 500 days. As a minimum, the following functionalities need to be available in the crew accommodation:

- Sleeping space, ideally private quarters
- Dining and communal activities
- Work space
- Exercise area and equipment
- EVA suit donning and doffing
- Medical care
- Hygiene
- Translation portals or pass-throughs

- Stowage access.

4.1.5 Crew Accommodation Requirements

Recommendations for net habitable volume depend on functions required of the mission, crew size and mission duration. 25 m^3 net habitable volume per person should be considered the absolute minimum for deep space habitats. However, this number is significantly smaller than the minimum net habitable volume of the ISS (85.17 m^3), and older stations like Skylab (120.33 m^3), Mir (45 m^3) and Salyut (33.5 m^3) which all have or had shorter mission durations than 500 days. Given the varied functions necessary in the habitat, the crew size and the long duration of the surface mission (500 days) a minimum net habitable volume of about 80 m^3 per person is recommended (RD[13], RD[14]). However, the Moon Village habitat is going to be deployed on the lunar surface only. This allows for the habitat to be transported in a folded state, using significantly less volume.

The structure and outfitting system of the habitat has to maximise the usability of the habitable volume provided to the crew. The module needs to be highly volume-efficient and designed to optimise habitability. It is recommended to design for flexibility and reconfigurability (e.g. for varying mission goals, different crews, future base development) of the habitat to save costs and time over operational lifetime. The interfaces of equipment and outfitting in the accommodation subsystem need to be easy to use.

4.1.6 Partial Gravity Challenges

With regards to human locomotion within a habitat in lunar gravity, several differences need to be considered (RD[15], RD[16]):

- Walking (slower)
- Running (slower, tendency to slip)
- Jumping (higher and farther)
- Loping (most comfortable in partial g)
- Posture (forward inclination)
- Traction (reduced – balance and locomotion hazardous).

Partial gravity may allow greater access to places that would not be accessible in Earth gravity, such as ceilings. However, it is important that the heights of the ceilings in the habitat are set at a height that ensures that no inadvertent contact arises from any locomotion. Ceiling heights should be at least 2.5 m.

Ladders take up less space than stairways, but they might pose problems with safe translation between levels when injured or carrying items. In case a ladder is used for vertical translation of the crew, options to move items without them being carried (such as a small elevator) should be provided. The stairs should have an inclination, ideally between 67°and 78° (RD[17]).

Activities in lunar gravity will occasionally require that astronauts can secure themselves in position. This means, restraints and rails need to be available, and located so they can be easily and safely accessed.

4.2 Baseline Solution

The baseline solution is presented by SOM's Moon Village habitat design.

4.3 Budgets

4.3.1 Galley and Food System

The galley has to provide facilities for food preparation during the 500 days of surface stay. A sink and a spigot that supplies hot and cold drinking water and can be used to rehydrate and warm up freeze-dried food are the bare minimal cooking facilities of the galley. However, due to the long mission duration, the galley should provide an oven, freezer, dishwasher and cooking supplies as well. The equipment will be integrated and supplies will be stowed in stowage racks located in the wardroom and galley area. Ample stowage space should be provided for food stowage and cooking and eating supplies. The crew should have sufficient space to congregate around a table, to eat together, enjoy social conversation, and hold crew meetings (RD[18]).

Based on recommendations by Stilwell et al. (RD[19]), Table 4-1 shows the mass and volume budgets for the galley and food system, not including life support system elements.

Galley and Food System	Mass	Mass Subtotal (kg)	Volume	Volume Subtotal (m^3)
*** Habitat**				
Oven/microwave oven	50 kg	50	0.3 m^3	0.3
Freezers	100 kg	100	0.5 m^3	0.5
Sink, spigot for hydration of food and drinking water	15 kg	15	0.0135 m^3	0.0135
Dishwasher	40 kg	40	0.56 m^3	0.56
1 Rack (ISPR)	104 kg	104	1.571 m^3 (internal volume)	
*** Cargo delivery**				
Cooking/eating supplies	2 kg/p	8	0.0056 m^3/p	0.0224

Table 4-1: Mass and volume budget of the galley and food system

4.3.2 Waste Collection and Hygiene

Body waste management, body cleansing, oral hygiene and grooming will be performed in the hygiene area. For olfactory and auditory separation, the toilet should be located away from the galley and feature an airtight door. Enough stowage racks should be available for storing hygiene supplies (household wipes, disinfectant, vacuums and other housekeeping equipment) and astronaut's personal hygiene kits (RD[18], RD[19]).

Based on recommendations by Stilwell et al. (RD[19]), Table 4-2 shows the necessary mass and volume for the waste collection and hygiene area, not including life support system elements.

Waste Collection and Hygiene	Mass	Mass Subtotal (kg)	Volume	Volume Subtotal (m^3)	Mass Margin (%)
*** Habitat**					
Vacuum	2 x 4 kg	8	2 x 0.02 m^3	0.0400	5
2 Racks (ISPR)	104 kg/rack	208	1.571 m^3 (internal volume/rack)		5
*** Cargo delivery**					
Hygiene supplies (consumables)	0.075 kg/p/d	150	0.0015 m^3/p/d	3.0000	5
Personal hygiene kit	1.8 kg/p	7.2	0.005 m^3/p	0.0200	5

Table 4-2: Mass and volume budget of the waste collection and hygiene system (excluding life support elements)

4.3.3 Sleep Accommodation, Health and Clothing

Each crew member is accommodated in a private quarter for sleeping, personal office, private conference and personal recreation. Those quarters have to provide optimal noise protection and personalised air conditioning and illumination control to provide comfortable living conditions. Various usage scenarios for the crew quarters should be provided for (e.g. sleep, work, conference, single or double). The crew quarters will be used during the entire mission of 500 days. Therefore it is particularly important that sufficient space is available for sleeping, dressing, working, and recreational activities and that the environment supports those activities with the necessary equipment (personal stowage, foldable desk, etc.). Surfaces in the crew quarters should be easy to clean, yet haptically pleasant. Astronauts will spend much of their time in the crew quarters, therefore it is of particular importance to shield those quarters from harmful ionizing radiation.

The habitat should be equipped with a washing machine and dryer to save mass and volume for the mission. Mass and volume estimations for providing fresh clothes without a washer and dryer during a 500 day mission amount to 920 kg and 3.04 m^3. A washing machine and clothes dryer including detergents would likely amount to 100-160 kg, and requires only 18.4 kg and 0.06 m^3 of clothes. Hence, about 800 kg mass and up to 2.2 m^3 of volume could be saved. The additional equipment will, however, create additional complexity to the life support, power and thermal control systems.

For astronauts to stay healthy in partial gravity conditions and isolation, exercising will be vital. Hence, equipment (e.g. ergometer, treadmill, elastic bands) need to be provided. Virtual/augmented reality sets could be used to give variety and increase the recreational value of exercising.

Due to the long duration of the mission and expected further development of the initial base, it is important to provide sufficient medical supplies. Medical equipment and supplies must be easily and safely accessible to the crew at all times. A medical/surgical/dental suite should be considered, in spite of its relatively high mass. Alternatively, at least a portable medical kit and medical supplies need to be available and surgical instruments might be 3D printed on board (see section 4.3.4 Operational supplies and maintenance) in an emergency.

Given the extreme physical environment and the relatively long mission duration, chances are that accidents and injuries might happen to the crew. Therefore, the layout of the habitat should be designed in a way that provides basic functionalities to a crew member who is temporarily or permanently incapacitated. This means, even with a sprained or fractured extremity, all important areas of the habitat should still be reachable.

The habitat needs to provide a “safe haven zone” which is located in an area of the habitat that is particularly protected from ionizing radiation. This safe haven will be used during solar particle events (SPEs) and needs to provide sufficient and easily and safely accessible supplies (i.e. food, hygiene, medical) and facilities (e.g. hygiene, sleeping, communication, medical) for the whole crew and the duration of such events.

Based on recommendations by Stilwell et al. (RD[19]), Table 4-3 shows the mass and volume budgets for sleep accommodation, health and clothing.

Sleep Accommodation, Health and Clothing	**Mass**	**Mass Subtotal (kg)**	**Volume**	**Volume Subtotal (m^3)**
*** Habitat**				
Private crew quarters (basic outfitting: bed and foldable desk)	100 kg/p	400	>2.5 m^3/p	10
Washing machine and dryer	100-160 kg	100-160	0.75-1.5 m^3	0.75-1.5
Medical/surgical/dental suite (TBD)	500 kg	500	2.00 m^3	2
4 Rack (ISPR)	104 kg	416	1.571 m^3 (internal volume)	
*** Cargo delivery**				
Personal stowage and recreational equipment	25 kg/p	100	0.38 m^3/p	1.5200
Clothing	4.6 kg/p	18.4	0.0033 m^3/kg	0.0610
Exercise equipment	145-300 kg	145-300	0.19 m^3	0.1900
Medical consumables	250 kg	250	1.30 m^3	1.300

Table 4-3: Mass and volume budget of the sleep accommodation and clothing system

4.3.4 Operational Supplies and Maintenance

The crew will have to perform occasional repairs and maintenance work. Maintenance equipment and operational supplies will be stowed or integrated in the stowage racks. A workstation dedicated to small repairs, testing, etc. should be available in the habitat. Hand tools, large machine tools and test equipment are accounted for with an estimate of 1100 kg. To cut down mass for maintenance tools and consumables, a 3D-printer could be used to manufacture tools and spare parts.

Based on recommendations by Stilwel et al. (RD[19]), Table 4-4 gives the mass and volume estimates for operational supplies and maintenance.

Operational Supplies and Maintenance	Mass	Mass Subtotal (kg)	Volume	Volume Subtotal (m^3)	Mass Margin (%)
*** Habitat**					
Restraints and mobility aids	50 kg	50	0.27 m^3/kg	13.5	5
3 Racks (ISPR)	104 kg/rack	312	1.571 m^3 (internal volume/rack)		5
*** Cargo delivery**					
Operational supplies (velcro, tape, ziplocks, etc.)	20 kg/p	80	0.002 m^3/p	0.008	5
Hand tools and accessories	200 kg	200	0.066 m^3	0.66	5
Spare parts/equipment & consumables	TBD				
Fixtures, large machine tools, gloveboxes, etc.	600 kg	600	3 m^3	3	5
Test equipment (oscilloscopes, gauges, etc.)	300 kg	300	0.9 m^3	0.9	5

Table 4-4: Mass and volume budget of the operational supplies and maintenance system for all repairs in habitable areas

4.3.5 Airlocks

It is assumed that extravehicular activity will play an important role during the 500 days of mission. The mission crew will likely have to perform extravehicular activities (EVAs) on a regular basis (i.e. about once every 6 days). To increase flexibility in mission goals and further development of a lunar base, it is recommended that the airlock should be a separate unit and not integrated in the habitat. The airlock units (two airlocks expected, for redundancy) can be sent separately from the habitat.

Preliminary mass and volume estimates for long duration mission (>180 days) airlocks are 1000-1500 kg and a total pressurised volume of 5-10 m^3 for all (RD[13], RD[20]). The EVA equipment (3 suits and consumables for 90 EVAs) will have an estimated mass of about 810-1086 kg and a volume of about 3 m^3 (RD[20]).

4.3.6 Power Requirements

Eckart et al. suggest that preliminary surface base power requirements for missions longer than 180 days demand at least 10 kW/person for the habitation area. An airlock for EVAs requires >10 kW/unit. Once the surface base is mature enough to feature a full laboratory, another 15 kW/person minimum would be required (RD[13]).

Based on recommendations by Stilwell et al. (RD[19]), Table 4-5 shows the power consumption estimates for electrical hardware for a 500-day mission in a lunar surface habitat.

Power Consumption of Crew Accommodation Hardware	**Average Power (kW)**	**Powered Time (% of a day)**	**Energy (kWh)**
Galley and Food System			
Microwave ovens	1.80	6	1296
Freezers	1.40	100	16800
Dishwasher	1.20	8	1152
Waste Collection System and Hygiene			
Vacuum	0.40	1.00	48
Crew Quarters, Clothing, Health			
Washing machine & clothes dryer	4.00	8.00	3840
Personal stowage	0.70	4.00	336
Exercise equipment	0.15	50.00	870
Medical/surgical/dental suite	1.50	1.00	180
Operational Maintenance			
Fixtures, large machine tools, gloveboxes, etc.	1.00	0.10	12
Test equipment (oscilloscopes, gauges, etc.)	1.00	0.10	12
TOTAL (kWh)			**24546**

Table 4-5: Power consumption budget of the electrical hardware in the crew accommodation

4.3.7 Total Budgets Over All Crew Accommodation Elements

Based on recommendations by Stilwell et al. (RD[19]), Table 4-6 shows the total budget estimates for a 500-day mission in a lunar surface habitat.

Total Budgets	Total Mass incl. Margins (kg)	Total Volume (m^3)	Total Energy (kWh)
Galley and food system (excl. life support elements)	**346.35**	**1.3959**	**19248**
- Habitat	337.95	1.3735	
- Cargo delivery	8.4	0.0224	
Waste collection and hygiene (excl. life support elements)	**391.86**	**3.0600**	**48**
- Habitat	226.80	0.0400	
- Cargo delivery	165.06	3.0200	
Sleep accommodation and clothing, health	**2125.87**	**15.821**	**5226**
- Habitat	1586.80	12.7500	
- Cargo delivery	539.07	3.0710	
Operational supplies and maintenance (all repairs in habitable areas)	**1619.10**	**18.068**	**24**
- Habitat	380.10	13.5000	
- Cargo delivery	1239.00	4.5680	
TOTAL	**2531.65 (Hab. Equipment) + 1951.53 (Cargo)**	**38.3449**	**24546**

Table 4-6: Total mass, volume and power consumption budget of the crew accommodation

4.4 Technology Needs

The selected crew accommodation strategy recommends further technology development in the following elements:

- Crew quarters envelope material:
  - The following properties need to be improved to provide sufficient comfort during long duration missions: ultra-light, excellent sound insulation, easy to clean, haptic characteristics to increase comfort, allowing for optimal technology integration (plug-in)
- Cooking oven:
  - Since the Space Shuttle missions, forced air convection ovens for habitats in LEO are available (RD[21]). Adaptation of such a device for the lunar environment (i.e. gravity, human factors) is necessary
- Dishwasher:

    - Developing a dishwasher for use in lunar gravity compatible with the habitat's LSS is required
- Washing machine and dryer:
    - An "Advanced Microgravity Compatible, Integrated Laundry" (AMCIL) prototype, a microgravity compatible liquid / liquid vapour, two-phase laundry system with water jet agitation and vacuum assisted drying was developed by Umpqua Research Company under a research contract with NASA (RD[22]). Further development to adapt such a system to lunar gravity and the habitat's LSS is necessary.

All those aspects should be taken into account to define the Technology Roadmap.

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5 CONFIGURATION - STRUCTURES

5.1 Challenges and Needs Within the Concept of Operations

Various challenges need to be addressed relating to the design of habitation modules to be manufactured, tested, transported and installed at a site on the moon as part of a village on the moon. These challenges will present conflicting requirements for the elements needed to build the village on the moon using building blocks. The points identified in the CONOPS list already provide an insight and need to be studied to balance the requirements.

The fact that early habitation, research or other modules are foreseen to be manufactured and assembled on Earth, result in some of the following configuration issues at different ends of the spectrum needing to be addressed:

1. A completely outfitted module. A module that can be manufactured, tested and brought to destination to be installed in a "plug-and-play" concept, will be completely outfitted and ready to go. Apart from some elements (as the SOM concept shows) that will be inflated at location, everything will need to be included. This does not necessarily mean that all consumables are present for the foreseen mission(s), but certainly all exterior and interior elements to start the mission as soon as the module reaches its destination. This will directly influence the balance of materials and building blocks needed to assemble the module. And the mass budget will constrain the module and its balance between elements (for obvious reasons of limitations during testing, transport and installation). The advantage, if managed, is that no further effort is needed to bring elements into the module and build the interior at destination. The interior may still have to be stowed and installed at arrival, since centre of mass location for launch and transport may drive the location of all these elements inside the overall structure and volume.
2. A "shell module" to be outfitted after installation. When this approach is taken, a basic shell structure with the necessary parts to start the module are manufactured, tested, launched and then transported and installed on the moon. The obvious advantage is that in this case the mass will be limited to the essential elements that make up the module to be used. Testing and transportation will also be less complex and limited by available infrastructure. However, after installation, all internal elements will have to be installed, and the module outfitted for its intended use. This will require not only additional logistics, but also adequate external access to the module and environment for the crew that will have to outfit the module for its intended purpose.

The points above illustrate the consequences for the configuration based on the design choices linked to the logistics of the building phase of the Moon Village before the modules will actually be used.

This will then also have to be translated to the structural concept and ensure the structural integrity of the module. When the module needs to be outfitted on the moon, access for various internal parts may require larger or dedicated hatches, driving the configuration and structural design.

The configuration presented by SOM, is a large conceptual module, the design of which can potentially be oriented towards either point above.

5.2 Baseline Design

The baseline conceptual design as proposed by SOM is shown in Figure 5-1. The left shows a possible compact launch configuration, while the right shows the inflated module as it is foreseen to be installed at the Moon Village.

Figure 5-1: Conceptual Design by SOM (external)

In Figure 5-2 some of the core interior can be seen through two cut-outs.

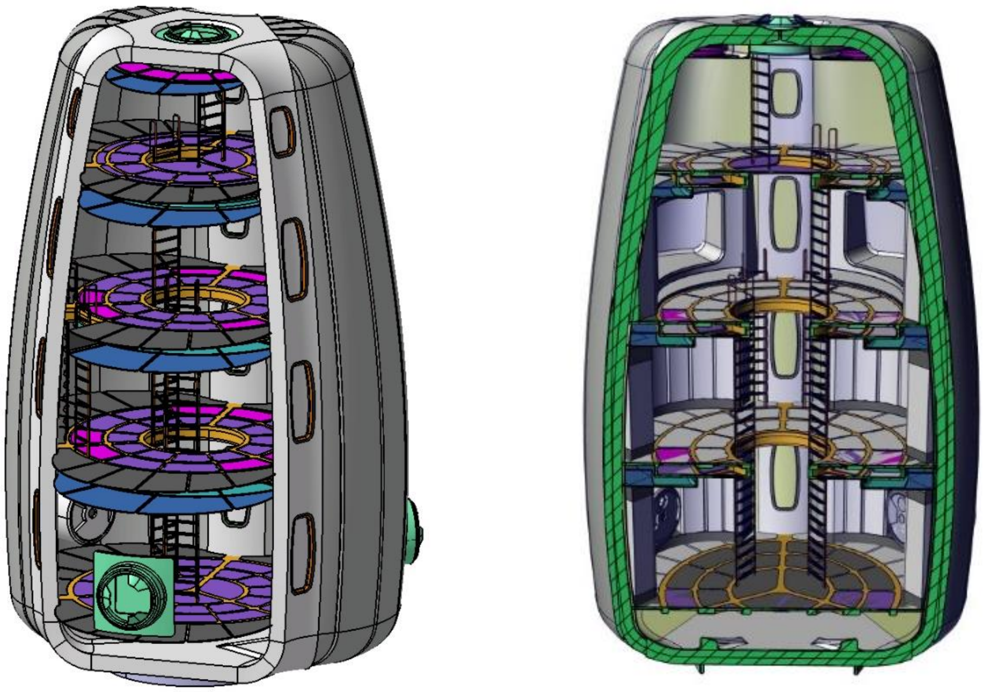

Figure 5-2: Conceptual Design by SOM (internal)

The internal design shows four levels, floors, which can be divided into living, research and logistics areas.

For a further specification of the different elements, Figure 5-3 first shows the main structure with the floors, windows and access hatches. Next to it (in the middle) different volumes shaped to fit the overall internal volume and serve as living quarters, storage, racks for computers and environmental control systems, and finally research stations. The image to the right shows the flexible shells, that will be deployed (inflated) after the module has been installed at its final destination.

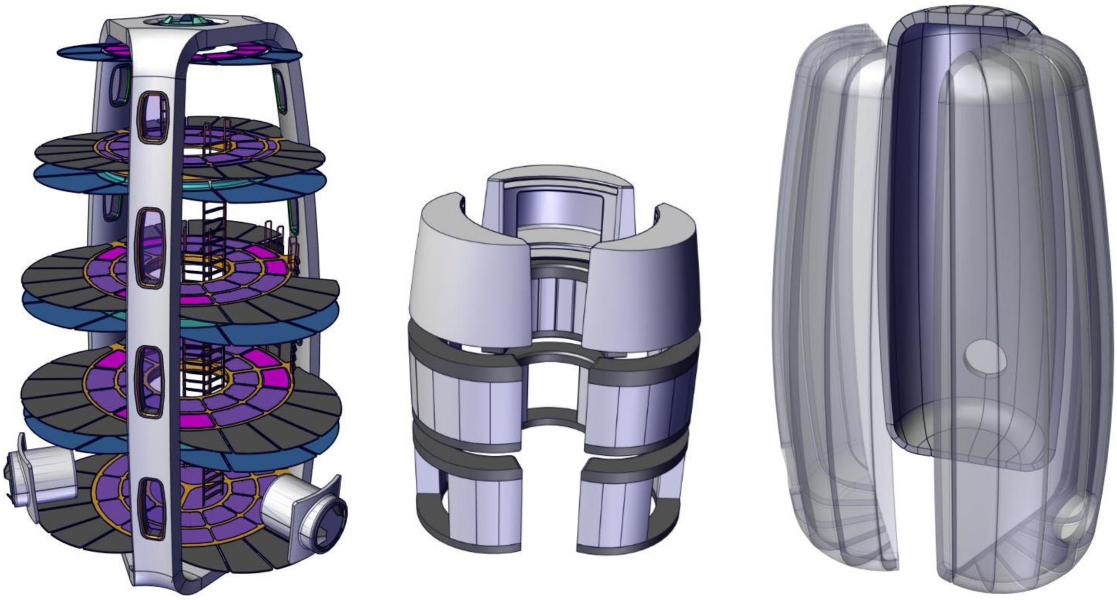

Figure 5-3: Conceptual Design by SOM, 3 main parts

Starting from the assumption that the module, completely being outfitted with internal equipment, is to be launched into orbit, transferred to the moon and then put on the surface of the moon. The first step will be to identify the launcher capable of launching the module. For the assessment of the system mass and what is required chapter 3.4, ofthe Systems chapter, will provide insight and strategy on what is possible.

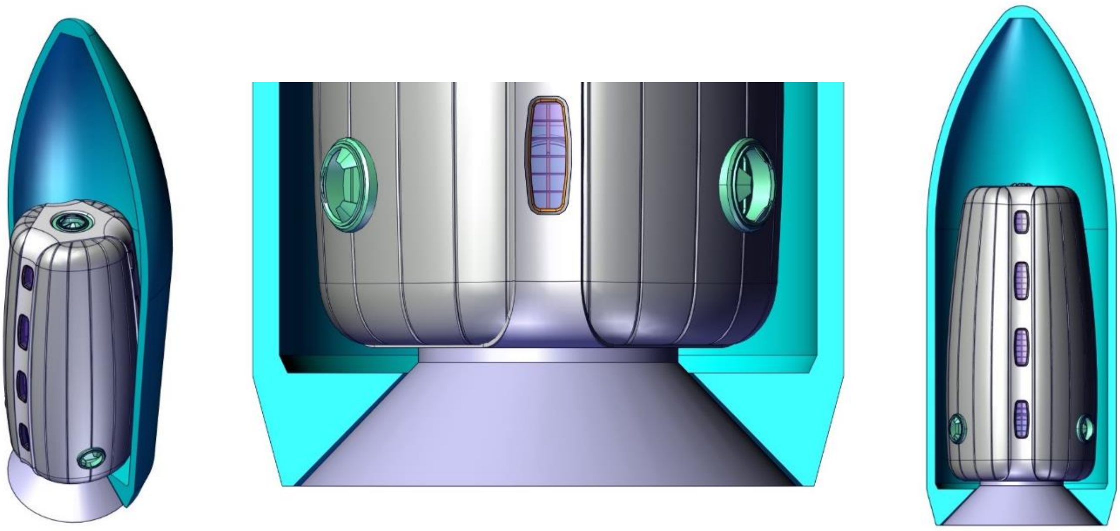

Figure 5-4: The module inside the SLS fairing

At this time, the only available launcher which has a launch fairing planned to contain the habitation module is the SLS launcher by the United States RD[23]. Figure 5-4 shows the module in stowed shape (flexible elements deflated and strapped down). This will require an interface ring on the structure of the module, to interface to the Launcher adapter. The selected adapter is the 4394H, H for Heavy to support the relative high launch mass of the payload. The interface is therefore dimensioned at a 4394 mm diameter, which provides a wide "base", advisable from a stiffness and stability point of view and spreading of the interface loads.

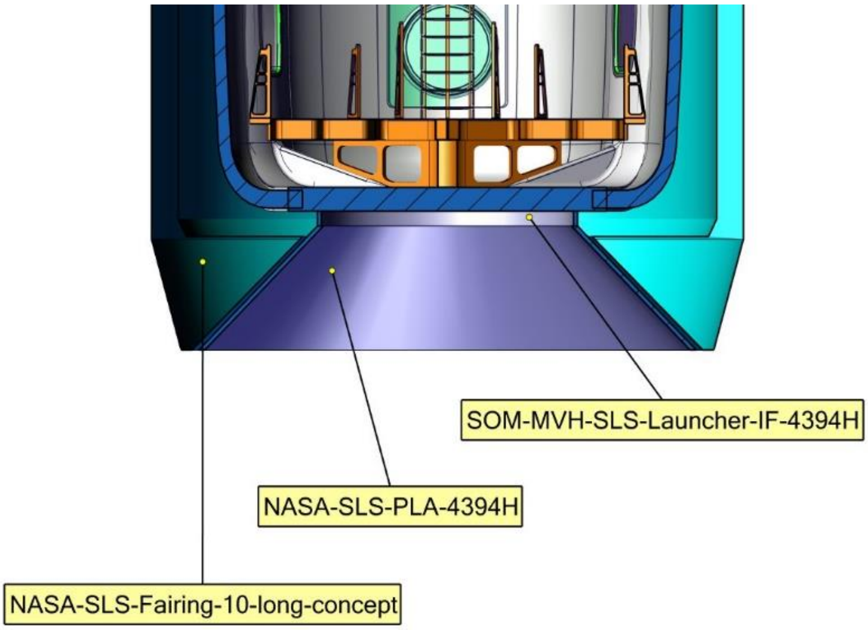

Figure 5-5: Module to Launcher interface

The internal diameter of the fairing is large enough to have the module inside with sufficient clearance to the wall of the fairing (see Figure 5-6).

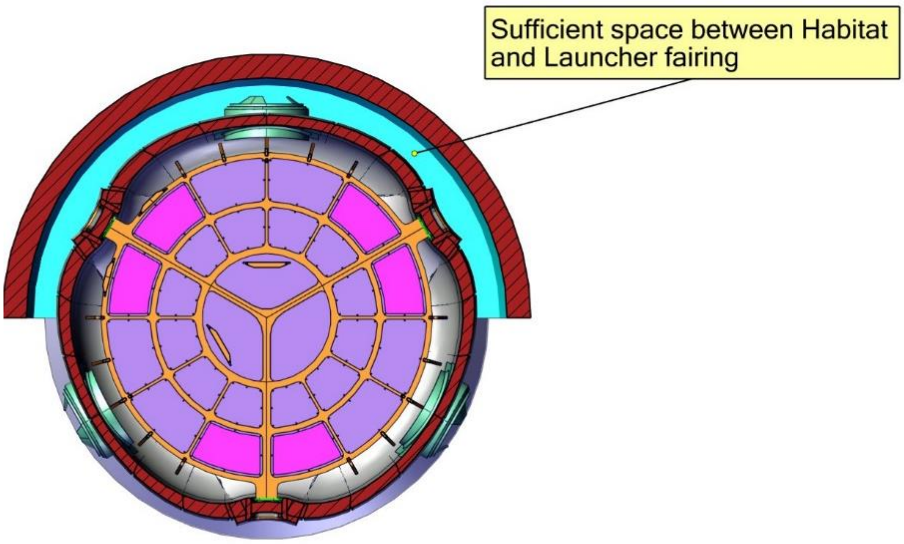

Figure 5-6: Clearance to the fairing

An important consequence of the design choice to deploy elements of the module, flexible shell – access adapters – floor elements, is that these will have to be properly latched down for launch. Especially the flexible shell will require proper handling and support, since wear and tear during launch could have detrimental effects on the function of the shell when it is deployed on the moon. Figure 5-7 indicates the necessary precautions for the launch and transport phases.

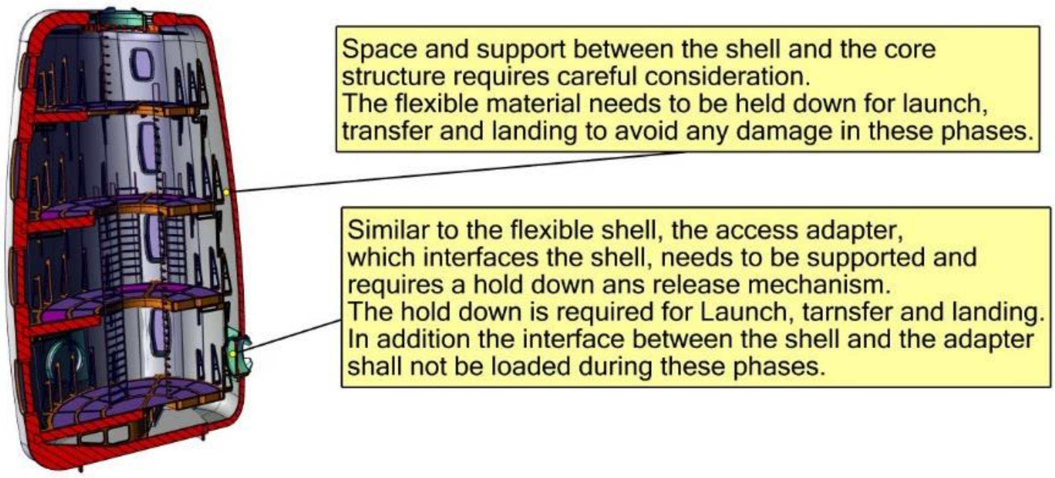

Figure 5-7: Stowage requirements

The main load-carrying structure will be supporting the module and its sub-systems during launch and transportation. Starting at the launcher interface ring indicated in Figure 5-5, all other parts are interconnected by the external three legged-main structure (to the right), and the floor elements that act as intermediate connections.

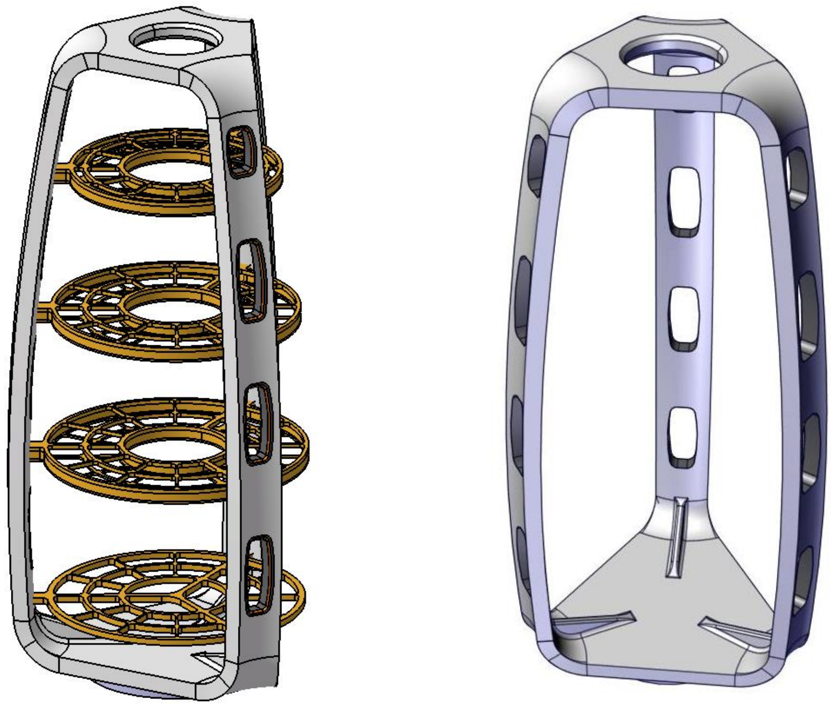

Figure 5-8: The main load-carrying structure

Structural analysis will have to show that the structure provides the required launcher stiffness, stability and strength for the functions the structure will have to perform during it life time. This will depend on the detailed design of the elements of the main structure. At the time of the study, and in the short amount of sessions for the study, these details were not available to be assessed. It is highly recommended to prepare a conceptual design with sufficient detail to assess the compliance to the requirements. This will result in a more precise estimate of the structural mass.

After the analysis it will also be important to establish the way the module will be launched and transfer into lunar orbit. In case there is a sufficiently large mass budget available, the module can be launched combined with a transfer module. This transfer module will then take the habitat module to a suitable lunar orbit. This transfer module would fit between the launcher adapter interface and the interface ring on the habitation module.

At this time, it seems unlikely that the overall payload launched by the SLS will include a complete landing module. This would then be required as a separate module. In addition it will need to interface and "connect" in lunar orbit in a clean way. Then the habitation module can be landed with the help of that module on the moon. The next step would be to disconnect this module.

For transport on the moon there will be the need for an interface for ground transportation equipment. Ideally this should be with already existing interfaces to reduce additional links and mass for those links.

The last point will be a secure location to install the module in its final location on the moon. This could be done with a clamping device using the already existing launcher interface ring. After installation the module will have to be made ready for operations. This may require the installation of internal elements that were too heavy to be included in launch and transport, not just for the total mass, but also for the centre of mass (regarding stability at transportation).

Outfitting will then complete the module for operations, an example shown in Figure 5-9.

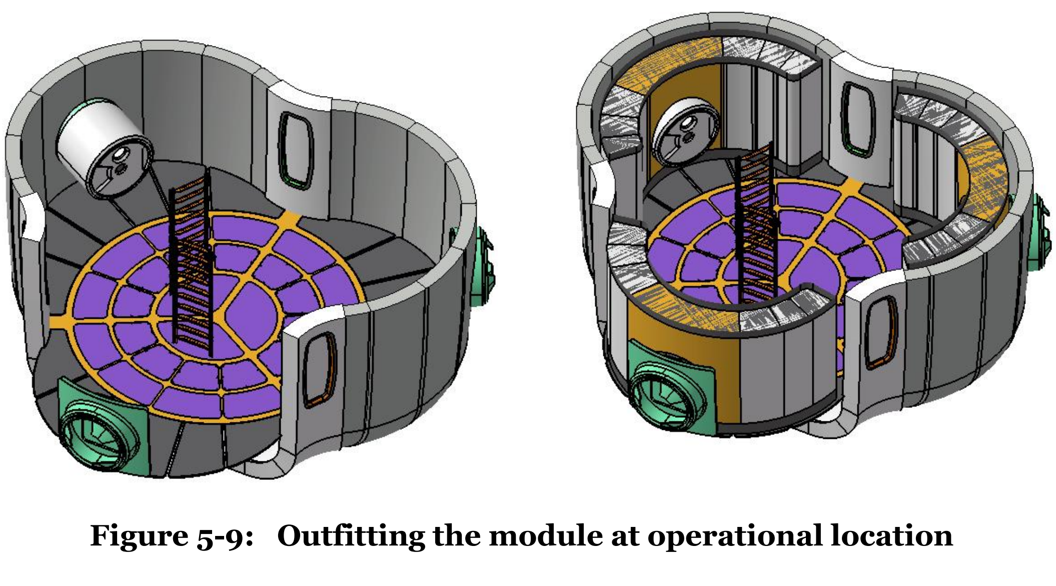

Figure 5-9: Outfitting the module at operational location

5.3 Budgets

The mass budget provided by SOM (Table 5-1) was taken for most structure components, since the detailing of the design was not known during the study sessions.

ACCESSIBILITY	649.50	KG	Aluminum 6061	2700	kg/m3
CEILING PANELS	1,023.00	KG	PMMA Plastic	1188	kg/m3
CENTRAL FLOOR PANELS	772.40	KG	CFRP (Carbon Fiber)	1415	kg/m3
EXTENDED CEILING PANELS	272.10	KG	PET Plastic	1541	kg/m3
GLAZING	251.90	KG	Polycarbonate, Clear	1200	kg/m3
EXTENDED FLOOR PANELS	308.60	KG	PET Plastic	1541	kg/m3
SECONDARY FLOOR STRUCTURE	4,435.30	KG	Aluminum 6061	2700	kg/m3
EXTERIOR FRAME	5,072.00	KG	Aluminum 6061	2727	kg/m3
EXTERIOR FRAME INTERFACE	131.90	KG	Aluminum 6061	2842	kg/m3
WINDOW FRAMES	259.60	KG	Aluminum 6061	2968	kg/m3
ADAPTER	3,439.00	KG	Aluminum 6061	2700	kg/m3
EXTERIOR SHELL	8,000.00	KG	TBD	70	kg/m3
TOTAL MASS	**24,615.30**	KG			

Table 5-1: Initial inputs for the Structures Mass budget

Some structural components were however subjected to a deeper investigation and design effort during this study. This was the case for the Deployable Shell and the Windows (previously referred to as “Glazing”), for which the sizing and mass estimate details are further discussed in the Materials and Processes and the Systems Chapters, respectively. Therefore, the final Mass Budget for Habitat Structures is presented in Table 5-2.

	mass (kg)	mass margin (%)	mass incl. margin (kg)
Hab (Habitat)	**25199.28**	**20.00**	**30239.14**
Access (Accessibility)	649.50	20.00	779.40
Adapter (Adapter)	3439.00	20.00	4126.80
Ceiling (Ceiling Panels)	1023.00	20.00	1227.60
Cen_floor (Central Floor Panels)	772.40	20.00	926.88
Depl_Shell (Deployable Shell)	7195.00	20.00	8634.00
Ext_ceiling (Extended Ceiling Panels)	272.10	20.00	326.52
Ext_floor (Extended Floor Panels)	308.60	20.00	370.32
Ext_frame (Exterior Frame)	5072.00	20.00	6086.40
Ext_frame_int (Exterior Frame Interface)	131.90	20.00	158.28
Sec_struct (Secondary Floor Structure)	4435.30	20.00	5322.36
Window_frames (Window Frames)	259.60	20.00	311.52
Windows (Windows)	1640.88	20.00	1969.06
Grand Total	**25199.28**	**20.00**	**30239.14**

Table 5-2: Final Mass budget - Structures

6 MECHANISMS

6.1 Challenges and Needs Within the Concept of Operations

From the mechanisms perspective numerous challenging areas have been identified, particularly in the post-launch phases of the mission up to inflation of the habitat on the lunar surface. These challenges principally include:

- Challenges limiting the mass of deployable systems
- Difficulty leveraging re-use of existing space solutions
- A high launcher fill ratio expected for the habitat, leading to a need for separate landed assets to perform major deployment and mobility function

From the mechanisms perspective several opportunities have been identified. These principally include:

- The availability of humans on the surface and local pressurized environments, presents an opportunity to utilize manual deployments
- The presence of non-zero but reduced gravity (with respect to earth), facilitates the movement and handling of larger structures such as the proposed habitat

6.2 Baseline Design

6.2.1 Clamp Band Restraints

Clamp bands are needed to restrain the tall habitat with soft outer structure, protect the inflatable material and ensure a deterministic launch and landing behaviour.

A classic clamp band design is proposed. This can be supplemented by softer restraint collars to restrain the folded inflatable material against the structure in between the windows and near the location of the floors. This ensures the restraints are applied at the structurally strongest parts of the habitat. This is shown in Figure 6-1.

The clamp bands could remotely be released by a pyro signal to allow removal by the lunar mobile crane.

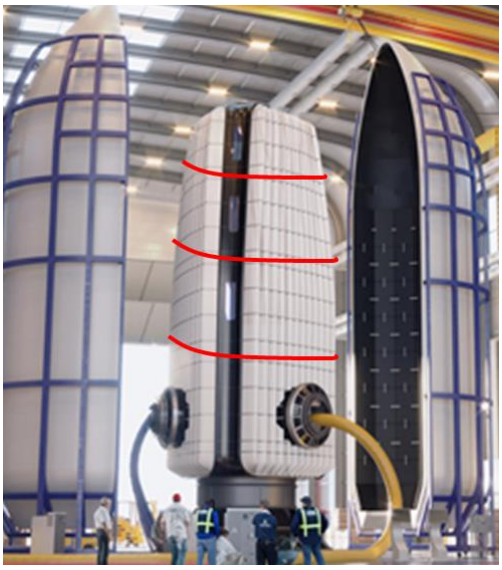

Figure 6-1: Proposed configuration of Clamp Bands around habitat (left) and typical pyro-released Clamp Band (right)

6.2.2 Berthing Hatches

These are needed to provide:

- Local stiffening and support
- Semi-passive hard capture, of structures to be attached (e.g. tunnels), to facilitate assisted mating with mobile crane
- Pressure sealing
- Hatch door (safe outward opening).

To satisfy these requirements, among others, several berthing mechanisms used to assemble ISS elements were studied RD[24]. The major difference with respect to the Moon Village is that in zero-g environments such as on the ISS, elements are currently positioned using robotic arms (e.g. SRMS or SSRMS/Canadarm2).

Design elements could be considered from the International Berthing and Docking Mechanism (IBDM). However as this system includes soft capture which is not assumed necessary, with lunar gravity a system similar to the Russian SSVP dock is considered (with significant changes). See Figure 6-2. Androgynous mating and standardised berthing interfaces such as NASA's ADBS project are also worth considering in the context of the Moon Village RD[25]. This would simplify the many berthing operations that are needed if the lunar base is to expand, as well as provide added flexibility in berthing any two objects (habitats, tunnels, airlocks, spacecraft, etc.)

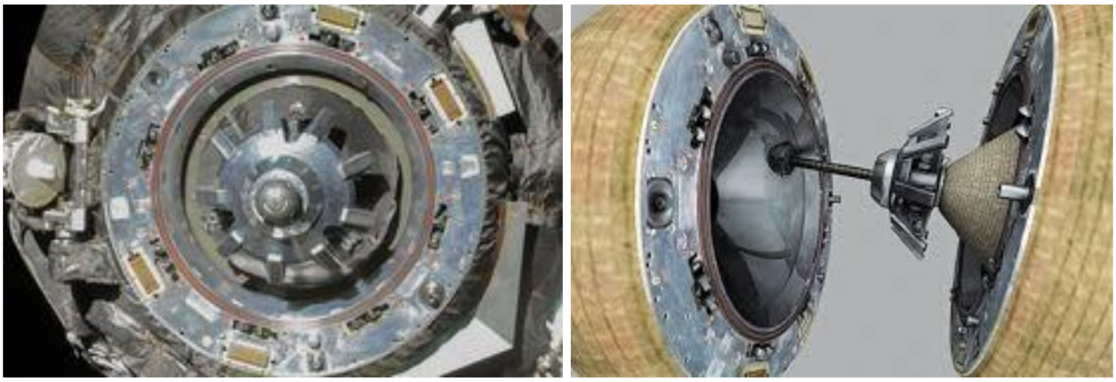

Figure 6-2: SSVP docking mechanism.

The hatches are assumed to mate to tunnels connected to a common airlock, one leading out from each of the 3 inflatable volumes. A 4th escape hatch is assumed on the top side of the habitat.

6.2.3 Deployable Floor Beam Hinges

These are needed to extend the available floor area of the habitat once deployed, as well as stiffen it structurally (something the inflatable material is assumed not to).

The proposed hinges are of simple design to allow manual deployment of floor support beams as proposed by SOM. The current design proposes 5 beams for each of the 3 inflatable volumes and on each of the 4 floors, thus totalling 60 hinges per habitat.

It is assumed that the hinges and beams shall be maintained deployed by the lunar gravity. The general floor thickness and local reinforcement shall be used to support the transfer of structural loads to minimize hinge size and mass. This is shown in Figure 6-3.

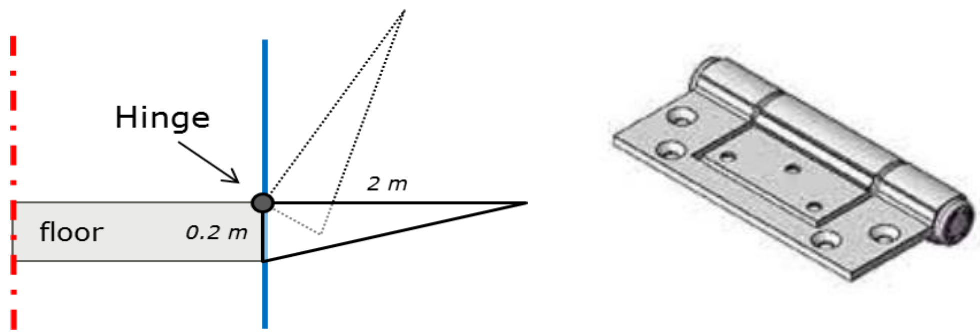

Figure 6-3: Floor beam hinge simplified design (left) and hinge example (right)

The hinges shall be sized according to the mass, envelope, location and number of fastening devices that are needed to restrain the deployable floor during launch.

6.2.4 Mobile Crane

The crane is needed to transport the habitat from the landing site. It is assumed the habitat cannot be used where it is landed as large amounts of lunar dust will be created by the lander. Uncontrolled lunar dust deposition (e.g. on habitats, solar cells, etc.) is undesired. As such a multi-functional electric mobile crane solution is proposed. This could be a gantry type RD[26] or pick-and-carry type. These are shown in Figure 6-4. Elements from the proposed UT, explored in an earlier HSV CDF study RD[27], are also considered.

Figure 6-4: Mobile gantry crane (left) and pick-and-carry crane (right)

A mobility system is proposed that is compatible with the lunar surface and a crane system is sized for the habitat weight on the lunar surface. A relatively simple Human-In-The-Loop (HITL) electro-mechanical control system with human control I/F is suggested. A battery can be used for remote charging before operation.

6.3 Budgets

6.3.1 Clamp Band Restraints

Based on standard clamp bands (i.e. LPSS) the mass can be estimated considering 7 kg/m diameter. For a stowed habitat diameter of approximately 5m the mass per restraint would be approximately 45 kg. The mass for a 3-ring restraint system would therefore near 150 kg. A 30% proposed margin is to be added.

6.3.2 Berthing Hatches

The budget is based on a combination of an active *IBDM* (approximately 150kg per side without hatch) and a passive Russian SSVP of which mass information is unavailable. It is assumed there is no soft capture (as per IBDM) but there is a hatch door and a pressure sealing similar to the SSVP. A mass of 150 kg per hatch and per side is estimated. Given a total of 4 hatches, the mass of the hatches per habitat is 600 kg. A 30% proposed margin is to be added.

6.3.3 Deployable Floor Beam Hinges

Manually deployable and passive hinges are sized considering lunar gravity acceleration, based on the following assumptions:

- Habitat diameter: ~4.5 m (stowed) / ~8.5 m (deployed)
- Flooring to hold: ~400 kg/m² (including floor panel mass & racks)
- Floor thickness: 0.2 m
- As per SOM: 5 beams on each of the 4 floors, for each of the 3 inflatable volumes (a total of 60)

This leads to the following torque and force estimations:

- Torque on each hinge beam: 472 Nm
- Load on each hinge: 2.4 kN

For these loads 200 g hinges can be used. The total mass of 60 hinges is thus ~12 kg. This does not include the structural mass of floor beams, panels or local reinforcements needed if composite floors are used. A 30% proposed margin is to be added.

A second load case will come from the launch vibrations. Assuming a mass of a single deployable beam of 100 kg, a peak acceleration of 10 g, and 6 additional restrain points, the magnitude of load for each hinge can be estimated to be 9.81*100*10/(6+1)~1.5 kN, with is the same order of magnitude as the load estimated from the lunar gravity.

6.3.4 Mobile Crane

A preliminary mass sizing proportional to payload weight (considering the relevant gravity environment) is proposed. Based on existing planetary locomotion and crane systems a 25% mass fraction is proposed as a challenging target. For either a gantry style or pick-and-carry style crane. For a 40 000 kg habitat this suggests a mobile crane mass of 10 000 kg. A 30% proposed margin is to be added.

A parametric study for power, based on planetary robots for lunar gravity leads to an estimated power of 0.017 W/kg/deg/sec. Thus, assuming a 50 000 kg mobile mass and a wheel speed of 1.0 deg/sec, a mobile crane power of 850 W is needed. A 30% proposed margin is to be added.

6.4 Options

None identified.

6.5 Technology Needs

The following areas have been identified as particularly low maturity for the target Moon Village application, and would need technology pre-development:

- Large lunar mobile crane system:
  - Would require actuator sizes that don't currently have heritage
  - Lubricated mechanisms will need to operate at cold temperatures, with a high level of reliability. This requires a proper combination of thermal control and/or the use of lubricant suitable for low temperatures (<-40°C).
  - The approach to the Human-In-The-Loop manipulation of large structures in low gravity is an area which would benefit from refinement
- Low gravity pressurised berthing mechanism

- Mechanisms necessary to assist the capture/fixation of large structures in low gravity (using HITL) is an area which would need development

Technology Needs						
*	Equipment Name & Text Reference	Technology	Supplier (Country)	TRL	Funded by	Additional Information
x	Large lunar mobile crane system	Crane	N/A	2	-	-
x	Low gravity berthing mechanism	Berthing mechanism	IBDM: QinetiQ (UK) under ESA contract SSVP: TsKBEM (Russia)	3	-	-

7 POWER

7.1 Challenges and Needs within the Concept of Operations

7.1.1 Testing

There are no unusual challenges foreseen related to the testing, but certainly the Habitat power system will be involved in testing. Even in the unlikely case that the Habitat has no intrinsic power generation or energy storage capability, it will have a power distribution network (with monitoring and protection functions) that will need to be tested.

7.1.2 Transportation to Launch Site

No unusual challenges are foreseen.

7.1.3 Launch

No unusual challenges are foreseen.

7.1.4 Dock with Transport Module / Space Servicing Module

The Habitat may be power autonomous or it may rely on an external module for power. (This is discussed further below). In the latter case, a high power (multi-kilowatt) interface with a robotic connect/disconnect function will be needed in order to transfer power from the external module to the Habitat.

7.1.5 Dock with Landing Module

The same consideration as for transport module / space servicing module (see above) applies here.

7.1.6 Landing

Mechanical forces of landing are one reason not to suggest deployable solar arrays on Habitat.

Dust mobilised by landing may stick to solar panels by e.g. static electrical phenomena.

7.1.7 Deployment from Lander

This may be a (semi) robotic action requiring electrical power. This power may need to be provided by the Habitat.

7.1.8 Transfer to Building Site

This may be a (semi) robotic action requiring electrical power. This power may need to be provided by the Habitat.

7.1.9 Assembly of Architecture

This may be a (semi) robotic action requiring electrical power. This power may need to be provided by the Habitat, in which case an intrinsic power system is required.

7.1.10 Inflation

This may be a (semi) robotic action requiring electrical power. This power may need to be provided by the Habitat.

7.1.11 Outfitting & Commissioning

The Habitat will need to be connected to an external power supply ("power plant"). Suitable connector and electrical interface standards will have to be established.

7.1.12 Installation

No specific unusual challenges.

7.1.13 Usage

Operational power demand is further discussed below.

The redundancy, reliability and safety concept may demand some degree of power autonomy in emergency cases, even if the Habitat is normally supplied by an external power plant. This autonomy could be provided by energy storage or by a combination of storage plus solar generation. The embarkation of solar generators alone without any energy storage would not be suitable, due to the periods of darkness.

7.1.14 Unoccupied Phase

There may be a power requirement for thermal control, monitoring, computer housekeeping systems. It is yet to be determined if an external power plant remains available during unoccupied phase.

7.1.15 Decommissioning

Energy storage systems can present a continued hazard after switch-off: As per the satellite EOL passivation issue, batteries must be discharged and isolated from charging sources. However, if there are no plans for humans to be in the vicinity again, then perhaps this is not required.

7.2 Design Requirements and Assumptions

7.2.1 Power Requirement Estimate by Comparison

Various studies have estimated a wide range of values for the electrical power requirements of a human lunar habitat. For instance, Cataldo and Bozek (1993) RD[30] estimate approximately 10 kW for a surface crewed outpost. Mason (2006) RD[31] estimates approximately 60 kW continuous consumption for a crewed habitat at full power.

The ISS is a source of real data for a crewed space habitat, albeit one in Earth orbit rather than on the lunar surface. The ISS typically has 6 crew, as compared to the 4 of the lunar habitat under study, but the total pressurised volume is similar. The ISS has a continuous power delivery capability of 84 kW, with maximum power output of 108 kW. However 25 to 35 kW of this total is available for payload operations.

7.2.2 Power Requirement Estimate by Subsystem

An original estimate of power requirement has been made in the course of the study, split by subsystem and, in some cases, at equipment level. This is presented Table 7-1.

During the transfer to the Moon, the power required is driven mainly by heaters, in order to keep the internal environment of the Habitat at the desired temperature. In nominal operations, both during the lunar night and day, the power budget is driven by the ECLSS.

A power of 5 kW has been allocated to science operations.

In total, including a 20% system margin, the average power requirement is 57 kW during the day and 60 kW during the night.

Row Labels	Transfer	Nom_Ops_Day	Nom_Ops_Night
Hab (Habitat)	7320	47791	47841
INS	0	491	491
Crew_Quarters (Sleep Accomodation and Medical Equipment)	0	105	105
Galley (Galley and Food Systems)	0	385	385
Mob_Aids (Restraints and Mobility Aids)	0	0	0
Waste_Hygiene (Waste Collection and Hygiene)	0	1	1
PWR	120	400	400
PCDU (Power Conditioning and Distribution Unit)	120	0	0
PDU_1 (Power Distribution Unit)	0	100	100
PDU_2 (Power Distribution Unit)	0	100	100
PDU_3 (Power Distribution Unit)	0	100	100
PDU_4 (Power Distribution Unit)	0	100	100
SYE	0	6900	6950
Hab_light (Habitat Lighting)	0	300	350
Laptop_1 (Laptop)	0	200	200
Laptop_2 (Laptop)	0	200	200
Laptop_3 (Laptop)	0	200	200
Laptop_4 (Laptop)	0	200	200
Laptop_5 (Laptop)	0	200	200
Laptop_6 (Laptop)	0	200	200
Laptop_7 (Laptop)	0	200	200
Laptop_8 (Laptop)	0	200	200
Sci_Ops_alloc (Science/Surface Operations Allocation)	0	5000	5000
TC	7200	0	0
TH_Hab_MLI_Heaters (Thermal Habitat MLI Heaters)	7200	0	0
ECLS	0	40000	40000
Bulk_ECLS_pwr (Bulk ECLS Power)	0	40000	40000
Surf_Rad (Surface Radiators)	0	0	2000
TC	0	0	2000
TH_Surf_Rad (Thermal Surface Radiators)	0	0	2000
Grand Total	**7320**	**47791**	**49841**
Grand total with system margin	**8784**	**57349**	**59809**

Table 7-1: Power requirements for each system mode (time-averaged power in watts)

7.2.3 Illumination Conditions at the Lunar South Pole

Because the lunar rotational axis is close to perpendicular to the ecliptic plane, the Sun elevation at the poles remains close to zero (horizontal). This means that points of high ground can be in (near) permanent sunlight.

Detailed studies have assessed the illumination conditions in the lunar south pole region. Data from the Lunar Orbiter Laser Altimeter (LOLA) instrument of the Luna Reconnaissance Orbiter (LRO) spacecraft have given rise to high resolution analyses such as those described by Gläser et al. in RD[28] and Marazico et al. in RD[29].

Gläser et al. (2014) reported that “We identified locations receiving sunlight for 92.27% of the time at 2 m above ground…….at these locations the longest continuous periods in darkness are typically only 3-5 days”. However, such “headline” figures refer to the most optimum 20 m x 20 m pixel. It is not realistic to assume that a lunar base could be placed *exactly* at such a point regardless of all other considerations of suitability.

Figure 7-1 is reproduced from Gläser et al. (2014), and shows possibly the most favourable area of the south polar region. Highlighted in black squares are the regions assessed to have at least 80% long-term accumulated illumination at 2 m height above the surface. We see that these areas are quite small, in the region of 100 m across.

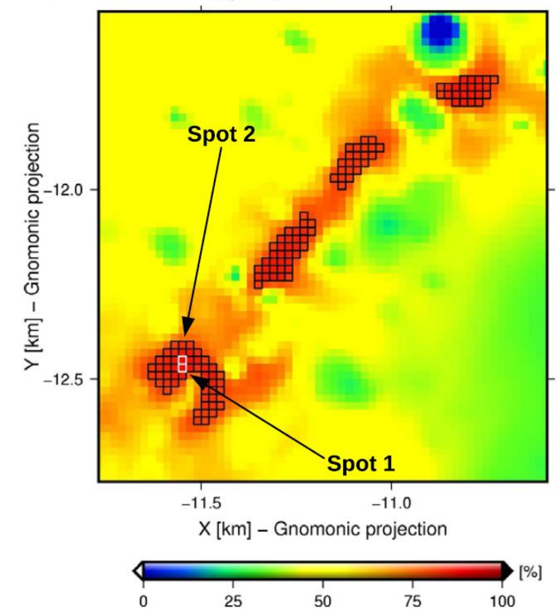

Fig. 17. Accumulated illumination map of CR1 (89.4555°S, 222.6192°E) over a period of 19 years at 2 m height above the surface level. Black outlined pixels represent spots with an average illumination higher than 80%. Spots 1 and 2 are outlined in white. The color bar indicates accumulated illumination over the considered time period. (For interpretation of the references to color in this figure legend, the reader is referred to the web version of this article.)

Figure 7-1: Illumination map from Gläser et al.(2014), RD[28]

For this CDF study, the selection of an optimum site should be assumed, but also consider the uncertainties in the data and practical constraints in habitat placement.

Therefore, the assumptions for illumination conditions for this study are the following:

- Average illumination long-term: 80%

- Longest darkness period: 5 days (120 hours).

7.3 Baseline Design

7.3.1 Habitat Intrinsic Power System

Analysis of the Concept of Operations (see Section 7.1) identifies potential reasons why the Habitat may require an intrinsic electrical power generation capability that is operable during the journey to the ultimate installation site, both in flight and during ground transfer. This principle has been further quantified only in one area – that of thermal control. Table 7-1 includes a value of 7.2 kW for TCS heaters in system mode "Transfer".

7.3.1.1 Structure-mounted solar panels

The structural concept of the Habitat leaves very little area which is available for solar panel mounting. Solar panels could be added on the spines in the area between the windows as illustrated in Figure 7-2.

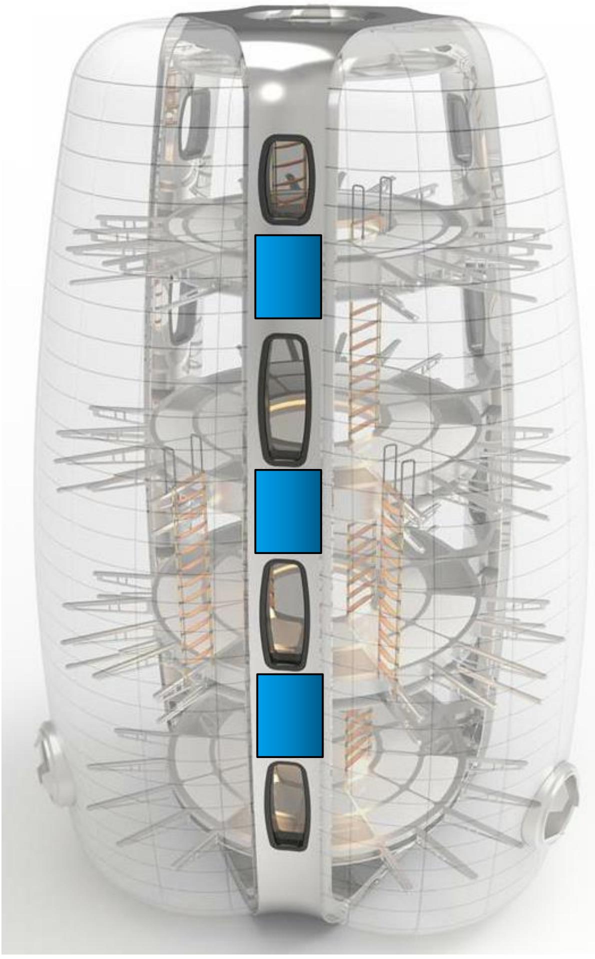

Figure 7-2: Concept for solar panel placement

This would provide approximately 5 m² of solar panel area for each of the three "spines", giving 15 m² in total. The solar panels mass would be around 56 kg in total.

The solar panels would generate approximately 1100 W during the transfer flight, assuming that the habitat is maintained in a slow thermal roll ("barbeque mode") with its main axis perpendicular to the sun direction.

On the lunar surface, the power generation would depend on the attitude of the habitat with respect to the sun direction, and of course the availability of the sun with respect to the local topography during the Habitat's transit from landing site to installation site. Taking an illustrative average case in which the rotational attitude of the habitat with respect to the sun is random and variable, and the accumulated illumination is 80%, the average power generation would be around 900 W. These power and mass calculations are further detailed in Table 7-2.

Solar array power generation estimation		
Number of spines	3	
Solar panel area per spine	5	m^2
Specific power of sun-pointed panel	300	W/m^2
Specific mass of panel incl. substrate, PVA, wiring & mounts	3.7	kg/m^2
Factor accounting for average sun angle and shadowing on surface	0.307	
Factor accounting for average sun angle in flight BBQ roll	0.31	
Accumulated sunlight at habitat site	0.8	
PMAD overall efficiency	0.8	
Long-term average total power available SURFACE	**884**	**W**
Long-term average total power available FLIGHT	**1116**	**W**
Solar panel total mass	**56**	**kg**

Table 7-2: Power generated by solar panels, and mass estimate

7.3.1.2 Battery

The Habitat intrinsic power system would require also some energy storage capability in order to provide continuous power delivery during periods of darkness or panel off-pointing.

If it is assumed that the surface average power generation of 884 W derived in Table 7-2 must be delivered continuously, and that the worst case darkness duration will be 5 days (see Section 7.2.3), then the required battery mass and volume can be estimated. This calculation is detailed in Table 7-3. The required battery is almost one tonne in mass, and is therefore assumed to be implemented as 20 separate modules of 49 kg and 49 litres each.

Battery size and mass estimation		
Power delivery requirement	884	W
Maximum duration of continuous darkness	120	hours
Effective mass-specific energy of integrated battery	109	Wh/kg
Battery density assumed	1	kg/l
Total mass of batteries required	**973**	**kg**
Number of battery modules	**20**	
Mass of each battery module	**49**	**kg**
Volume of each battery module	**49**	**l**

Table 7-3: Battery size and mass estimate

7.3.1.3 Power conditioning and distribution

A mass allocation of 15 kg (before margin) is estimated for the PDCU which will manage and distribute the ~1 kW of power delivered by the Habitat intrinsic power system, via interface with the installed solar panels and the internal batteries.

When the habitat is attached to an external power supply in nominal conditions after installation at the permanent site, it is reasonable to assume that primary power conditioning is handled by the external system (the "power plant"). Therefore, only protection, distribution & monitoring "PDUs" are required. However these will have to manage and protect a very high power system, so will be significant in mass terms. A total of 80 kg is allocated for these PDUs, divided in to 4 units of 20 kg each.

7.3.1.4 Inadequacy of the installed power system for the estimated thermal control requirements

As detailed above, an intrinsic installed power system seems credible only for continuous power delivery in the region of 1 kW. The thermal calculations show that 7.2 kW is required (before margin). In the absence of alternative solutions (see Section 7.6), such power would have to be provided by external elements.

For instance: during transfer flight, it is credible to assume that the Habitat could be supported electrically by the transport module spacecraft. Similarly, the lunar surface assets that transport the habitat from landing site to installation site could provide power through umbilical connection.

7.3.2 Habitat Electrical Power Quality

The electrical power standard for the NASA Orion crew module and the lunar Gateway is derived from the ISS standard: it seems logical to assume it as a future human space flight standard that could be employed for the internal power network of the Habitat. This implies a 120 V unregulated DC supply (in practice 104 to 136 V depending on e.g. battery state of charge).

7.4 External Power Plant Options

It seems clear that a lunar Habitat as considered here would be supplied with electrical power by some kind of external lunar surface "power station". Most probably, this power station would provide energy to not only the habitat, but to a variety of lunar surface assets at the crewed outpost. Some sizing estimates are made below, in order to trade-off different technology options and give some sizing context to the power plant assumption. These are based on a continuous power requirement of 59 kW, i.e. that of the habitat alone. The parametric assumptions made in the sizing calculation are also presented.

7.4.1 Solar Power Plant

A solar power plant at the south pole would have to maintain an array of solar panels facing horizontally, with a rotational capability in order to track the sun through 360° of azimuth over the course of one month. This leads to concepts like that illustrated in Figure 7-3. However, such a power station would have to overcome the problem of mutual shadowing, in which some panels would always be shadowed from the horizontal sun by some neighbouring panels, or indeed by any other surface constructions such as the habitat itself.

Furthermore, the darkness periods (assumed to be of duration up to 5 days – see Section 7.2.3) demand the inclusion of large energy storage capability.

High level estimates of the mass and panel area of such a solar power station, to provide 59 kW continuous power, are presented below in Table 7-4 and Table 7-5 for a battery-supported and an RFC-supported system respectively.

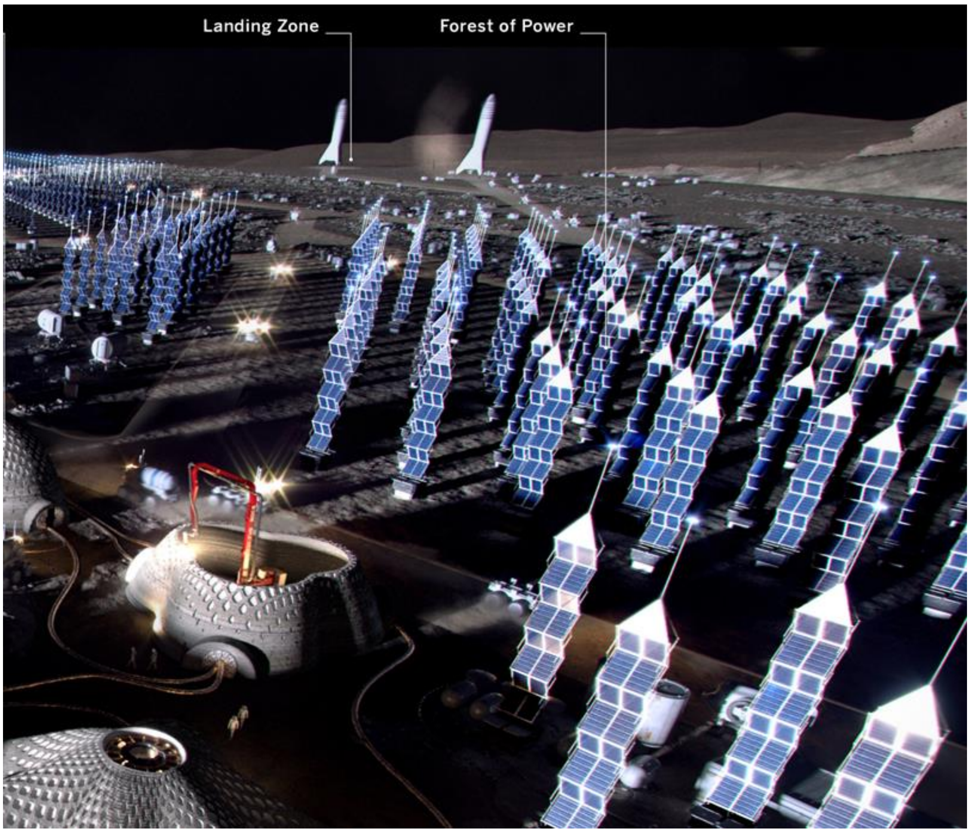

Figure 7-3: Lunar surface solar power concept (Skidmore, Owings and Merrill LLP)

Solar - battery		
Power required by Habitat	59	kW
Darkness max duration	120	hours
Energy Storage Requirement	7080	kWh
Effective mass-specific energy of space Li-Ion batteries	109	Wh/kg
Required mass of Li-Ion batteries	65	tonnes
Area-specific power of pointed solar panels	300	W/m^2
Accumulated sunlight at habitat site	0.8	
PMAD overall efficiency	0.9	
Battery round-trip efficiency	0.95	
Power required from solar panels	85	kW
Solar panel area required	282	m^2
Area-specific mass of solar panels	5	kg/m^2
Mass of solar panels	1.4	tonnes
Area-specific mass of panel mount and rotation system	5	kg/m^2
Mass of panel mount and rotation system	1.4	tonnes
Mass-specific power handling of PMAD equipment	500	W/kg
Mass of PMAD equipment	169	kg
Total power station mass	68	tonnes

Table 7-4: Assumptions and mass estimate for a solar (with battery) power station

Solar - regenerative fuel cell		
Power required by Habitat	59	kW
Darkness max duration	120	hours
Energy Storage Requirement	7080	kWh
Mass-specific energy of space H2-O2 regen. fuel cell sys.	700	Wh/kg
Required mass of RFC system	10	tonnes
Area-specific power of pointed solar panels	300	W/m^2
Accumulated sunlight at habitat site	0.8	
PMAD overall efficiency	0.9	
RFC round-trip efficiency	0.55	
Power required from solar panels	99	kW
Solar panel area required	329	m^2
Area-specific mass of solar panels	5	kg/m^2
Mass of solar panels	1.6	tonnes
Area-specific mass of panel mount and rotation system	5	kg/m^2
Mass of panel mount and rotation system	1.6	tonnes
Mass-specific power handling of PMAD equipment	500	W/kg
Mass of PMAD equipment	197	kg
Total power station mass	14	tonnes

Table 7-5: Assumptions and mass estimate for a solar (with RFC) power station

7.4.2 Nuclear Fission Power Plant

Nuclear fission reactors for lunar surface application are a technology under active development, most notably within NASA development programmes in the USA. RD[32].

A nuclear fission reactor would certainly be more compact, for a given power capability, than a solar power farm. However, extensive cooling radiators are required to reject the waste heat at a temperature low enough to suit the power conversion principle involved.

Protection of crew and systems from the ionising radiation emissions of an operating reactor would be achieved by a combination of distance and shielding by regolith (most likely by burial). Figure 7-4 shows a design concept of a 40 kW lunar surface fission reactor RD[33]. In a lunar pole application, the radiators would face upwards rather than horizontally.

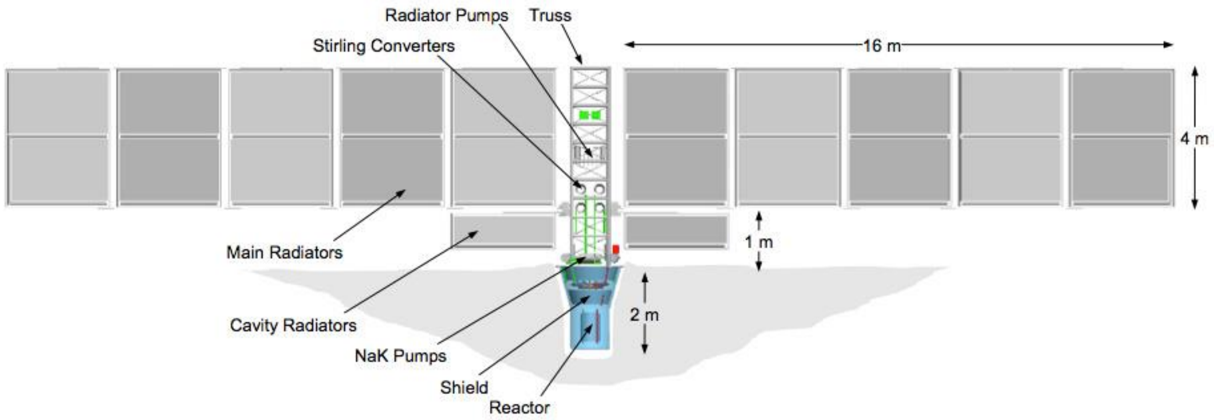

Figure 7-4: Design reference concept of a 40 kW fission surface power system (NASA/LANL, RD[33])

The mass of a space fission reactor system depends on a multitude of design parameters and assumptions. Nonetheless, in order to determine an approximate generic relationship between power output and mass, the outputs of a variety of ESA and open literature studies have been compared (see Figure 7-5). This leads to an assumption of 12 W/kg at the power range of interest (~60 kW). Using this key input value, a high-level estimate of the mass and radiator area of a fission power station, to provide 59 kW of continuous power, is presented below in Table 7-6.

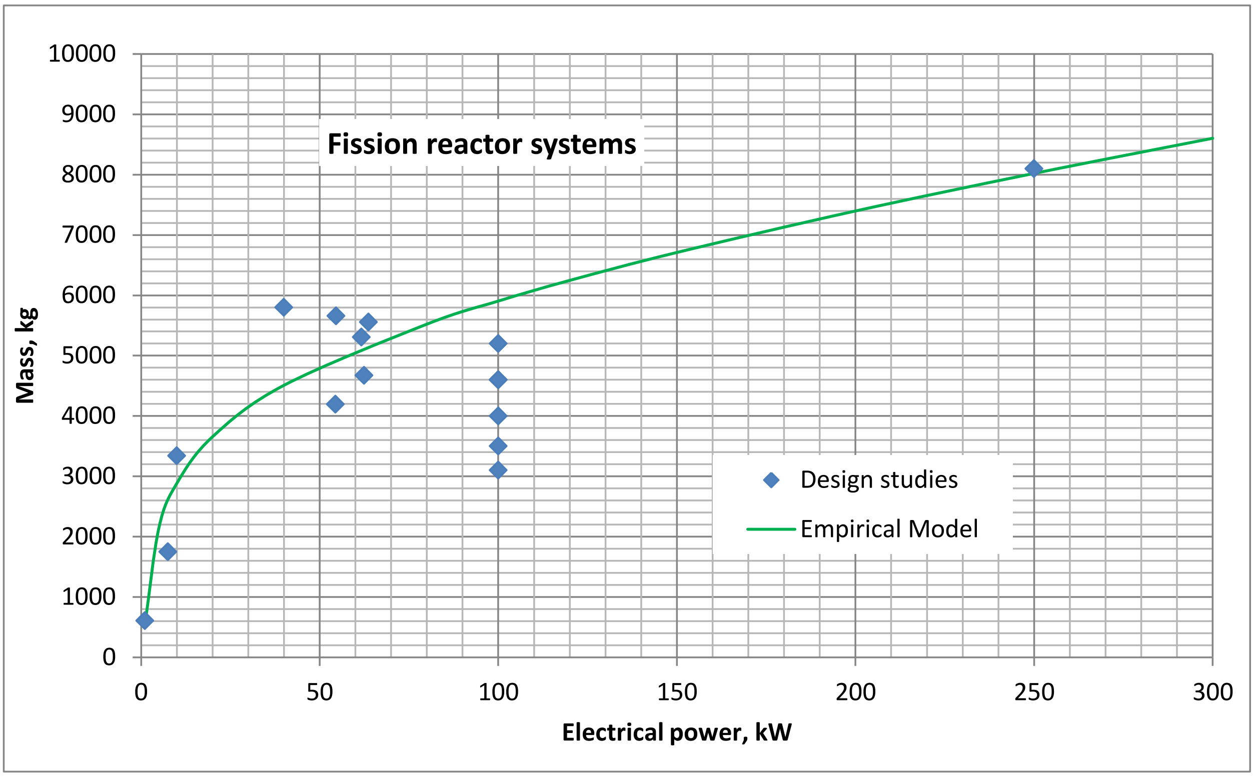

Figure 7-5: Mass vs. power data points for space fission reactors, derived from ESA and open literature studies

Fission reactor system		
Power required by Habitat	59	kW
PMAD overall efficiency	0.9	
Power required from reactor system	66	kW
Mass-specific power lunar surface fission system	12	W/kg
Required mass of lunar surface fission system	5.5	tonnes
Mass-specific power handling of PMAD equipment	500	W/kg
Mass of PMAD equipment	131	kg
Total power station mass	5.6	tonnes
Power-specific radiator area	2.6	m^2/kW electric
Radiator area required	173	m^2

Table 7-6: Assumptions and mass estimate for a nuclear fission power station

It is clear that the fission reactor system would be a lighter and more compact option than a solar power plant. It would also have the advantage that its development would be applicable to non-polar lunar applications, providing wider mission flexibility. On this basis, the nuclear fission option is selected as the baseline assumption.

7.5 Budgets

A list of power system equipment (both for the Habitat and the external power plant) is presented below, with mass estimates, in Table 7-7.

	mass (kg)	mass margin (%)	mass incl. margin (kg)
Hab (Habitat)	**1123.89**	**6.51**	**1197.11**
Bat_hab_01 (Battery_habitat)	48.67	5.00	51.10
Bat_hab_02 (Battery_habitat)	48.67	5.00	51.10
Bat_hab_03 (Battery_habitat)	48.67	5.00	51.10
Bat_hab_04 (Battery_habitat)	48.67	5.00	51.10
Bat_hab_05 (Battery_habitat)	48.67	5.00	51.10
Bat_hab_06 (Battery_habitat)	48.67	5.00	51.10
Bat_hab_07 (Battery_habitat)	48.67	5.00	51.10
Bat_hab_08 (Battery_habitat)	48.67	5.00	51.10
Bat_hab_09 (Battery_habitat)	48.67	5.00	51.10
Bat_hab_10 (Battery_habitat)	48.67	5.00	51.10
Bat_hab_11 (Battery_habitat)	48.67	5.00	51.10
Bat_hab_12 (Battery_habitat)	48.67	5.00	51.10
Bat_hab_13 (Battery_habitat)	48.67	5.00	51.10
Bat_hab_14 (Battery_habitat)	48.67	5.00	51.10
Bat_hab_15 (Battery_habitat)	48.67	5.00	51.10
Bat_hab_16 (Battery_habitat)	48.67	5.00	51.10
Bat_hab_17 (Battery_habitat)	48.67	5.00	51.10
Bat_hab_18 (Battery_habitat)	48.67	5.00	51.10
Bat_hab_19 (Battery_habitat)	48.67	5.00	51.10
Bat_hab_20 (Battery_habitat)	48.67	5.00	51.10
PCDU (Power Conditioning and Distribution Unit)	15.00	20.00	18.00
PDU_1 (Power Distribution Unit)	20.00	20.00	24.00
PDU_2 (Power Distribution Unit)	20.00	20.00	24.00
PDU_3 (Power Distribution Unit)	20.00	20.00	24.00
PDU_4 (Power Distribution Unit)	20.00	20.00	24.00
Sol_pan_hab_1 (SolarPanelHabitat)	6.17	10.00	6.78
Sol_pan_hab_2 (SolarPanelHabitat)	6.17	10.00	6.78
Sol_pan_hab_3 (SolarPanelHabitat)	6.17	10.00	6.78
Sol_pan_hab_4 (SolarPanelHabitat)	6.17	10.00	6.78
Sol_pan_hab_5 (SolarPanelHabitat)	6.17	10.00	6.78
Sol_pan_hab_6 (SolarPanelHabitat)	6.17	10.00	6.78
Sol_pan_hab_7 (SolarPanelHabitat)	6.17	10.00	6.78
Sol_pan_hab_8 (SolarPanelHabitat)	6.17	10.00	6.78
Sol_pan_hab_9 (SolarPanelHabitat)	6.17	10.00	6.78
Pwr_plant (Power Plant)	**5594.07**	**20.00**	**6712.89**
LSFR (Lunar Surface Fission Reactor)	5462.96	20.00	6555.56
PCU (Power Conditioning Unit)	131.11	20.00	157.33
Grand Total	**6717.96**	**17.74**	**7910.00**

Table 7-7: Power subsystem equipment mass budget

7.6 Options

7.6.1 European Large Heat Source (ELHS)

The power required by the thermal control subsystem could be reduced if European Large Heat Sources (ELHSs) were used in the Habitat. European radioisotope heat sources, using Am-241 fuel, are currently at TRL 4; an activity is underway to achieve TRL 5 at the end of 2020. The ELHS is primarily designed as a heat source element for an RTG, but could also be used as a large RHU to provide additional heat without electrical consumption. Data for the ELHS are reported in Table 7-8.

Property	Value
Thermal power	190 W
Mass	6.6 kg
Specific power	29 W/kg
Mass of Am-based fuel	2.3 kg

Table 7-8: Data for the ELHS – A European ^{241}Am-based radioisotope heat source

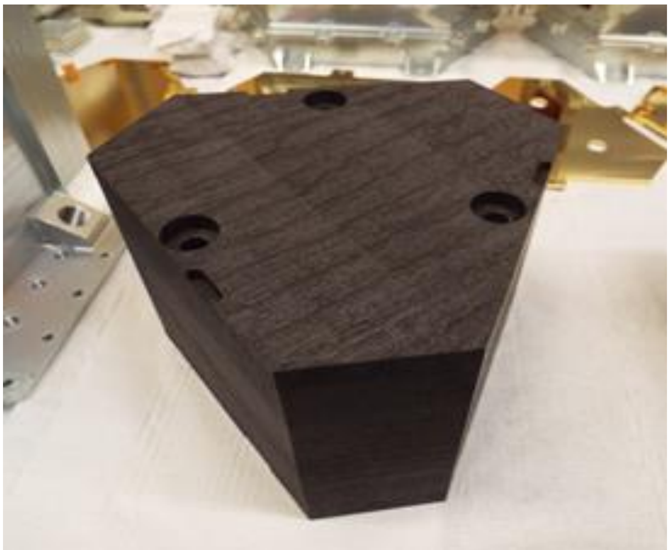

Figure 7-6: Prototype ELHS manufactured at the University of Leicester

In order to provide, for example, around 1 kW of thermal power to the habitat, five ELHS would be required:

- Heat supplied: 950 W
- Mass of Am-based fuel: 11.5 kg
- Total ELHS mass: 33 kg, without any heat distribution system.

7.7 Technology Needs

The following technology developments may be required:

7.7.1 Multi-kilowatt Long Lifetime Lunar Polar Power Station

The option that includes solar arrays has challenges of energy storage in darkness periods, or mutual shadowing periods for the vertical solar panels. It would have a high mass, due mainly to the energy storage element. Use of a regenerative fuel cell system would bring mass benefits as compared to present-day Li-ion space batteries, but the maturity is much lower.

A nuclear option (specifically a fission reactor system) would be a lighter and more compact option. It would also have the advantage that its development would be applicable to non-polar lunar applications, providing wider mission flexibility.

7.7.2 Human Space Exploration Electrical Interface Standards

The emerging standards for electrical interfaces, based on flight systems such as the Orion Crew Capsule and the Gateway, will need to be expanded to cover higher power systems and lunar surface compatibility.

8 GUIDANCE NAVIGATION AND CONTROL

8.1 Challenges and Needs Within the Concept of Operations

In each scenario the two main tasks for the GNC subsystem are rendezvous (in GTO or Lunar orbit) and precision landing at the Moon Village site.

8.1.1 Rendezvous Challenges

The main challenges for any rendezvous scenario are the size of both the chaser (tug) and target (habitat or lander), and the controllability of the resulting stack:

- The control function will have to cope with an evolving stack (tug-habitat, then tug-habitat-lander and finally habitat-lander) and the resulting changes in CoM and MCI.
- The control authority could be marginal if all thrusters are localised on one end of the stack (e.g. tug-habitat). More control authority might be available if both tug and lander are part of the stack, with the added complexity of making the lander attitude thrusters controllable by the tug.
- GNC for autonomous Lunar rendezvous has to be developed at European level (shall be demonstrated by upcoming Chang'e 5 and also by the Mars Sample Return mission to which ESA will be contributing RD[37]).
- GNC for autonomous rendezvous in highly elliptical orbit such as LTO has to be developed.
- The location of the rendezvous sensors (normally at the front/top of the chaser, facing V-bar) can be an issue, especially in a scenario where a tug already docked to the habitat would then dock to the lander, which would imply either to mount the sensors on a deployable mast or to have an additional set of sensors on the habitat.

8.1.2 Landing Challenges

Landing challenges are as follows:

- The lander with its habitat payload will be top-heavy, with an elevated CoM, which is a challenge for the control.
- Landing the first habitat (or a precursor mission) shall require the use of precision navigation and hazard detection and avoidance (both already under development for missions such as PILOT RD[35] or Mars landers RD[36]). These techniques should scale up easily to the large size of the lander required for this mission; however, finding a large flat, horizontal area large enough to land safely will be more a challenge for the HDA than it is for the current generation of small landers (less than 4m diameter).
- Follow-up landers can benefit from infrastructure deployed by precursor missions (landing pad, navigation beacons).

8.2 Baseline Design

- Rendezvous sensors:

  - Long-range sensors (down to a few 100m): narrow-angle camera.
  - RGPS (already used on ATV; chaser and target use the same GPS receptor) might be possible for LTO rendezvous, but would require further study due to low signal strength and unfavourable geometry.
  - Short range: wide angle camera visible, infrared, or combination of both, either with separate units or a single multispectral sensor.
  - Light beacons or reflective markers could be installed on the targets to help the tracking algorithms.
- Landing sensors:
  - Wide-angle cameras for vision-based navigation, first with absolute navigation (position estimation with landmark recognition) at high altitude (down to 2km altitude), then handover to relative navigation (velocity estimation with features tracking) until touchdown.
  - Laser altimeter to provide altitude information for the relative navigation.
  - LiDAR for HDA scans – could be combined with the laser altimeter.
  - For follow-up flights : range & range rate radio beacons can be disposed in a 20km square centred on the landing site to provide 10m navigation accuracy during approach and vertical descent (RD[34]).

8.3 Budgets

Given the large size and mass of the tug assembly and lander, the weight of the rendezvous & landing sensors can be considered negligible. Mass of up to a few kilograms can be budgeted for each sensor unit.

Rendezvous is performed with cameras which are low-power sensors. Landing on the other hand would have to rely on a power-hungry LiDAR. A full landing navigation system, including LiDAR, camera and dedicated processor would require 160W at peak usage during the LiDAR scan for hazard detection and avoidance.

8.4 Technology Needs

Technology Needs						
*	Equipment Name & Text Reference	Technology	Supplier (Country)	TRL	Funded by	Additional Information
		Autonomous RdV in LLO	GMV (Spain/Romania/Poland)	4/5	HRE (MSR)	To be demonstrated by Chang-e 5
		Autonomous RdV in LTO	SENER			RdV in elliptical orbit TRL 9 by 2020 (Proba-3) Also TRP on RdV GNC in

Technology Needs						
*	Equipment Name & Text Reference	Technology	Supplier (Country)	TRL	Funded by	Additional Information
						NRO
	Advanced GNC for Assembly of Large and Flexible Structures and Vehicles	GNC for assembly of large components	GMV	4	GSTP	
		Lander control with elevated CoM				To be started
		HDA for landing of large landers	NGC (Canada) or Spin.Works (Portugal)	4/5		PILOT will demonstrate HDA for smaller lander (Luna-27)
		Lunar landing beacon design		2/3		To be started
		Lunar landing with beacons		2/3	DLR	

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9 THERMAL

The objective of the Thermal Control System (TCS) is to guarantee that all units, equipment, parts and components remain within their design temperature ranges during the mission lifetime.

TCS requirements valid for each mission phase are summarised in the table below.

	Requirement	Comment
TC-1	Keep the habitat within required temperature range (interior as well as external side).	During transfer: Insulated compartment 10°C, non-insulated compartment -20°C On lunar surface: average temperature in the habitat 22°C
TC-2	Keep the gradient across the habitat (top-bottom, across, circumference) within required values.	
TC-3	Keep temperature stable (reduce fluctuation per lunar day, seasons).	

Table 9-1: TCS Requirements

9.1 Challenges and Needs Within the Concept of Operations

The challenges to fulfil requirements mentioned in Table 9-1 are different for each phase of the mission. The most important ones are summarised in Table 9-2.

	CONOPS phase	Challenges
1	Transfer from Earth to the Moon	To keep the interior of the habitat within required temperatures, it needs to be insulated with MLI and temperature inside needs to be regulated by heaters: - The thermal design (especially MLI) needs to be designed in a way that it does not disturb the deployment of the habitat on the Moon. - The MLI needs to be able to adapt its shape to an increasing volume whilst maintaining its insulation performance. - Heater power needs to be provided during the whole transfer phase.
2	Landing	While planning landing operation/configuration and habitat TCS insulation, following effects shall be taken into account: - Depending how the landing system is done (integrated into habitat or attached) there will be heat flow from thrusters to the habitat. - Also it is possible that parts of the habitat are going to be illuminated by the Sun or exposed to IR heat load from the Moon's surface, due to the orientation during landing.

	CONOPS phase	Challenges
		- The thermal mass of the habitat might be high enough to compensate heat losses during a short landing, i.e. power for thermal control might not be necessary for the pure landing phase. - If the landing takes too long, relevant amounts of heat can be injected into habitat and increase equipment temperatures.
3	Transfer to the building site, assembly, inflation, installation	- The habitat needs to be powered for thermal control to work. If the habitat is disconnected/unpowered - a way to keep it within temperatures required needs to be found. - If the habitat has to be assembled on the lunar surface, additional care needs to be taken to keep internal elements of the habitat in their respective temperature ranges: that may require the use of some additional thermal insulation on disassembled parts before mounting the parts of the habitat together.
4	Usage	Concept of TCS for the habitat during operations on the surface assumes that it is going to be insulated from the external environment (to limit influence of lunar day/night) as much as possible: - The heat dissipated inside the habitat needs to be transported and rejected outside to the environment. Possible place for the radiators near/on the habitat needs to be found. - For waste heat transport, pumped fluid system needs to be developed or adapted from current ISS use: fluid to be used internally shall be non-toxic. - The heat distribution system of the habitat is only connected to an external (additional) part of the infrastructure to provide the necessary pumping and heat rejection capability. - Use of lunar regolith to insulate parts of the habitat can be foreseen. It might help in insulating the habitat during night-time but it also prevents heat rejection during day-time and full operation. Another option might be to cover the fluid lines with regolith or use regolith as thermal buffer.

Table 9-2: Major Thermal Challenges

9.2 Baseline Design

In this CDF study, TCS design is focused on 2 CONOPS phases:

- Transfer (stowed habitat configuration to be transported from Earth to the Moon),
- Usage (deployed habitat on the Moon, fully functional).

9.2.1 Baseline Design for Transfer

During the transfer to the Moon, the habitat is going to be in stowed configuration and it will be switched OFF. During that phase the main task of TCS is to keep the habitat above minimum required temperature (requirement TC-1).

9.2.1.1 Habitat external layer

In order to find the best thermal protection, various coatings have been taken into account. A list of possible options is presented in Table 9-3. It is assumed that the external protection shall be the same for the transfer and operations on the Moon. Therefore, it needs to be flexible enough to be in semi-stowed configuration during transfer and then to be fully deployed on the Moon. And it shall provide effective thermal protection in all phases. Due to that fact, use of white paint is not an option as it needs to be applied on rigid structures and might degrade in stowed configuration. Also whether SSM can be applied to foldable structures is questionable, so it is not going to be further considered in this study.

	Coating	Comment
1	VDA	External MLI layer coating
2	White Paint	Not further analysed due to possible issues with application on foldable structure.
3	Kapton®	External MLI layer coating
4	BetaCloth	External MLI layer coating
5	SSM	Not further analysed due to possible issues with application on foldable structure.

Table 9-3: Possible coatings for thermal protection

MLI is considered the best option for external thermal protection of the habitat. Three possible MLI options have been analysed, in order to assess the heater power needs during the transfer and during lunar day and night phases on the surface of the Moon.

Properties of the materials are presented in Table 9-4. The presented values are optical surface properties of the external layers. For each material the BoL properties are used for the transfer calculations. For calculations on the surface of the Moon dedicated EoL properties are used, to take into account the impact of lunar dust.

	Coating	BoL		EoL (with dust)	
		α	ε	α	ε
1	VDA	0.13	0.05	0.26	0.1
2	Kapton ®	0.44	0.75	0.88	0.97
3	BetaCloth	0.29	0.85	0.58	0.97

Table 9-4: Possible coatings for thermal protection

For each MLI the same effective emissivity of $\varepsilon^*=0.1$ has been assumed. The internal temperature of the habitat is set to 0°C for transfer case and to 22°C for lunar cases. The results of these calculations are presented in Table 9-5.

	Temperature inside the habitat [°C]	VDA	Kapton	BetaCloth	Comments
Transfer	0°C	2.8 kW	12.2 kW	12.4 kW	BoL optical properties
Lunar night	22°C	9.6 kW	17.6 kW	17.6 kW	EoL optical properties
Lunar day	22°C	-46.5* kW	-4.7* kW	0.1 kW	

* *In this cases the amount of reported heat is the heat to be rejected additionally to heat dissipated inside the habitat.*

Table 9-5: Heater power estimation, all values are in kW

It can be easily noticed that when VDA is used as external layer, heater power needed for the transfer and lunar night is significantly lower than for other two materials. However, with VDA as external layer, the habitat absorbs a huge amount of heat during lunar day (it absorbs almost the same amount of heat as is dissipated in the habitat) and to reject this additional amount of heat, the radiator surface area would have to be increased, around the habitat. Thus the VDA option is discarded in the scope of this study. Kapton® and BetaCloth show similar performance during the transfer and lunar night. The difference is during lunar day: The Kapton option also absorbs heat flux from the environment while BetaCloth option does not. Consequently, the BetaCloth option requires smaller radiators on the Moon than other options.

MLI with BetaCloth as external layer and effective emissivity of 0.1 is going to be considered as a baseline thermal protection for the habitat and as calculation baseline for further computations in this study

The habitat needs to be covered with MLI. The MLI needs to be applied in a way to allow for the deployment of the habitat.

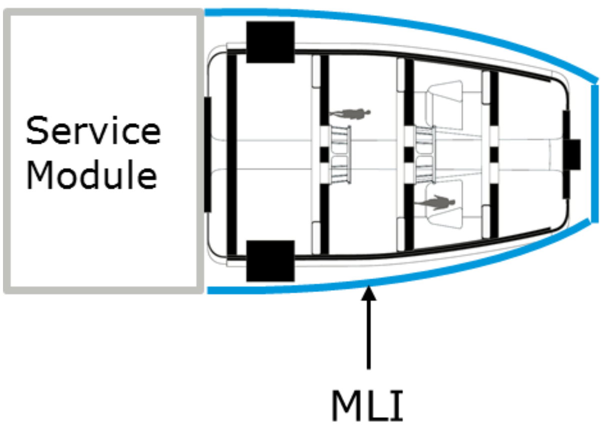

Figure 9-1: TCS Concept for Transfer phase

9.2.1.2 Heater power estimation

After the selection of the MLI concept, a heater power estimation is done according to requirement TC-1.

For the estimation of heater power, it was assumed that the insulation is provided by MLI only. All other layers are neglected, because their final configuration was not known during the assessment. Moreover, all materials are going to be in stowed configuration during transfer to the Moon – it is not yet known how the folding will affect the thermal performance of the lay-up. Thus, the calculations here assume the worst case conditions.

The heater power needed for the transfer depends on the temperature that needs to be kept inside the habitat.

The relation between minimum internal temperature to be kept and heater power needed, can be seen on the graph below. The heater power was estimated for cold case (no solar flux, heat exchange with deep space only).

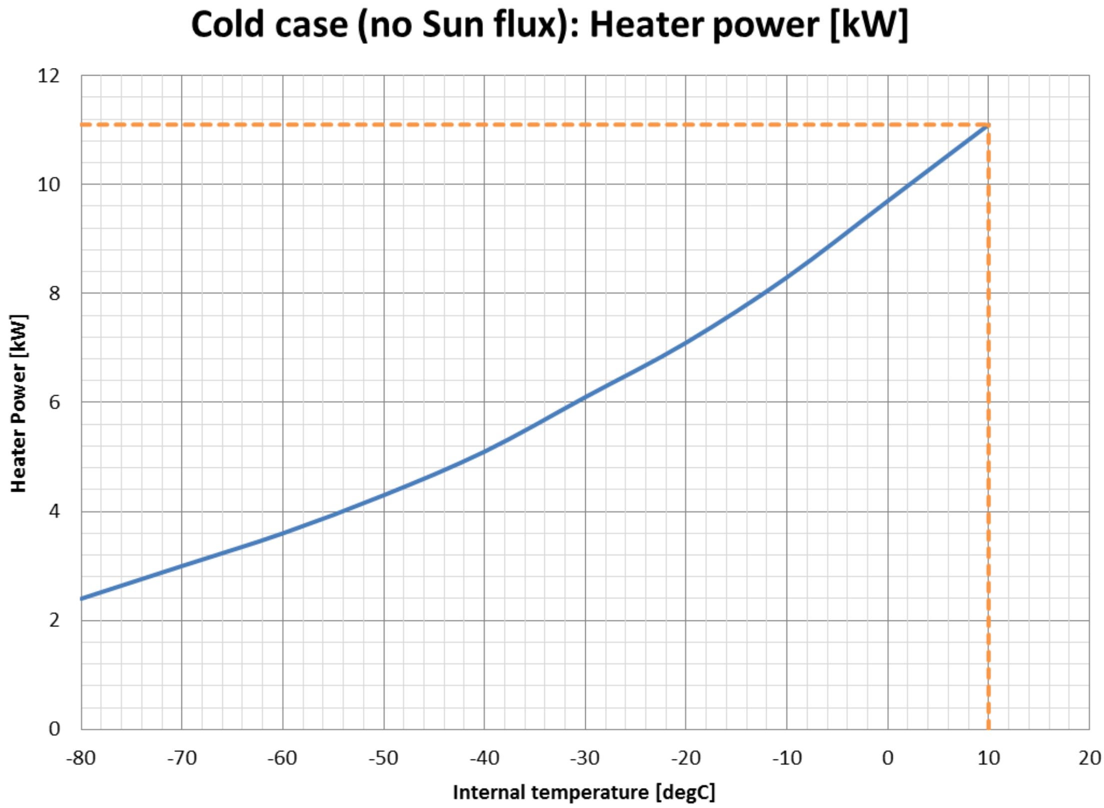

Figure 9-2: Heating Power vs Habitat Internal Temperature

It can be seen that to keep the inside of the habitat at 10degC (temperature required for ECLS equipment transportation), around 11kW of heating power is needed.

However, not all parts of the habitat need such a high temperature. Thus, in order to save some heating power, a part of the equipment inside the habitat (like ECLS equipment) is going to be extra insulated. This approach allows to keep the sensitive equipment close to a required temperature of about 10degC, while the temperature of the rest is going to be lowered to -20degC.

The "internal warmer compartment" is assumed to be wrapped with MLI with eps*=0.1 and VDA external layer to limit the radiative heat exchange. It is also assumed that there is a conductive link between the insulated compartment and non-insulated compartment.

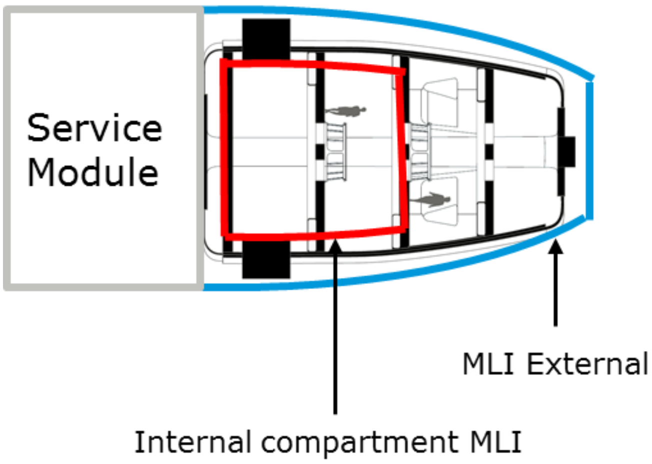

Figure 9-3: TCS Concept for Transfer phase

Table 9-6 shows the heating power needed in this configuration for different sizes of an internal compartment. The total estimated power needed for the heaters is around 7.2kW.

It can be noticed that change of the size of the internal compartment does not change the total heater power needed. This is caused by the limited fidelity of the used thermal model at this study stage. Due to the fact that in all the cases the internal side of the habitat has a temperature of -20°C, the same amount of power is needed to compensate heat losses to deep space.

Size of internal compartment [% of the total volume]	Cold case (no Sun flux)		Total [kW]
	Insulated compartment [kW]	**Non-insulated compartment [kW]**	
50	3.4	3.8	7.2
40	2.8	4.4	7.2
30	2.2	5.0	7.2
20	1.6	5.6	7.2
10	1.0	6.2	7.2

Table 9-6: Heating Power

The advantage of this solution, comparing with the case with no internal compartment, is that power needed to keep part of the habitat at 10°C is significantly lower than power needed to keep the whole habitat at this temperature (7.2kW comparing to 11kW).

Further improvements are possible by improving MLI efficiency or further decoupling between internal compartment and the rest of the habitat.

9.2.1.3 Thermal hardware for Transfer Phase

To sum up, the TCS needed for the transfer includes MLI blankets for insulations and heaters and thermistors for thermal control inside the habitat.

TCS hardware	Description
MLI blankets	External blankets + Internal blankets ε*=0.1, lay-up 20-25 layers (TBC)
Heaters	Total heater power: 7.2kW Heaters mounted in whole habitat
Thermistors	For temperature monitoring and heater control (in case of S/W controlled heaters)
Thermostats	Passive thermal control of survival heater line actuation.

Table 9-7: Thermal Hardware for Transfer Phase

9.2.2 Baseline Design for Usage

The baseline for the thermal design of the habitat is driven by the narrow temperature range required by the presence of humans, the size and total dissipation of the habitat, and the challenging lunar surface environment. In the following, these three aspects will be discussed and the baseline design will be presented.

9.2.2.1 Thermal lunar surface conditions

The temperature on the surface of the Moon ranges from 30 K in permanently shadowed craters at the poles to about 395 K at the sub-solar point on the lunar equator RD[38]&RD[39]. The lunar regolith works as an insulating blanket that covers the Moon. It has a very low thermal conductivity ranging from range 0.009 W·m-1·K-1 to 0.035 W·m-1·K-1 [RD[40], RD[41], RD[42]], depending on temperature, local density and mineralogy. The regolith furthermore has a high solar absorptivity of around 0.7 to 0.9 [RD[39], RD[43]] and an emissivity of about 0.92 to 0.98 [RD[39], RD[43]]. Thus the Moon is almost a black body from a thermal perspective with a poor thermal transport on its surface. On top of that, the axis of rotation of the Moon is only inclined by about 1.5 deg, which leads to almost no seasons. The Moon is in a bound rotation with the Earth which leads to a total duration of a lunar day of approx. 29.5 Earth days. The Moon does not have an atmosphere to buffer heat exchange with deep space.

All the factors mentioned above lead to the yields in the fact that the temperature of the lunar surface is driven mainly by the local angle of incident of the Sun. Due to the poor thermal conductivity and the low rotation speed of the Moon, the local temperatures can be assumed to be steady state at any given moment in time with good approximation for the purpose of thermal engineering. This also shows the impact of the local topography on the to-be-expected temperatures. Although global maps of the Moon show a decrease in temperature towards the lunar Poles (see Figure 9-4), local temperatures are dominated by the angle between the local surface normal and the angle of incidence of the Sun. Thus, even at the South Polar Regions temperatures of up to 370-380 K can be encountered, as can be seen in Figure 9-5.

For this study, LRO Diviner data was used to derive the temperature at the proposed habitat location at the lunar South Pole. As can be seen in Figure 9-6, the local surface temperature ranges from 65 K to approx. 295 K. These values are used subsequently for the computation of heat losses during lunar night and heat loads on radiators during the lunar day.

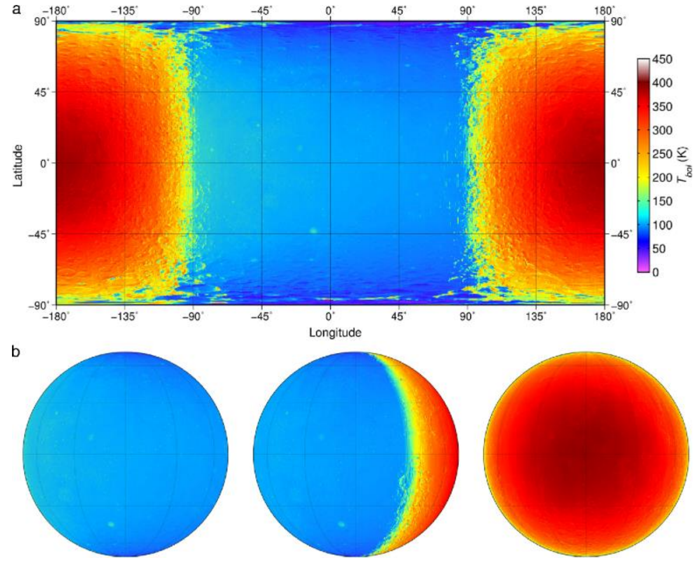

Figure 9-4: Temperature of the Moon as derived by Lunar Reconnaissance Orbit (LRO) Diviner instrument data [RD[44]]

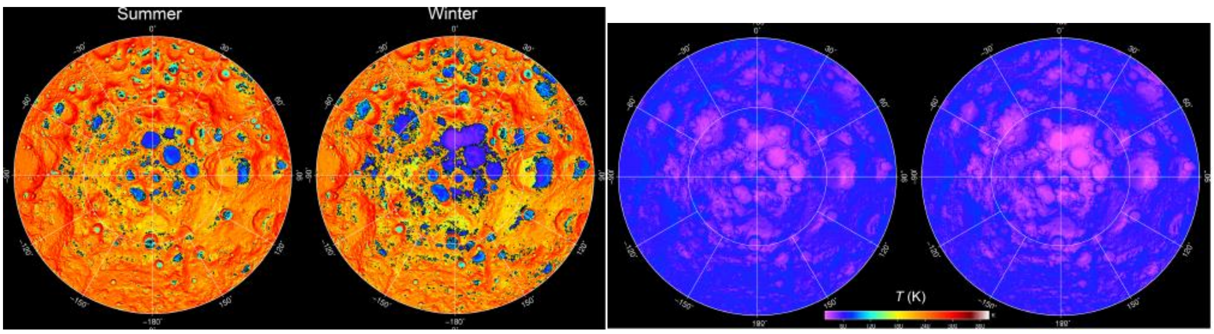

Figure 9-5: Lunar South Pole temperatures for summer and winter, Left: maximum diurnal temperatures, Right: Minimum diurnal temperatures [RD[45]]

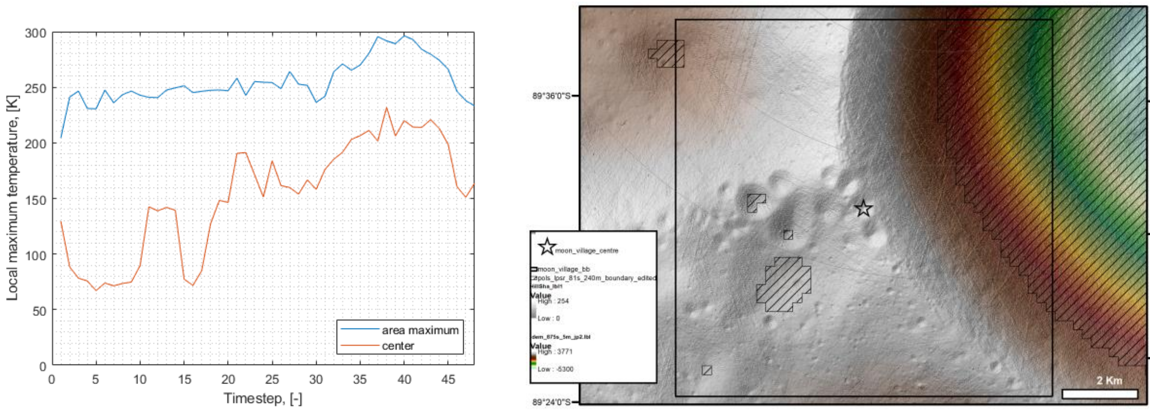

Figure 9-6: Temperature in the vicinity of the habitat location for the course of one arbitrary lunar day

9.2.2.2 Impact of lunar dust on TCS

The challenges of lunar dust in the return to the Moon have been addressed by many authors. For the thermal control subsystem the biggest impact is the degradation of optical surface properties. There are several reports from the Apollo and Luna missions of degradation of radiator performance due to lunar dust. In studies performed to investigate the impact of lunar dust on radiators [RD[46]] it was identified that even small sub-mono layers of lunar dust (simulant material) increased the absorptivity of white paint and second surface mirrors. A linear relationship between dust coverage and increase in α/ε ratio was concluded. As such it was assumed for the purpose of this study an increase of α by a factor of 2 and a maximum emissivity of dusted radiators of ε = 0.97 which is the maximum emissivity of the regolith.

9.2.2.3 Habitat heat losses

For the computation of the habitat heat losses it is assumed that the interior of the habitat needs to be maintained at 22°C at all times, due to the presence of humans. Based on the assumed MLI efficiency of eps*=0.1 and the external optical surface properties of BetaCloth the following heat losses were derived for the shape and surface area of the habitat:

	Internal temperature	Heat dissipated through the walls of the habitat
Lunar night (surface temp 65K)	22˚C	17.6kW
Lunar day (average (half in the shade, half illuminated) surface temp 210K)	22˚C	0.1kW

Table 9-8: Habitat Heat Losses

9.2.2.4 Radiator sizing

The sizing of radiator areas for the habitat was based on the following assumptions. The backside of the radiator was assumed to be perfectly insulated from any heat exchange with the ground. The radiator temperature was set to 15°C. The radiator temperature is computed by the maximum allowable temperature internal, i.e. 25°C in very hot cases, plus a delta temperature of 10°C between the habitat and the radiator. As mentioned in section 9.2.2.1, the maximum lunar surface temperature was assumed to be 295 K. The dissipated power internal to the habitat was assumed to be 57.4 kW (including system margin). A margin on the radiator surface area of 20% was accounted for. Furthermore it was assumed that the radiator is covered with Optical Solar Reflector (OSR) tiles, which is the most effective radiator material at the moment.

The view factor between the surface of the Moon and the radiator is a function of the radiator's inclination. As can be seen in Figure 9-7, there is a run-away effect between radiator surface area and radiator inclination. The radiator area is a function of internal heat dissipation and external heat fluxes. The external heat fluxes are made up of solar heat flux, solar albedo heat flux, and IR heat flux. The more the radiator is tilted the higher get all three contributions. Due to the location of the habitat at the lunar poles, inclining the radiator will also lead to a larger amount of heat being absorbed from direct solar illumination. Yet, the largest contributor is the IR heat flux from the Moon. This is because of Kirchhoffs law, stating that emissivity = absorptivity for a given wavelength range. In case of radiators on the Moon this means that the surface of the Moon is at a similar (or higher) temperature than the temperature of the radiators. Thus the efficiency of the radiator decreases to the point where it even might absorb heat rather that rejecting it.

As a result from Figure 9-7, the inclination of the radiator should be restricted to angles below 15°, otherwise the radiator surface area becomes prohibitively large

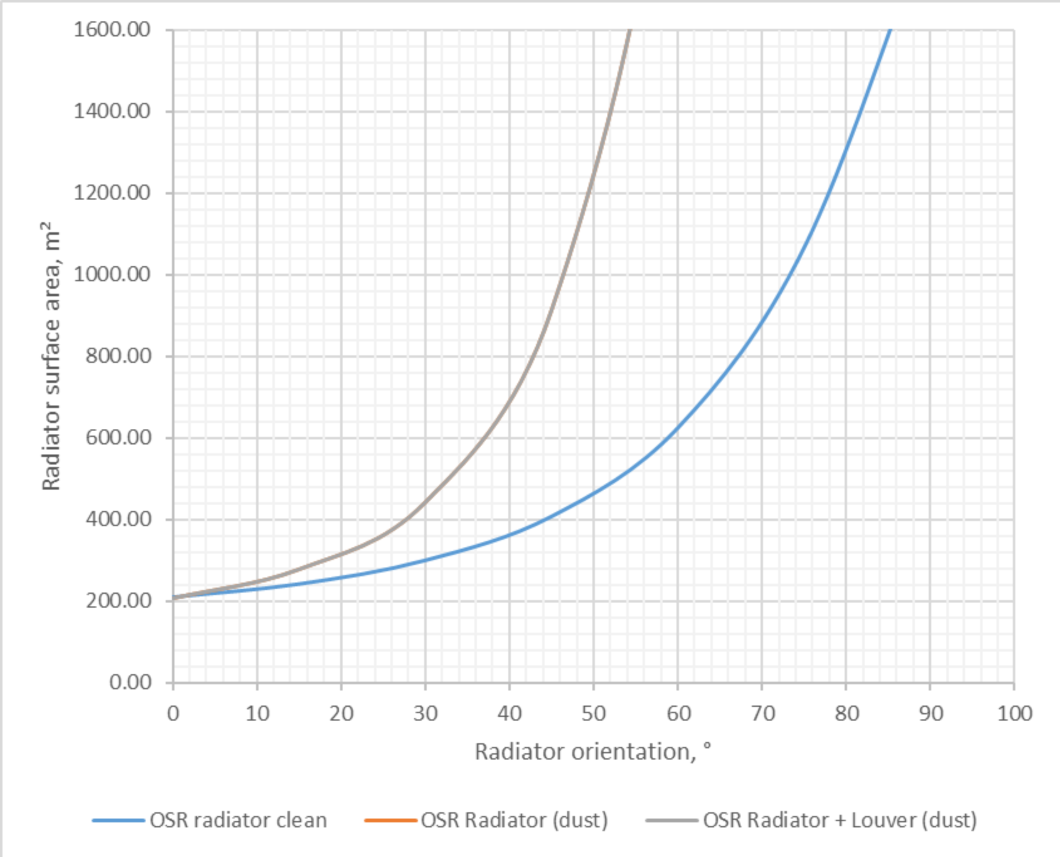

Figure 9-7: Radiator surface area in hot case, including 20% margin on the area of the radiator. Radiator orientation of 0° means the radiator is parallel to the surface of the Moon Note: the line for dusted radiators (orange) is identical to the radiator with louver (grey) as they have the same emissivity in this simplified model.

9.2.2.5 Heater power demand

The radiator area is the input to the computation of the heater power demand during the cold phases of the lunar day, i.e. lunar night. For the purpose of computing the heater power demand, it is assumed that there is still a dissipation of 57.4 kW internal to the habitat. Furthermore, it is assumed that the internal temperature of the habitat is 20°C which is a worst case cold for the presence of humans. The radiator temperature was assumed to be 10°C, i.e. to have a delta of 10 K between the interior and the radiator.

The TCS has to compensate for the 17 kW of heat losses through the MLI and all the losses caused by the radiators (see Table 9-8). Figure 9-8 shows the resulting heater power demand, as a function of the orientation of the radiator. The radiator area was determined by the back-load from IR and solar heat fluxes.

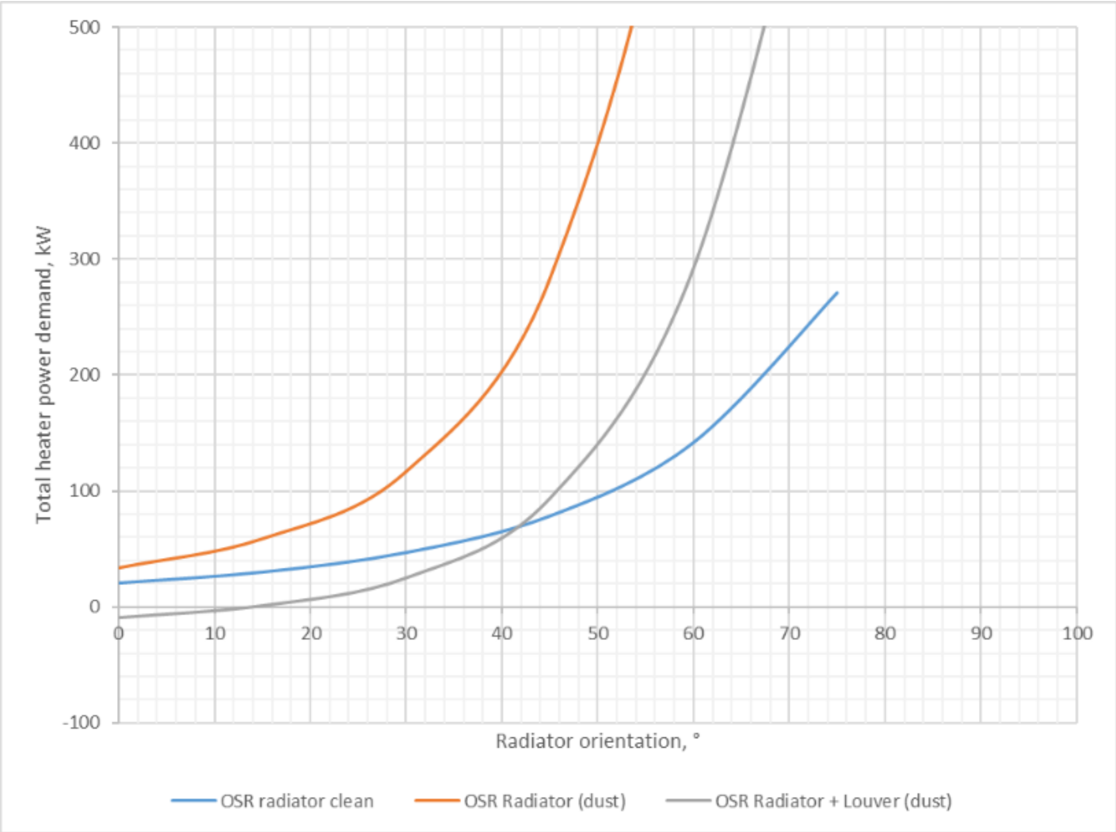

Figure 9-8: Heater power demand in cold case, assuming 57.4 kW internal dissipation and 17 kW heat losses through the MLI

It can be seen from Figure 9-8 that a clean radiator will perform better than a dusted radiator, this is due to the fact that the absorbtivity and emissivity are smaller. In the hot case this leads to a smaller amount of absorbed heat flux and in the cold case the lower emissivity leads to less heat losses. For comparison, an option is shown in which a louvered radiator is used. Theoretically a louvered radiator is able to reduce the emissivity in the cold case, hence the heat losses and as such also the required heater power.

Table 9-9 summarises the radiator surface area and heater power demand for radiator inclinations of 0° (parallel to lunar surface), 15°, and 30°. Three different alpha/epsilon options are compared, i.e. a radiator with clean OSR tiles, a radiator with dusted OSR tiles and a louvered radiator with dusted OSR tiles.

	α	ε (open/ closed)	Area [m²] @ T=15°C internal temperature and local noon (worst case hot)			Heater power demand [kW] for the computed radiator area @ 10 °C internal temperature and no external heat fluxes (worst case cold)			Area specific radiator mass [m²/kg]†	Radiator total mass [kg]		
Radiator tilt to local surface normal			0 °	15 °	30 °	0 °	15 °	30 °		0 °	15 °	30 °
OSR Radiator / clean	0.10	0.80	209	241	299	31	40	57	17.00	3552	4100	5085
OSR Radiator / dust	0.20	0.80/0.97	208	279	444	43	68	126	17.00	3538	4751	7548
OSR Radiator + Louver‡ / dust	0.20	0.8/0.4	208	279	444	0.1	10	34	21.50	4475	6008	9545

Notes:

† Radiator mass based on ISS heritage; increase in mass due to different environments μ-g vs. 1/6-g possible.

‡ Louvered radiator mass based on Rosetta (Louver) heritage

Table 9-9: Summary of radiator / heater sizing options

9.3 Budgets

Estimation of the mass of TCS is based on assumption that it is around 6% of the total habitat mass, where 1% is mass of MLI and Heaters and 5% is the mass of Active Thermal Control System (ATCS).

Moon Village	Mass [kg]	% of total mass
Full Habitat (early estimation)	50000	100%
TCS Total	3000	6%
TCS: ATCS	2500	5%
TCS: MLI+Heaters	500	1%

Table 9-10: Habitat TCS Mass

Moreover, mass of the surface radiators has been estimated:

Moon Village	Mass [kg]	
Radiators	4475	(louvered radiators)

Table 9-11: External Radiators Mass

Heater power needed for both phases is presented below:

	Heater Power [kW]
Transfer	7200
On the Moon	0.1*

*** Louvered radiators baselined**

Table 9-12: Required Heater Power

Due to limited data during the study, current estimations do not include power needed for ATCS.

9.4 Options

Options for a reduction of heater demand:

The heater power demand is driven by the radiator area. For a given radiator area the heater power losses are defined by the radiator temperature and its emissivity.

A possibility to adapt the emissivity of the radiator is to foresee louvers, as it was done for example for the Rosetta mission. A louver reduces the emissivity of the radiator in cold cases and thus strongly reduces the heat losses. Other options are to use coatings with variable emissivity such as electrochromics (ECH) or thermochromics (TCH). These change the emissivity either actively or passively which allows to have a high dissipation in hot case and a reduced heat rejection in cold cases. Variable emissivity coatings usually have the disadvantage of a higher solar absorptivity than OSR, SSM or white paint. This makes ECH or TCH only interesting if they are protected from direct solar illumination.

A different approach is to use lower fluid temperatures. This requires a dedicated heat exchanger and a fluid that can cope with lower temperatures without freezing. The disadvantage of such fluids is often their toxicity for humans.

A further option is to switch-off branches of the radiator, but also this requires a fluid that has a lower freezing point or a system that is tolerant to multiple freeze-thaw cycles.

The surface area of the radiators is larger than the surface area of the habitat. Thus, in all cases a dedicated external thermal control 'hub' will be necessary to which the habitat or several habitats can be connected. This thermal control 'hub' will have to provide the pumping capability, the capability to distribute heat to several radiator loops, to receive heat from different habitat modules and potentially to host heat exchanger(s) between habitat internal and external radiator loops

9.5 Technology Needs

Included in this table are:

- Technologies to be (further) developed
- Technologies identified as coming from outside ESA member states
- Technologies available within European non-space sector(s)

Technology Needs						
*	**Equipment Name & Text Reference**	**Technology**	**Supplier (Country)**	**TRL**	**Funded by**	**Additional Information**
	Louvered Radiators	Temperature activated (passive) louvers in front of a radiator	SENER	9		Last flown on Rosetta (hence lower 'real/current' TRL) Active mechanism might be necessary for Moon application
	Two phase – Loop Heat Pipes	Loop heat pipe system; also heat diode function / switchable	IberEspacio / European Heat Pipes	2		Nominal LHP are flying, but dedicated Moon specific LHP would have to be developed (lower cold case temperatures)
	Electrochromics	Application of voltage leads to change of transmissivity (active)	Thales –F, CAE, IREIS, ICMCB	2		TRP activity about to finish / still high solar absorptivity; Delta emissivity ≈ 0.3
	Thermochromics	Vanadium-dioxide change in transmissivity based on temperature (passive)	Thales –F, LPPI,	3		TRP activity about to finish / still high solar absorptivity; Delta emissivity ≈ 0.3

* Tick if technology is baselined

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10 RADIATION

Space radiation effects on astronauts and spacecraft systems are one of the main concerns for long-term human lunar missions. At Earth the magnetosphere and thick atmosphere provides a substantial protection against space radiation. The Moon, however, has a tenuous atmosphere, no global magnetic field to uphold a magnetosphere of its own, and only around 25% of the Moon's orbit is found within the magnetosphere of Earth. Hence, the natural protection against space radiation is insignificant. For this study the main radiation sources are Galactic Cosmic Rays (GCR) and Solar Particle Events (SPE). Prolonged exposure to the radiation belts of Earth could potentially be an additional problem for the crew. However, this can easily be avoided by keeping the Earth-Moon transfer time to a minimum. Hence, the focus in the report is the radiation exposure due to GCRs and SPEs on the lunar surface.

10.1 Overview of Space Radiation Effects On Humans

Space radiation is a major barrier to human exploration of the solar system because of the poor understanding of biological effects of high energy and charge (HZE) ions, which are the main contributors to radiation risks in deep space. Predictions of the nature and magnitude of the risks posed by space radiation are subject to very large uncertainties. Great efforts have been dedicated worldwide in recent years toward a better understanding of the oncogenic potential of galactic cosmic rays RD[48]. However, further studies are needed to understand the effects of the wide range of heavy ions, energies and fluencies.

Radiation damage to biological systems includes direct damage, when radiation interacts directly with DNA, but the most common process is indirect damage, when radiation mainly interacts with H2O and creates free radicals that in the end will interact with DNA. Health effects can also be divided into acute and delayed. Delayed effects include cancer and genetic effects. Acute effects occur within a few days or less and includes vomiting, nausea, loss of appetite, and fatigue.

The space radiation effects on humans can be classified into two main categories:

- **Stochastic effects** (cancer, leukaemia, hereditary effects)
  - No threshold dose, exposure provide an increased risk
  - Probability of the effects increases with the dose, not the severity
  - No definitively associated with the radiation dose received
- **Deterministic effects** (cataracts, dermatitis, sterility, radiation syndrome, etc.)
  - Threshold dose, above which they always appear
  - Damage grows usually with the dose intensity
  - Typically they manifest soon after exposure.

Deterministic effects are significantly related to Solar Particle Events and they can produce a big impact on the astronauts' health causing the failure of the mission.

HZE produce densely ionizing tracks causing significant damages and complex DNA breaks: "clusters" containing mixtures of damaged biological material. Under these conditions, the DNA repair is more difficult and the cell death is more frequent RD[48].

For space exploration, it has been concluded that new experiments are needed, and they should focus on late effects at low doses.

10.2 Radiation Dose Limits and Shielding Requirements

Radiation limits set by ESA for LEO missions are shown in Table 10-1. Career effective dose limits from NASA given in mSV for a 1-year mission are listed in Table 10-2, the average life-time loss due to the radiation exposure is also included within brackets. The ECSS-E-10-04 space environment standard provides additional limitations and recommendations. Comparisons with other Space Agencies dose limits can be found in RD[47].

Limit	Value	Comment
Career	1 Sv. (1000 mSv)	ICRP—no age or gender dependence
Blood Forming Organs (BFO)	0.25 Sv. for 30 d 0.5 Sv. for annually	ISS consensus limits
Eye	0.5 Sv. for 30 d 1.0 Sv. for annually	
Skin	1.5 Sv. for 30 d 4.0 Sv. for annually	

Table 10-1: ESA dose limits, from RD[47]

Age, yr	Career effective dose limits in units of mSv for 1-year missions (Av. Life Loss in years)	
	Males	Females
30	620(15.4)	470(15.7)
35	720(15.4)	550(15.3)
40	800(15.0)	620(14.7)
45	950(14.2)	750(14.0)
50	1150(12.5)	920(13.2)
55	1470(11.5)	1120(12.2)

Table 10-2: Examples of career effective dose limits for male and female astronauts by NASA. Corresponding estimates of average life-time loss due to radiation exposure are shown in brackets. Table is obtained from RD[47]

10.3 Radiation Dose Estimates

The fluence and dose vs. depth curves were computed using the OLTARIS (On-Line Tool for the Assessment of Radiation in Space) RD[49].

The types of dose considered for the study are:

- Effective Dose: tissue-weighted sum of the equivalent doses in all specified tissues and organs of the human body. It represents the stochastic health risk to the whole body.
- Blood Forming Organ (BFO) dose: equivalent dose for the specific organ that is responsible for the acute radiation syndrome.

More details on the models, the tool and radiation quality factor are available on the OLTARIS website RD[49].

10.4 Galactic Cosmic Rays

Galactic Cosmic Rays (GCRs) are a continuous flux of very high-energy particles arriving isotropically from outside of our Solar System. The energies range from less than 1 MeV/u to more than 10,000 MeV/u with a median energy of around 1,000 MeV/u RD[50]. Around 99% of GCRs are protons and alpha particles RD[51]. The remaining 1% are composed of fully ionized nuclei of all kinds of charges from lithium to uranium, but with a strong decrease in the flux of particles with charge higher than 28 RD[51]. The GCRs with energies less than 2,000 MeV/u varies with the solar cycle, at solar minimum the GCR fluxes are about twice as large as at solar maximum RD[50].

The GCRs, if not shielded, produce the largest contribution to the total effective dose (>500 mSv/year) in a long duration lunar mission and shielding from GCR is still an open challenge due to their capability to produce secondary particles when interacting with matter as illustrated in Figure 10-1.

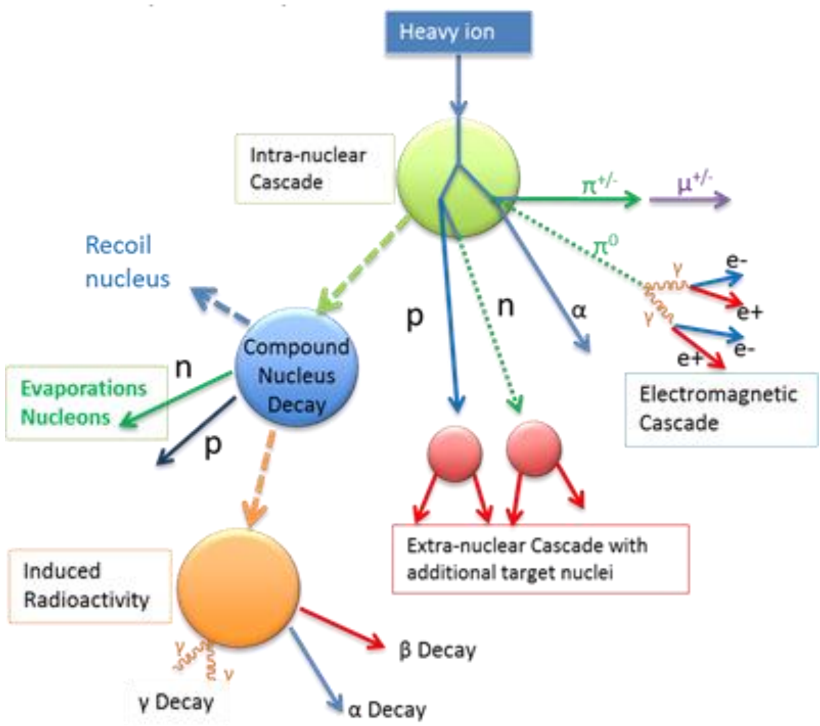


Figure 10-1: Production of secondary particles due to the interaction of radiation with spacecraft structures

For this study the GCR spectra from RD[52], referred to M2012, is used. They studied measurements from the Cosmic Ray Isotope Spectrometer (CRIS) on-board the Advanced Composition Explorer (ACE) spacecraft, measured during the time period 1997 to 2012, and the Oulu neutron monitor to develop a GCR flux model. The model predictions agree well with GCR measurements of an extended ACE/CRIS data set, results from the balloon experiment BESS and measurements from the Isotopic Composition of Primary Cosmic Rays Experiment (C-2) on-board the High Energy Astrophysics Observatory (HEAO-3) spacecraft RD[52]. The Effective Dose Equivalent (in blue) and the BFO Average Dose Equivalent (in red) obtained using the M2012 model is presented in Figure 10-2. The dose equivalents are for the lunar surface.

There are also other GCR flux models available, for example the Badhwar-O'Neill 2010 model (BON2010) RD[53], which is also included in Figure 10-2 (dashed lines) for a comparison. As can be seen in the figure the two models provide different dose rates for the same material thickness. The M2012 GCR flux model was used in the Moon village habitat analysis since it provides the highest dose rates of the four different models available on OLTARIS, so a conservative estimate is being made, and the model predictions are in good agreement with earlier GCR flux measurements RD[52].

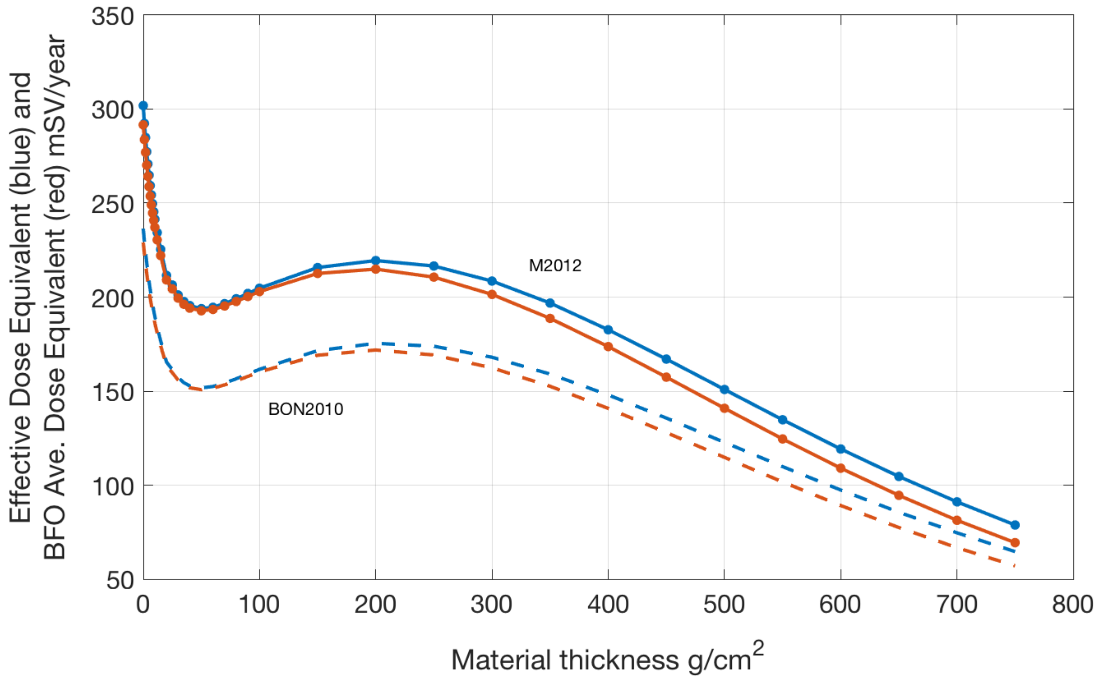


Figure 10-2: Effective Dose Equivalent (blue) and BFO Average Dose Equivalent (red) as a function of the shielding thickness, for the lunar surface and obtained from two different models. The model from Matthiä et al. 2012 RD[52] (solid lines with round markers and marked M2012) is used for this study. The so called Badhwar-O'Neill 2010 model RD[53](dashed lines and marked BON2010) is included in the figure for comparison. The effect of the passive shielding against GCR is limited and increasing the shielding thickness up to around 50 g/cm2 progressively reduces its efficiency. The increased production of secondary particles for increased material thickness explains why 50 g/cm2 provides better shielding than 200 g/cm2.

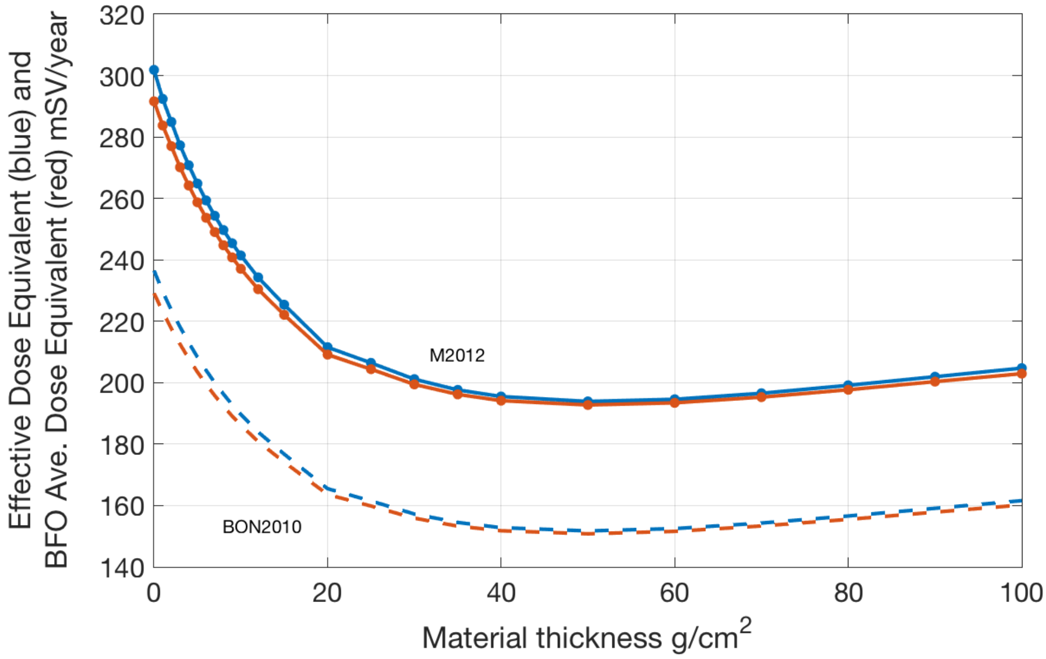


Figure 10-3: Same as in Figure 10-2: but with material thickness up to 100 g/cm².

10.5 Solar Particle Event Selection and Dose Estimates

Solar Particle Events (SPE) occur when protons from the Sun gets accelerated by solar flares or Coronal Mass Ejections (CME) shocks, to energies between around 1 MeV and a few hundred MeV. The events also include smaller amounts of helium and heavier ions. The number of SPE varies with the solar cycle, which is an 11 year cycle that marks the changing activity of the Sun. SPEs occur around 5 to 10 times per year, except near solar minimum RD[50].

SPEs produce large fluxes of solar energetic particles (SEPs) that are encountered in space as enhancements many orders of magnitude above the background GCR levels. The Earth's magnetic field provides a varying degree of geomagnetic shielding depending on the particle rigidity, the location of the spacecraft and the viewing direction whilst the atmosphere also acts to attenuate particles. However, as the Moon lacks a global magnetic field and a dense atmosphere the attenuation of SEPs are negligible.

The understanding of SPEs is still very limited. This is due to the sporadic nature of SEP events and the complexity of the mechanisms involved in their generation and evolution (i.e. acceleration and propagation), the nature of the underlying physical processes and the lack of detailed, spatially distributed in–situ observations. For this reason, forecasting and/or nowcasting systems are considered a major challenge for Moon, Mars and deep space missions. ESA and the other Space Agencies are deeply involved into these studies and developments of new forecast systems RD[56].

Since the Moon Village habitat has to provide an adequate safe shelter for the crew, the selection of the SPEs, to be used for the dose computations, plays an important role for the radiation shielding design. For this study a set of past event were selected and the most representative have been used to derive dose vs. depth curves. Figure 10-4 shows the differential proton fluence as a function of energy for a selection of relevant events. Some of the events are considered possible but extremely rare (e.g., the Carrington event) and most of the events are heavily based on data extrapolation and as a result, in some cases, the spectral shape does not appear physically consistent in the full energy range (e.g., February 1956 - LaRC model). The high-energy range is very important to determine the minimum shielding thickness for the shelter in order to satisfy the 30 days dose limit, of 250 mSv for the BFO, reported in Table 10-1. The low energy range becomes the main dose contributor in case of Extra Vehicular Activities (EVA) and surface operations on the Moon that are not described in this document.

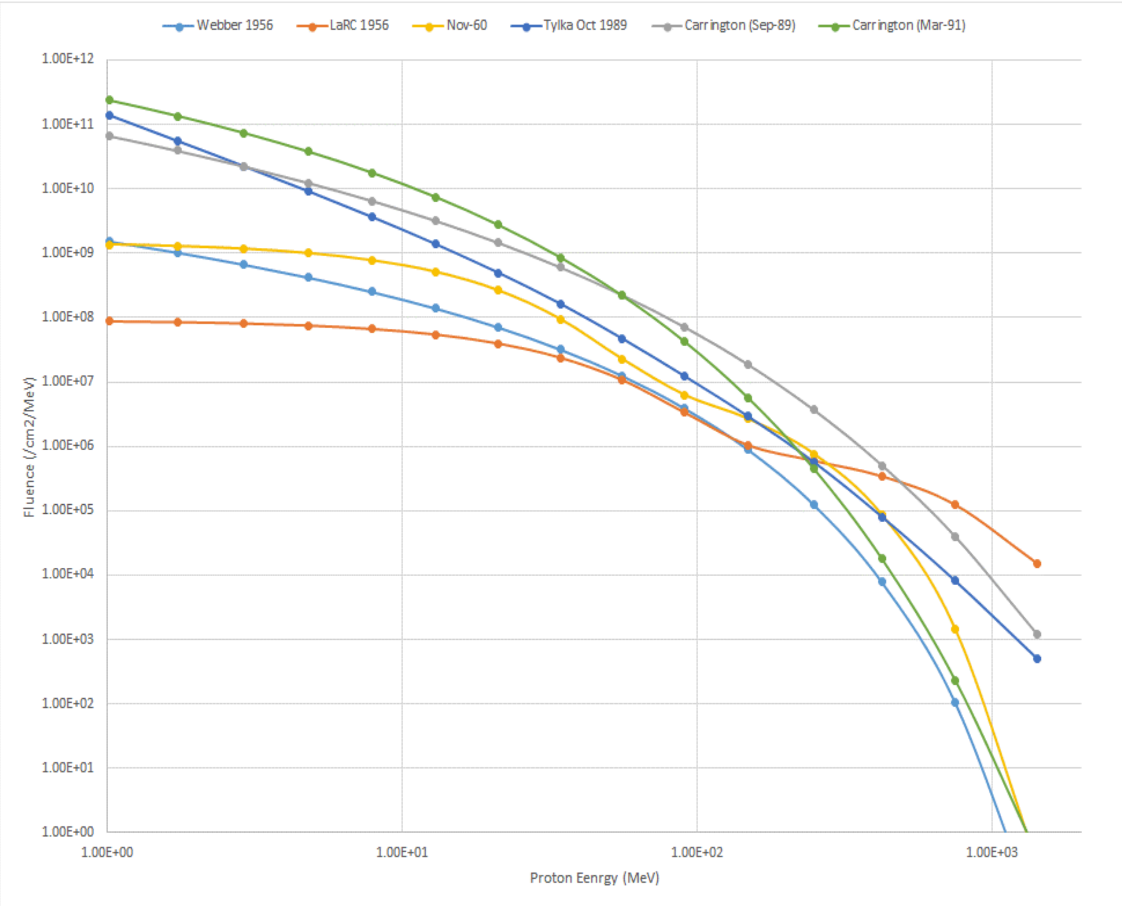


Figure 10-4: Differential Proton Fluence as a function of energy for a selection of relevant Solar Particles Events. The fluences are obtained from OLTARIS RD[49]

In order to adopt a safe and conservative approach, two main events will be considered in this report: Sum of October 1989 events (Tylka model RD[54]) and September 1859 (Carrington - September 1989 hard fit RD[55]).

Figure 10-5 shows the effective dose and Blood Forming Organ (BFO) dose for the Carrington (Sept. 89 hard fit, RD[55]) and Sum of Oct.89 (Tylka, RD[54]) events. No margin has been applied to the Carrington event (being an extreme case) and a margin of 2 has been applied on the October 89 dose. The dose is a function of the shielding thickness for aluminium in a spherical shell configuration. The 30 days and annual dose limit for BFO are reported for comparison. The dose shown in the plot are for deep space missions while, for the studied habitat, the Moon surface will provide a natural shielding on half of the solid angle and the actual dose will be reduced by a factor 2. Results for the Moon Village habitat configurations are presented later in the document.

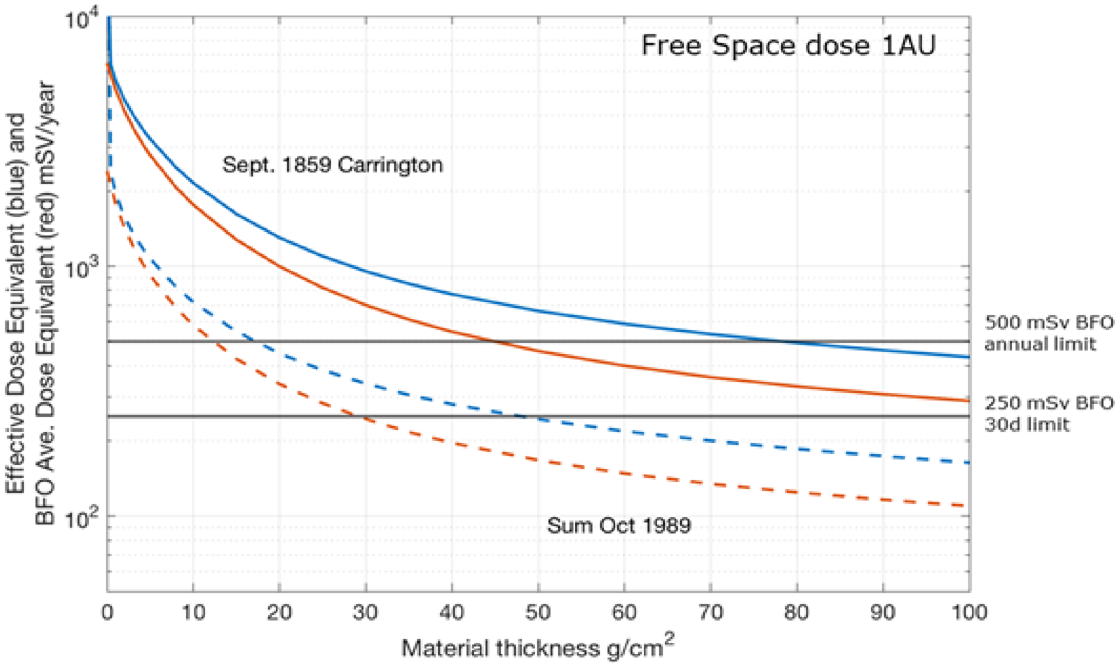


Figure 10-5: Effective Dose and BFO equivalent dose as a function of shielding thickness RD[51]. The presented October 89 dose includes a margin of 2. For the estimated effective dose equivalent at the Moon's surface, the values should be divided by 2

10.6 Radiation Shielding for the Moon Village Habitat

Current strategies for the risk mitigation from space radiation mainly involve the use of different thicknesses of shielding materials (including structural materials, instrumentations, food and water supplies, etc.). Shielding sizes and features must necessarily be related to the mission profile and spacecraft design, which determines the radiation environment impacting on the astronauts.

Different shielding scenarios can be envisaged, as:

1. A uniform shielding of the entire space habitat to reach the minimum areal density of shielding materials. This possibility is greatly demanding in terms of the amount of material.

2. Creating a safe area (shelter) in the space habitat, characterized by an increased wall thickness for shielding purposes, where astronauts can spend a great part of their time (e.g. crew quarter). This approach is **recommended** and can be a good compromise in terms of mass and efficiency.

3. Finally, creating a micro-shelter, i.e. a very small shelter inside a pressurised habitat: as a drawback there is the possibility to spend entire days in a narrow space (SPEs may last up to 10 days).

For the analysis presented in this report only option 2 has been considered.

10.6.1 Configurations Analysed During the Study

The habitat model used for the analysis is presented in Figure 10-6. The material choice for the inflatables are as stated in the Chapter Safety, Material and Processes. The chosen materials provide an area density of 1.7 g/cm². The main properties of the inflatables are listed in Table 10-3 row 1. The inflatables include a foam which is covering 10% of the total area, this provides a thickness of the inflatables of 25 cm. If the foam, however, were evenly distributed over the whole area the material thickness would be only 4.4 cm. The inflatables do not provide sufficient radiation shielding on their own. This is clear from comparing the area density of 1.7 g/cm² to the Effective Dose Equivalents presented in Figure 10-2, Figure 10-5, and Figure 10-7. Despite this, the habitat without radiation shielding has been included in the ray tracing simulations, as a reference and the results are listed in Table 10-4 and Table 10-5 as Configuration 1.

The largest surface areas exposed to the surrounding radiation environment are the inflatables (Figure 10-6, beige), since they do not provide sufficient radiation shielding the focus of the study has been to include and optimise additional radiation shielding within or around the inflatables. The habitat model also includes other structures that provide slightly more shielding, but they are unevenly distributed over the habitat and the additional radiation shielding that they provide is therefore not sufficient to reduce the dose to an acceptable level. The additional structures of the habitat model are a support structure and floors (Figure 10-6, green), 12 windows (light blue), crew quarters and storage facilities (yellow) and a top hatch (light grey). The total mass of the studied habitat model is around 47 tons.

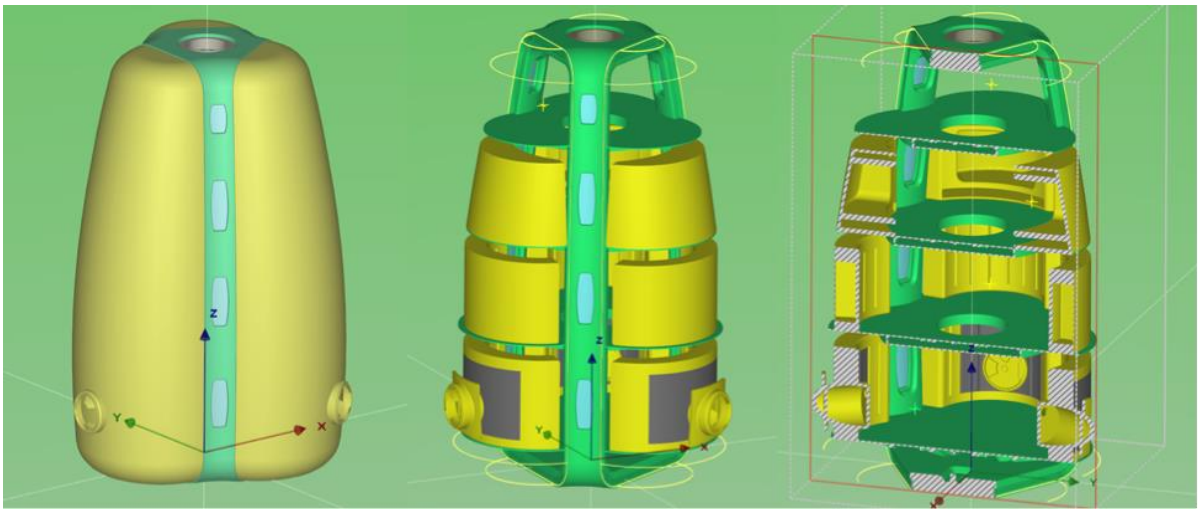


Figure 10-6: The Moon village habitat including the inflatables (beige), the support structure and the floors (green), the windows (light blue), the crew quarters and storage facilities (yellow) and the top hatch (light grey)

Three different configurations have been studied and assessed for this analysis:

- **Configuration 1:** Includes the habitat without any additional radiation shielding. Shown in Figure 10-6.
- **Configuration 2:** This configuration has been obtained as a result of a parametric study performed by iterating the Ray Tracing analysis, varying the shielding thickness and location. The resulting configuration includes the habitat plus the minimum shielding required to obtain a total BFO Average Dose of 500 mSv/year and 250 mSv/30 days. The additional material is used to:
  - Shield the entire inflatable structure, 2 cm of sintered lunar regolith (equivalent to 4 cm of loose regolith): this is the minimum thickness to get a BFO Average Dose Equivalent below 250 mSv/year in the most exposed part of the habitat, i.e., the top floor.
  - Create a sheltered area located in the ground floor with the minimum shielding required to obtain a BFO Average. Dose Equivalent of 250 mSv (corresponding to 30 days limit dose) during the chosen SPE. The shelter is obtained shielding the ground floor with an additional 20 cm of sintered lunar regolith (equivalent to 40 cm of loose lunar regolith) and including a water tank with around 10 cm of water in the shelter roof. The configuration is shown in Figure 10-7, left panel.
- **Configuration 3:** This is a safer option, used as a reference case to demonstrate the advantages of additional shielding by placing part of the habitat underground and surrounding the structure by 25 cm of sintered lunar regolith (equivalent to 50 cm of loose lunar regolith). In addition, a water tank with 20 cm of water covers the habitat. This configuration is shown in Figure 10-7, right panel.

	Thickness (cm)	Average density (g/cm^3)	Area density (g/cm^2)	Mass (kg)
Inflatables	25	0.7	1.7	7,205
Configuration 2 Loose (or sintered) lunar regolith covering all inflatable surfaces	4 (2)	1.5 (3)	6	25,238
Configuration 2 Loose (or sintered) lunar regolith for shelter	40 (20)	1.5 (3)	60	118,187
Configuration 2 Water (in roof of shelter or habitat)	10	1	10	~8,000
Configuration 3 Loose (or sintered) lunar regolith surrounding the habitat	50 (25)	1.5 (3)	75	492,798
Configuration 3 Water (in roof above habitat)	20	1	20	~16,000

Table 10-3: The characteristics of the inflatables and the habitat radiation shielding for different configurations

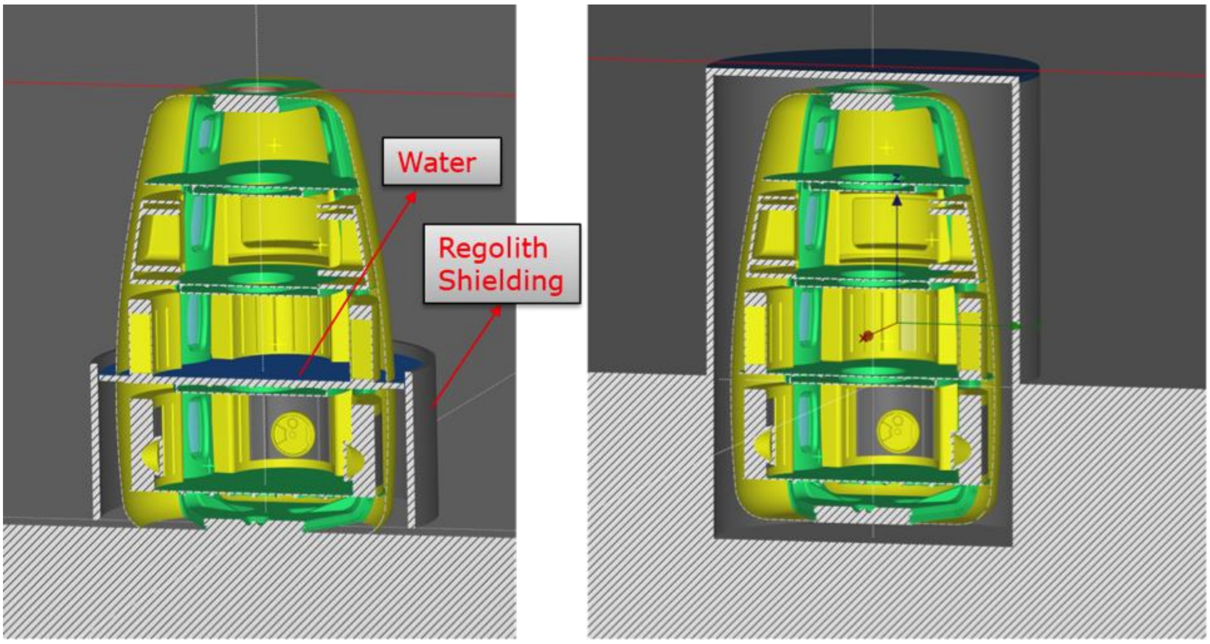


Figure 10-7: Two suggestions of the shielding of the habitat. The left figure shows Configuration 2 that includes lunar regolith inside or around the inflatables and an SPE shelter that consists of additional lunar regolith shielding and a layer of water on top of the ground floor of the habitat. The right figure shows Configuration 3 which includes additional measures; the ground floor has been lowered under ground, the whole habitat is covered with a thick layer of lunar regolith and the water is stored as a roof above the habitat. This version is not recommended for the studied habitat but is simply included to show the advantages of additional shielding and placing the habitat under ground.

10.6.2 Result of Radiation Shielding Analysis

The analysis, based on Ray Tracing, has been performed with FASTRAD for the habitat presented in Figure 10-6 and provides a first estimation of the dose for the different configuration described in 10.6.1. More complex analyses such as Monte Carlo simulations were not performed due to the large number of assumptions and parameters to iterate. For the final design of the habitat Monte Carlo simulations, which provide more accurate analyses, are recommended.

As a conservative approach, one of the largest ever recorded SPE, the Carrington event RD[55] has been used to assess the radiation shielding required in order not to exceed the dose limit in case of exceptional and rare SPEs. For comparison, the dose estimated with another more recent SPE, October 89 (Tylka model described in RD[54]), has been added to the results (within brackets) in column 1 Table 10-1 and Table 10-5. It should be noted that more moderate SPEs occur on average 5 to 10 times per year, except close to solar minimum. The analysis is performed with the assumption that the crew would spend the total duration of the event on the ground floor (shelter).

For the analysis the dose obtained in the ground floor of the habitat and the top floor is studied. An annual total dose have been obtained from the sum of the yearly dose due to GCRs, so the continuous flux of radiation, in the top floor of the habitat and the dose obtained from one extreme SPE, either the Carrington event or the October 89 event, in the ground floor of the habitat. This approach will give a slight overestimation of the GCRs dose due to the fact that the crew do not spend the whole year in the top floor of the habitat. However, in this study EVAs, which would provide an increased dose rate, have not been considered.

To estimate the minimum amount of shielding material to be placed in the habitat, the dose limits provided by ESA and presented in Table 10-1 were used: **the BFO dose equivalent may not exceed 500 mSv/year and 250 mSv/30 days**. With these dose limits the career of the astronaut will be limited to two years and the average life-time loss can be expected to be more than 10 years, as reported in Table 10-2. A safer and recommended shielding configuration is therefore more in line with what is proposed in Configuration 3.

Configuration 1:

Simulations on Configuration 1 are performed on the habitat model without any additional shielding. Row 1 Table 10-5 shows that the BFO average dose, of 720 mSv, at the ground floor during this SPE far exceeds the yearly BFO dose limits stated by ESA, Table 10-1, even without taking the radiation dose due to the GCRs into account. The total annual BFO average dose equivalent of 994 mSv is close to the total career dose of 1000 mSv for ESA astronauts, see Table 10-1. Even when considering the BFO Average dose equivalent from the Oct 1989 SPE, of 241 mSv, the annual total dose still exceeds the 500 mSv/year limit. From these simulation results, it can be concluded that the habitat needs additional radiation shielding.

The difference between the estimated BFO average dose equivalent, in Table 10-5, due to GCRs at the ground floor, of 225 mSv/year, and the top floor, of 274 mSv/year, shows the additional shielding that is due to the material, such as storage, which is located in the ground floor and the additional shielding that the upper floors provides for the ground floor.

The simulation results from configuration 1 clearly show that additional radiation shielding is needed. Using additional habitat material, such as extra layers of inflatables, water, or lunar regolith, could solve the issue. The disadvantage of adding additional fabric layers is the additional launch mass, costs, and risks. It is therefore recommended to use the materials already present on the Moon, such as the lunar regolith or rocks, to shield the habitat. For this study, it has not been assumed that any larger amounts of water are available for extraction from the Moon and the water used for the radiation shielding has to been launched from Earth. The water that would be needed for the life support of the crew could be stored in a tank located in the roof of the habitat or the shelter and hence act as additional radiation shielding. The amount of water needed for the minimum shielding, around 8 tons for configuration 2, has been chosen in order to be in agreement with the amount needed for life support, see Chapter Life Support.

Configuration 2:

The minimum shielding required to stay below the BFO average dose equivalent limits of 500 mSv/year and 250 mSv/30 days are listed in Table 10-3. For the surfaces covered by the inflatables a minimum of 2 cm of sintered lunar regolith or 4 cm of loose regolith, is needed. With this amount of shielding the BFO average dose equivalent from GCRs in the top floor of the habitat will reach 248 mSv/year, see Table 10-3 column 3. The ground floor has been converted into an SPE shelter with an additional 20 cm of sintered lunar regolith or 40 cm of loose lunar regolith, which results in a BFO average dose equivalent of 228 mSv for the Carrington SPE. Assuming a monthly GCR dose of 21 mSv, to obtain a total dose of 248 mSv/year, the total monthly dose due to SPE and GCR adds up to 249 mSv. Hence, the 250 mSv per 30 days have been reached. This configuration results in a total annual BFO average dose equivalent of 466 mSv, which is below the 500 mSv/year limit. The Effective dose equivalent stated on row 2 Table 10-1, for the same event shows an annual total Effective dose equivalent of 575, which is below the career effective dose limits for a 1-year mission for NASA astronauts, male above 30 years old or female above 40 years old, and presented in Table 10-2. These limits still results in an average life-time loss of around 15 years for these specific astronaut groups.

Configuration 3:

The recommended radiation shielding should be more in line with what is presented for Configuration 3. Here the ground floor of the habitat has been lowered into the ground, which results in dose estimates of less than ⅓ for the studied SPEs, see Column 1 of Table 10-4 and Table 10-5. For the Carrington event the BFO Average Dose Equivalent goes from 228 mSv to 82 mSV and for the October 89 event the dose reduces from 82 to 27 mSv. In addition to the SPE dose, the BFO Average Dose Equivalent from GCRs have been reduced from 201 to 128 mSv. These results are a strong argument for placing future Moon habitats partially or fully underground.

The whole habitat is also covered with 25 cm of sintered lunar regolith, or 50 cm of loose regolith. Above the habitat is a water tank with 20 cm of water. This additional shielding result in substantial reductions in the estimated doses and the resulting doses are well below the upper limits recommended by both ESA and NASA, see Table 10-1 and Table 10-2. The chosen sintered regolith wall thickness, of 25 cm and area density of 75 g/cm², actually produce slightly larger GRC BFO doses than what a wall of 17 cm and area density of 51 g/cm² would do, due to the increase in production of secondary particles. However, the shielding against SPEs with the thicker wall still results in a lower annual dose. In order to also see substantial reduction in the GCR BFO Average Dose Equivalent even thicker shielding is needed. For example, the BFO Average Dose Equivalent due to GCRs detected in the ground floor are reduced to 76 Sv/year when using a 1.2 m thick sintered lunar regolith wall.

Configuration 3, shown in right panel Figure 10-7, is not recommended for the suggested habitat, but included in the analysis simply to show the advantages of moving the habitat underground and/or using extensive shielding.

	SPE, ground floor (mSv)	GCR, ground floor (mSv/year)	GCR, top floor (mSv/year)	Annual total (mSv)
Configuration 1: No shielding	887 (300)	228	280	1167
Configuration 2: Minimum shielding	323 (117)	203	252	575
Configuration 3: Additional shielding	106 (40)	131	208	314

Table 10-4: The estimates effective dose equivalent for the ground floor and top floor of the habitat, for three different configurations. The SPE Effective dose equivalent for an average SPE is given within brackets in column 1. The annual total effective dose equivalent (Column 4) is the sum of the SPE Effective dose equivalent (Column 1) in the ground floor and the GCR Effective dose equivalent (Column 3) in the top floor

	SPE, ground floor (mSv)	GCR, ground floor (mSv/year)	GCR, top floor (mSv/year)	Annual total (mSv)
Configuration 1: No shielding	720 (241)	225	274	994
Configuration 2: Minimum shielding	228 (82)	201	248	466
Configuration 3: Additional shielding	71 (27)	128	206	277

Table 10-5 The estimates BFO average dose equivalents for the ground floor and top floor of the habitat, for three different configurations. The SPE BFO Average dose equivalent for an average SPE is given within brackets in column 1. The annual total BFO average dose equivalent (Column 4) is the sum of the SPE BFO average dose equivalent (Column 1) in the ground floor and the GCR BFO average dose equivalent (Column 3) in the top floor.

The analysis presented above shows the need for additional radiation shielding for the proposed Moon village habitat, it estimates the minimum radiation shielding needed for one proposed configuration of the habitat, and it shows the reduction in radiation exposure that can be accomplished with additional shielding. It does not provide an optimal radiation shielding configuration for the suggested habitat simply because the habitat design is still in such an early stage that only general recommendations can be made, which is also what has been provided.

10.6.3 Conclusions

The Ray Tracing analysis performed on the 3 configurations clearly shows that additional radiation shielding has to be included in the design of the habitat in order to provide a safe habitat for the crew. The recommended radiation shielding should be in line with what is presented for Configuration 3, in which the ground floor of the habitat has been lowered into the ground and lunar regolith and water are used for additional shielding. The suggested shielding results in substantial reduction in the estimated doses and the resulting doses are well below the upper limits recommended by both ESA and NASA, see Table 10-1 and Table 10-2.

The uncertainties on the minimum required shielding thickness is very high and it strongly depends on the SPE selection. In this study two conservative solar events (Carrington and Sum of October 89) described in 10.5 have been adopted. Other results and estimates are available in literature: e.g. in RD[57], the author suggest to use 75 g/cm^2 as a minimum thickness for shielding the lunar habitats. This value was obtained using as input the February 56 event (LaRC model).

It is important to remark that the dose exposure shall always follow the ALARA (As Low As Reasonably Achievable) principle. Other considerations and recommendations are the following:

- New dose limits have to be agreed between the agencies for BLEO missions
- Storm shelter shall be available at any time (including installation phase, EVA) and this could require the use of special rovers and portable shielding.
- Preferable low Z materials have to be used for the radiation shielding to improve the shielding efficiency against Galactic Cosmic Rays.
- During SPE, the crew will spend whole days in the radiation shelter and therefore it should be equipped with beds, essential instrumentation and food storage. The recommended option is to shield the crew quarter.
- SPE warning system is needed to avoid the risk of EVAs or surface operation during solar storm. Astronauts shall be equipped with personal dosimeters to optimise the crew rotation when the dose limits are reached.
- If installation of the habitat is not automated and the crew have to oversee the process, the mission should be performed during solar minimum due to the decreased risk of SPEs. Even though the GCR fluxes are larger during solar minimum the risk of encountering an SPE before the shelter is fully installed could compromise the survival of the astronauts. The SPE encounter risk should therefore be kept to a minimum by executing the mission during solar minimum.

11 LIFE SUPPORT

11.1 Challenges and Needs Within the Concept of Operations

High level challenges have to be highlighted, which are leading to specific and highly demanding needs.

The human mission scenario as it is defined, i.e. 300-500 days mission duration (500 days assumed as sizing case for Life Support subsystem and consumables), with only 1 re-supply from Earth per year, for a crew of 4, as a first step of a Moon settlement, will require full redundancy of Life Support Systems. This primary need will have a significant impact on mission requirements such as mass, power, etc.

The purpose of the foreseen mission has no concrete precedence in the space exploration history: building/deploying habitats to pave the way of a Moon Village, in an extremely harsh lunar environment. This will require an additional in depth analysis for the establishment of detailed requirements regarding living scenarios. From there, a step-wise Life Support deployment and implementation strategy, while ensuring survival and safety of the crew at any time, shall be designed. One could envisage though that deployment will start from all ventilation and atmosphere management tolls, then progressively all regenerative processes.

The proposed habitat volume is high; its architecture is conceptually very different from current Space Habitat, making difficult a “simple” extrapolation from e.g. existing systems, piping and instrumentation layouts, type of racks. To ease maintenance and trouble-shooting, inserting distribution/collection lines, storage and collection tanks, monitoring instrumentation, in the walls and in the floors should be studied. Radiation storm shielding would benefit from locating Life Support Systems on the ground floor, which would result in savings on systems mass, e.g. on the avionics subsystems.

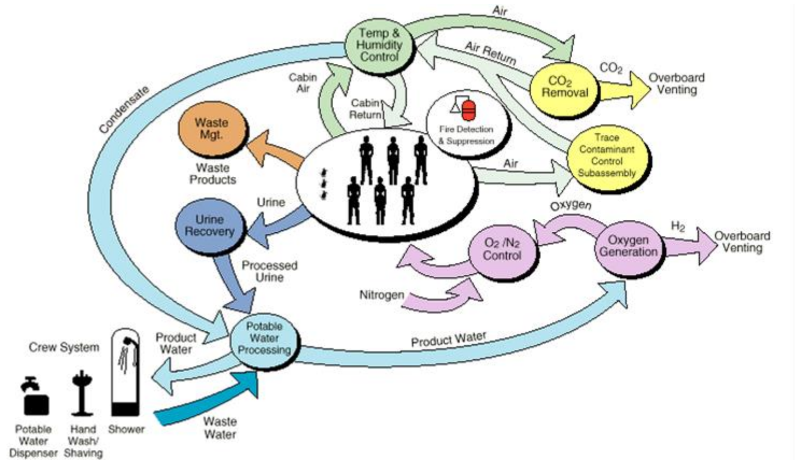
Some technologies considered in this study, or to be considered in the future for a Moon Habitat, today lack maturity (e.g. food preparation, greenhouses, ISRU, ..) to enable their accurate quantification (e.g. sizing, mass, power, crew time,…).

As a general rule, depending on their function and the technology implemented, Life Support systems may be transported full (e.g. supply function), semi-dried (e.g. to prevent from known technical issues at start), or conditioned with a dew point of 0°C. To avoid condensation/freezing at any time, the systems should be kept at a temperature of around 4°C in any point of the systems, which necessitates a transfer temperature of potentially 9 or 10°C.

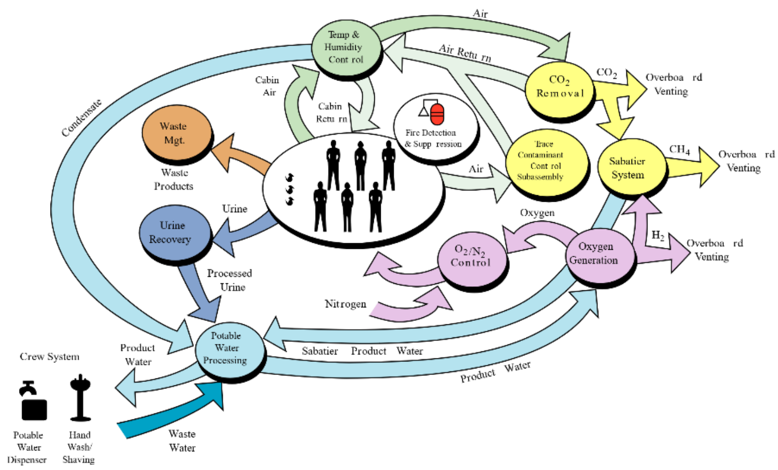
11.2 Baseline Design

Figure 11-1 illustrates various Life Support Systems architectures, with different levels of loop closure.

Current Life Support baseline on-board ISS, for a large part in open loop



With an additional level of resources regeneration



The closed loop regenerative approach of the MELiSSA project (courtesy of the MELiSSA Foundation)

Figure 11-1: various Life Support Systems architectures

In view of the mission scope and duration, combined with the size of the crew and the resupply from Earth limited to one iteration per year, a few key drivers have been identified to design a baseline solution:

- Regenerative closed loop systems for air and water are recommended, with as high as possible recovery efficiencies, to reduce supply from Earth;
- A first step towards on-site food production is highly desirable, i.e. production limited to up to 5% of the daily diet, to prepare for future bigger crew sizes, when supply–from-Earth strategy will become economically unsustainable;
- On-site storage of wastes, preferably outside the habitat, is proposed at this stage; recycling of wastes would become attractive when food production would become fully operational and therefore resulting in the generation of significant mass of inedible biomass.
- Full redundancy (i.e. based on different technologies) seems mandatory in the current context, to address all kind of emergency situations with the appropriate safety level.

At a later stage, implementation of greenhouses and use of in-situ resources will address the need to reduce on the one hand, re-supply from earth and, on the other hand, quantities of ultimate wastes generated.

Additional aspects, such as radiation protection, emergency situations, maintenance and trouble-shooting will have to be taken into consideration for the final accommodation of systems, tanks, etc., in the habitat, per floor (depending of their function and usage), potentially in the walls, or even in the floors.

In line with the key drivers and additional remarks made previously, and from the list of currently available or in development options, the following core technologies are proposed:

- For air revitalisation: ESA's Advanced Closed Loop System (see http://www.esa.int/Science_Exploration/Human_and_Robotic_Exploration/Research/Advanced_Closed_Loop_Systemoption Main technologies and Life Support System Rack picture below), implemented cold redundant with a MELiSSA Compartment 4a based photobioreactor (see RD[58] and ARTEMISS picture below), colonised by *Limnospira Indica*, an edible microorganism commercialised on Earth as food supplement under the name "Spirulina";
- By operating a photobioreactor such as mentioned here above, up to 5% of food can be produced;
- For water recycling: the currently on-board ISS Water Recycling System (WRS), implemented cold redundant with MELISSA Compartment 3 based Urine Treatment Unit (see https://www.melissafoundation.org/page/nitrification) coupled with ESA Grey Water Treatment Unit RD[59].

Figure 11-2: Life Support System Rack (or Advanced Closed Loop System) ©ESA/NASA

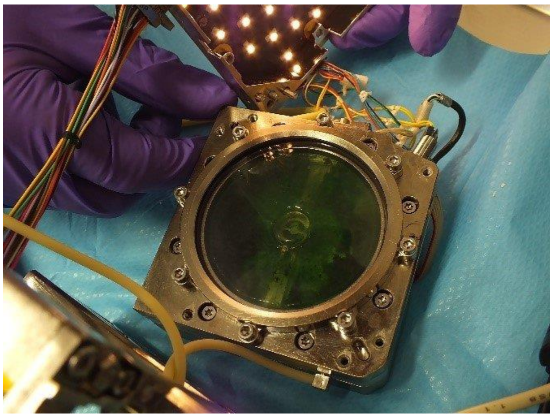


Figure 11-3: Open view of ArtEMISS flight hardware, courtesy of the MELiSSA Foundation

Figure 11-4: The ISS Water Recycling System, including the Urine Processing Assembly, the Water Processing Assembly and the Potable Water Dispenser

Besides these core technologies, additional systems, multiple interfaces and ancillary equipment will be necessary:

- For atmosphere monitoring and control:
  - Ventilation
  - Temperature, humidity and pressure control
  - Gas trace contaminants monitoring (e.g. ESA ANITA RD[60])
  - Microbial contamination monitoring
- For food production and preparation:
  - A biomass harvesting unit
  - A food processing unit
- For waste collection and handling:
  - A Space toilet
  - A waste compaction/inertion unit (e.g. NASA Heat Melt Compactor RD[61])
- Storage tanks for water and gases (oxygen, nitrogen)
- All necessary piping and instrumentation.

11.3 Budgets

Any preliminary budget shall start from the correct dimensioning of operational parameters.

Regarding the **need of crew consumables**, their quantification should be tailored to the mission scenario.

Water, among the various metabolic consumables needed by a human, is by far the heaviest part of the daily bill: about 3 kg/d.CM as a minimum, and one may expect higher demand linked to hygiene – and household- related uses RD[62]. Oxygen and food follow, representing each about 1kg/d.CM.

Waste waters are mainly composed of habitat condensates, water used for hygienic and/or household purposes, all together the so-called “grey waters”, and urine, also named “yellow waters”.

Oxygen is mainly wasted in the form of Carbon Dioxide, which level in the habitat atmosphere requires to be controlled under appropriate thresholds.

Values are presented in Table 11-1.

		consumables
Consumables	**Description**	**kg/CM.d**
Water	Potable water: drinking water and water for food hydration	3.8
	Hygienic water: urinal flush, personal hygiene, shower, laundry, dish-washing	15
	Medical water	0.5

Consumables	Description	consumables kg/CM.d
Oxygen		0.82
Dry food		0.6
Dry food packaging		0.3
Other	cleaning wipes for personal hygiene, household wipes, disinfection wipes...	0.2

Table 11-1: quantities of consumables needed per day and per crew member

Water and oxygen needs can be fulfilled, for the major part, by the regenerative Life Support Systems :

- 95% recovery for water (from the collection and recycling of urine, habitat condensates, hygienic and medical waste waters), meaning 5% has to be re-supplied
- 99% recovery for oxygen (from the collection and processing of carbon dioxide), meaning 1% has to be re-supplied.

In the proposed baseline, 5% of the food (in mass) will be produced on the Moon and therefore does not need to be supplied from Earth. The corresponding dry food packaging will be avoided.

Following assumptions are made for the habitat and habitat atmosphere:

- Nominal pressure: 96-103 kPa
- Oxygen partial pressure control range: 19.5-21.3 kPa
- Total controlled pressure range: 65-103 kPa
- Maximum oxygen concentration: 30%
- Nominal temperature: 21- 22°C
- Pressurised volume: 700 m^3
- Volumetric mass (air, 22°C): 1.2 kg/m^3
- Air velocity: the baseline suggested is the one of the Deep Space Gateway iHAB module, with 66,7% of air velocities between 4.57 and 36 m/min
- Atmosphere leak rate: taking as a baseline 10 times the leak rate of the Deep Space Gateway iHAB module (roughly 0.02%vol/d for iHAB), due to the overall design of the habitat (hatches, windows, inflatable structure,..), roughly 1.4 m^3/d, i.e. 1.7 kg/d of air will be lost.
- Microbial contamination levels: the baseline suggested is the one of the ISS, as presented in Table 11-2.

	Maximum for Bacteria	Maximum for Fungi
Air	1,000 CFU/m^3	100 CFU/m^3
Internal surfaces	10,000 CFU/ 100 cm^2	100 CFU/100 cm^2

Table 11-2: Microbial contamination levels (from ISS)

Table 11-3 presents the quantities to be supplied from Earth, to cover all consumables which cannot be recovered thanks to the regenerative Life Support Systems, or produced on-site.

Wastes or losses to be compensated by ISRU or supply, or weight coming with new supply (packaging, other)			
Crew Consumables			
	For 1d , per CM		For 500 d and 4 CM
	calculation	kg/CM.d	kg
Water	5% of 19.3 kg/CM.d	0.965	1930
Oxygen	1% of 0.82 kg/CM.d	0.0082	16.4
Dry food	95% of 0.6 kg/CM.d	0.57	1140
Dry food packaging	95% of 0.3 kg/CM.d	0.285	570
Other	100% of 0.2 kg/CM.d	0.2	400
Compensation of habitat atmosphere leak			
	Calculation based on	kg/d	kg
Oxygen	20% of 1.4 m^3/d@ 22°C	0.375	187.5
Nitrogen	80% of 1.4 m^3/d @22°C	1.325	662.5

Table 11-3: Consumables required from Earth

Overall budgets for all consumables and equipment are established respectively in Table 11-4 for equipment, and in Table 11-5 for all consumables, based on the currently available level of knowledge. Mass margins are given, tailored to the TRL level of equipment or as best estimation for consumables.

Description	Raw mass (kg)	Mass margin (%)	Total mass (kg)
Initial mass process water	320	10	352
Inoculum for biological processes	20	10	22
Packaged dry food	1,710	5	1,796
Other supplies	400	20	480
Total water to be supplied	1,930	5	2,027
Oxygen	204	20	245
Nitrogen	663	20	795
		TOTAL	**5,716**

Table 11-4: Overall mass budget for consumables

Description	Number of items	Dry mass per item (kg)	Mass margin (%)	Total mass (kg)
Gas tank (132L, 200 bars, 35kg metal for 35kg gas)	30	35	5	1,103
Per gas tank, piping and instrumentation (30kg)	30	30	20	1,080
Water tank (300L, 28.5 kg material for 280kg water) and	7	28.5	5	209
Per water tank, piping and instrumentation (30kg)	7	30	20	252
ACLS for 4 CM	1	850	10	935
MELiSSA C4a compartment (photobioreactor) for 4 CM	1	1,300	20	1,560
Urine Treatment Unit for 4CM	1	250	20	300
Grey Water Treatment Unit for 4 CM	1	600	20	720
WRS (UPA+WPA) for 4 CM	1	1,383	10	1,521
Biomass Harvesting	2(1)	100	20	240
Food processing unit	2(1)	50	20	120
Waste compaction/inertion	2(1)	50	20	120
Space toilet	2(1)	50	10	110
Gas trace contaminants monitoring	2(1)	30	10	66
Microbial contamination monitoring	2(1)	30	20	72
Temperature and humidity control	6(2)	230	20	1,656
All interfaces	1(3)	400	30	520
(1)2 redundant units (2)6 subsystems distributed over the habitat (3)bulk estimation **TOTAL**				**10,584**

Table 11-5: overall mass budget for equipment

From this very preliminary assessment study, it is established that approximately **11 t of equipment** and **5.7 t of consumables** would have to be shipped from Earth to allow for safe living of the 4 member crew over 500d. The corresponding bulk power budget is estimated around **40 kW**.

Several limitations have to be reported regarding the figures established:

- The strategy proposed for Life Support is going much beyond what is currently in operation on-board ISS, resulting in uncertainties on systems mass and on all the interfaces needed;

- Recycling efficiencies of processes have a significant impact on the consumable budget: as an example, increasing the recovery of water from 95 to 98%, would result in the reduction of supplied water from 1,930 kg down to 772 kg;
- The overall leak rate of the habitat is a key issue, however rather difficult to estimate, due to the unknowns on the habitat structure and materials;
- Additional systems for e.g. EVAs, fire suppression, have not been discussed yet but should be included at a later stage, when operations will be defined in more details;
- Spare parts strategy has not been discussed at this stage (neither for Life Support Systems nor for the whole settlement), however impact on the mass budget can be extremely significant RD[63].

11.4 Options

N/A

11.5 Technology Needs

All the technologies identified in this study are enablers for the feasibility of this and/or other future ESA missions, when involving mid to long-term permanent presence of humans. Apart from the technologies already in use on-board ISS or in other space-crafts (e.g. tanks used on-board ATV), they all need to undergo development phases to reach the appropriate level of maturity.

ESA roadmaps (e.g. ESA Harmonisation roadmaps, ESA Space Resources Strategy) and ESA Working Groups (e.g. Life Support Working Group) are tentatively proposing development strategies for these technologies. Several ESA Member States, European Large Space Integrators, several European industries, universities, research organisations are contributing to these developments.

11.6 Ideas For Patents

N/A

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12 MATERIALS AND PROCESSES

Human rated inflatable modules are a transformative capability, which have successfully proven over the past number of years as habitable space environments. These lightweight structures allow for the capability of launching much larger habitable volumes into space than is possible with rigid shell structures. With a history of development spanning over five decades these structures include space stations, habitats, airlocks and deployable tunnels for missions, both in space and on planetary surfaces RD[71]. Worldwide the leader in space inflatable modules is the US with several developments along the decades where the latest program –TransHAB- ultimately lead to an in-flight validated module produced by BIGELOW Aerospace (US) RD[64].

BIGELOW Aerospace has in fact launched an expandable module named BEAM (Bigelow Expandable Activity Module), the inflatable module has been deployed on the ISS since 2016 and following its successful operation BEAM has received authorization to be life extended and used as storage on the ISS until the late 2020s RD[69].

The module is flight proven in LEO- ISS environment, however a transfer to the Moon environment may impose some materials adaptations due to higher levels of radiation or new challenges such as moon dust presence. Furthermore, CONOPS will be different from ISS and may impose technology development on inflatables as well. For obvious heritage reasons BEAM is the baseline for the materials selection on this Moon Village CDF study.

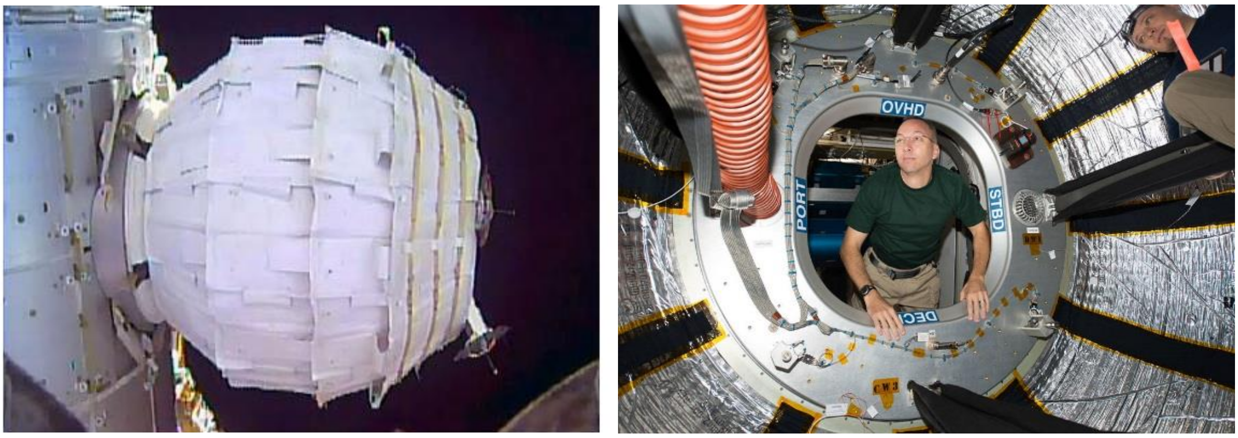

Figure 12-1: BIGELOW expandable mode BEAM attached to the ISS from outside (left) and picture from the inside (right) RD[67]

In Europe, Thales Alenia Space Italy (TASI) is highly experienced in designing and manufacturing habitable modules for human spaceflight. The first ones have flown with the Space Shuttle such as SpaceLab and SpaceHab. In LEO especially under ISS, TASI has contributed actively and has delivered multiple modules (Node2&3, Cupola, Columbus, ATV, PMM, MPLM, Cygnus etc.). More recently TASI is one of the main suppliers of the European Service Module (ESM) for ORION.

Over the years and relying on its experience in human rated structures, TASI has developed inflatable habitable technologies across several projects. European and

Italian funding have paved the current European experience on expandable human rated structures- examples are shown below in Figure 12-2 (see also RD[65] for further information).

Figure 12-2: Inflatable module demonstrator (left); inflatable docking mechanism (upper right); flexible expandable module with foldable micrometeoroid and debris protection (MMOD, lower right)[RD[65]

Due to the very large surface area of the inflatable structure, issues regarding packing efficiency, foldable designs and structural containment will be important factors for pre-launch activities. Figure 12-3 gives an overview of previously developed foldable structures.

Figure 12-3: Full size crew lock foldable structure for ISS, with high packing efficiency and embedded sensor network RD[65]

In terms of materials technology this Moon Village study took into account some previous experiences, and highlights potential fields where further materials related development is recommended in order to achieve a safe and functional Moon Village.

The next paragraphs will focus on these proposals which will be mainly linked to the inflatable section since it is considered more challenging or simply newer than the quite consolidated (demonstrated and routine technology) metallic or composite materials. Nevertheless, rigid composites structures do have a limitation of being porous thus not hermetically sealing and specific R&D is therefore needed to produce a gas impermeable composite or to develop multilayer barriers for CFRPs- in addition the issue of rigid / inflatable interfaces will be a challenge which needs to be address through proper design and materials choice.

12.1 Challenges and Needs within the Concept of Operations

From a materials and processes perspective, a number of challenges have been identified, during the testing phases, launch, inflation and the unoccupied phases of this mission.

During the testing phases, system safety verifications including off-gassing testing is needed at system level to certify low off-gassing (toxicity) values from materials. In addition, in order to qualify an inflatable, sector level leak tests should be possible, other than testing the fully integrated system with the rigid structure. The issue of rigid structure/ inflatable interface testing has also been addressed as a major issue during materials selection, processing and testing- namely independent localised leak testing at these interfaces, mechanical and structural integrity testing.

In addition, while limited guidance on qualifying and verifying the long term durability and design performance of these inflatable structures was previously limited to using factors of safety (FOS- usually 4), an updated approach now take materials variability (e.g. due to nonlinear behavior and interactions of soft goods), effects of long term degradation and damage into account to present a more realistic safety factor. These aspects should be adapted, applied and factored in during any new materials design approach RD[71].

Challenges highlighted during this phases of operations include among others- the control or mitigation against flammability (due to high amounts of internally exposed polymers), which can be a high concern from a safety perspective. From a manufacturing and processing perspective (e.g. the processing of the MMOD fabric layer) – this may include the use of fabric, knots or looped materials with variations in homogeneity, density and structure which can result in mechanical and structural variations. In addition, these foldable joints also do not allow traditional non-destructive inspection (acceptance testing is not like metal/CFRP structures) - hence qualification at materials level and verification of the processing will be an issue. Regarding the transportation to launch site and prelaunch activities, the issue of the packing of this foldable structure remains a challenge both in terms of the design (large area is to be used), multilayer materials composition and delicacy of layers (e.g. fragile MLI) and integrity during the folding and containment to maintain structural integrity.

During the launch phase the assumption for materials is that the habitable module will be in vacuum during launch and cruising, thus the design shall also use vacuum compatible materials. Furthermore the non-operational temperature range will be minus 100C to plus 100C. This temperature range is not expected to pose specific challenges however it remains a matter of concern for single materials, its joints and its performance since the films and fibres will be at glassy state thus not in its operational range. Also of concern is the launch and venting design for MLI and various outer layers. In addition, given that internal airtight layers will also be used, e.g. the air containment bladder will inflate during launch unless specifically designed not to do so.

Following landing, deployment and transfer of the lunar habitat, it has been envisaged that during the inflation process, inflatable materials at cryogenic temperatures (especially the air containment bladder) may not be foldable or flexible compared to nominal conditions, therefore before inflation these layered materials will potentially need to be heated in order to avoid thermal stresses of various materials and configurations. During the usage of the habitable structure, materials and components of the layered structure will potentially be damaged, degrade or can fail in use (e.g. holes, impacts, leaks etc.)- hence it is critical from a safety and structural perspective that the majority of materials uses can be repairable in-situ.

Finally during the unoccupied phase to reduce prolonged stresses on the layered structures and reduce the flammability risk a lower pressure environment can be an option.

12.2 Baseline Design

The baseline materials selection has been chosen as a compromise between functionality, safety, performance heritage and design and environmental criteria. This has resulted in an intricate interlayer of high performance materials where the materials compositional lay-up is split between two groups;- group 1 materials: the external or space / vacuum exposed materials (where issues of vacuum comparability, dust contamination, micro-meteoroids, outgassing, cosmic and ionising and non ionising radiation are predominant) and group 2 materials: the internal pressurised side – inflatable section (inner layers towards the crew habitat)- where issues of crew safety , flammability off-gassing , toxicity and permeability are pre-dominant.

These are detailed in RD[70] where layer numbers, density and materials compositional information can be found and below in Figure 12-4 and Figure 12-5 below.

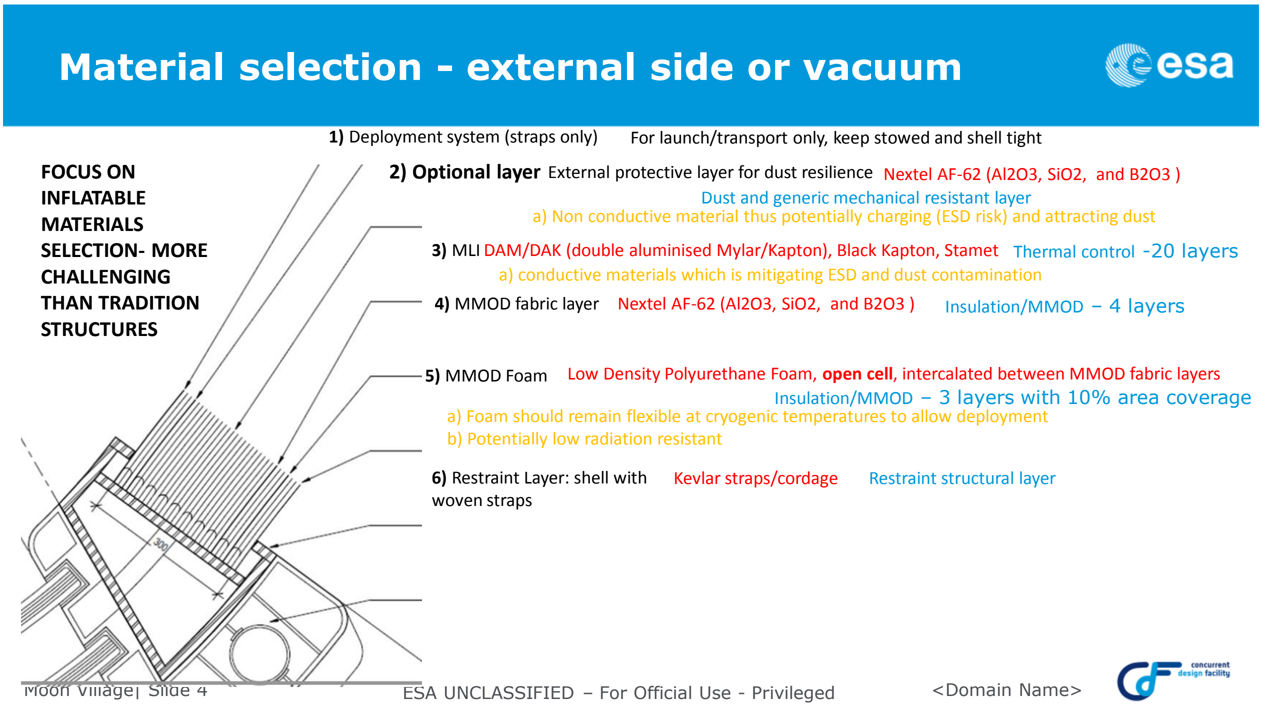

Figure 12-4: Group 1 materials- the external or space / vacuum exposed materials

Group 1: materials- external or space / vacuum exposed materials (Figure 12-4**)**

- Layer 0 – (not in Figure 12-4) -Starting from the outside on the external side – the first (optional) layer of sintered/loose regolith is suggested in order to meet safety radiation habitable limits as detailed in section 12.3 below. This regolith will provide needed radiation protection to crew and reduce extreme radiation exposure to the inflatable structure. Here new in-situ technology is assumed.
- Layer 1 – The first external materials layer is the deployable system (these are external straps used in the containment of the inflatable system during launch / transportation and are kept stowed and shell tight.
- Layer 2 - Moving further inwards in the external layered materials the next (optional) external protective layer is chosen for external protection layer for dust resilience and generic mechanical resilience. Nextel AF-62 is a ceramic yarn combination of Al_2O_3, SiO_2 and B_2O_3. Potential issues could be the thermo-optical degradation due to potential charging effects of the non-conductive surface therefore attracting dust.
- Layer 3 - The addition of an MLI layer was chosen for thermal control. The usual multilayer (20 layer) combination of DAM/DAK (double aluminised Mylar/Kapton), Black Kapton or Stamet is used for also since its conductive surface can mitigate and reduce dust contamination and build up effects. However, it has little protection from MMOD.
- Layer 4 - For this MMOD layer again Nextel AF-62 is chosen (based upon previous heritage. Again an Al_2O_3, SiO_2 and B_2O_3combination in the form of a ceramic yarn is chosen. There are no major issues foreseen in this layer.

- Layer 5 - This is the MMOD foam support used between the MMOD layers. These foam blocks are usually light weight polyurethane foam open cell structures. Potential issues here include polymeric flexibility at lower cryo temperatures and also the issue of radiation protection. For this design, 3 layers with a 10% area coverage is suggested.
- Layer 6 - The next layer moving inwards is the restraint Kevlar layer which is used for structural restraint. Here high performance fabrics such as Kevlar are suggested. Here 2 layers of this interwoven fabric material is suggested. Issues may include potential creep issues after longer time periods due to prolonged tensile stresses on the yarns and fabrics (hence leading to localised protrusion of internal materials) or lack of available adequate testing and quality to ensure homogeneity of the fabric straps and cordage. This layer is also sensitive to UV.

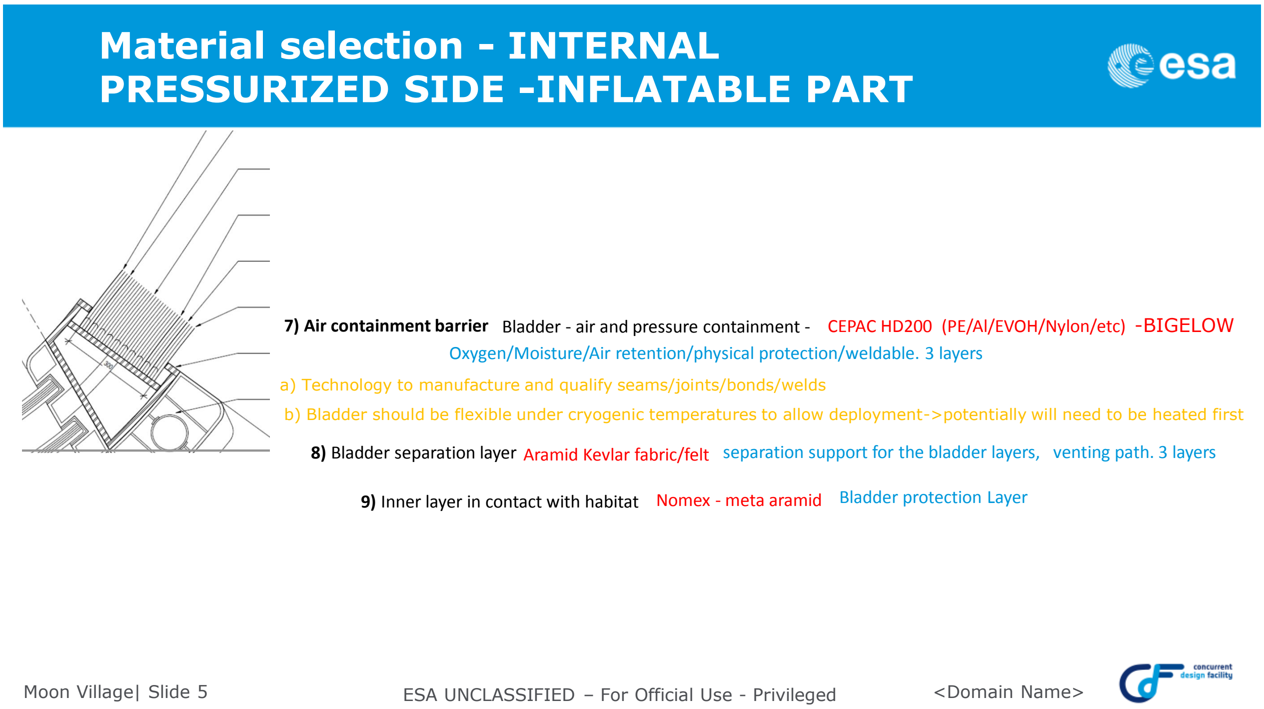

Figure 12-5: Group 2 materials) the internal pressurized side – inflatable section (inner layers towards the crew habitat)

Group 2 materials: the internal pressurized side – inflatable section (inner layers towards the crew habitat) Figure 12-5

- Layer 7 - This first inner inside the restraint layer is the soft bladder balloon like material. This layer is used for air containment within the crew habitable zone. From heritage (Transhab) a complex multilayer material combination is used consisting of CombiTherm/ Silicone /Polyurethane. 3 layer barrier film and separation could be used. However this had been updated in recent times following the on-board success of the Bigelow. Here a combination of CEPAC HD200 was used as an air , oxygen , moisture and pressure containment barrier layer. This is a multilayer (PE/Al/ EVOH/ Nylon) material however poses many challenges including flammability resistance, technology maturity, flexibility at low temperatures and easily repairable.

- Layer 8 - Bladder separation layer- This layer is composed of Aramid Kevlar fabric and used as a bladder separation layer in order to avoid friction between the bladder balloon like layers. It also acts as a venting path. Again since this is internal to the habitable crew zone the issue of flammability mitigation needs to be maintained.
- Layer 9 - As a final inner bladder protection layer –nomex aramid fabric has been chosen to protect these inner layer again inadvertent contact and provide protection.
- Layer 10- (optional) - internal water layer for increased radiation protection. If this was to be used then obvious issues of containment and crew safety needs to be met.

Note: Full materials details (including proposed window materials) giving composition, materials design descriptions, materials processing issues, layer thickness and density as well as other information can be found in RD[70].

12.3 Budgets

Following materials design choices, including materials type, aerial density, number of layers and thicknesses used, the main driver in materials choice was functionality versus weight and radiation protection. Here a balance was required in order to meet the structural demands and the following budgets for materials are presented in Figure 12-6 below.

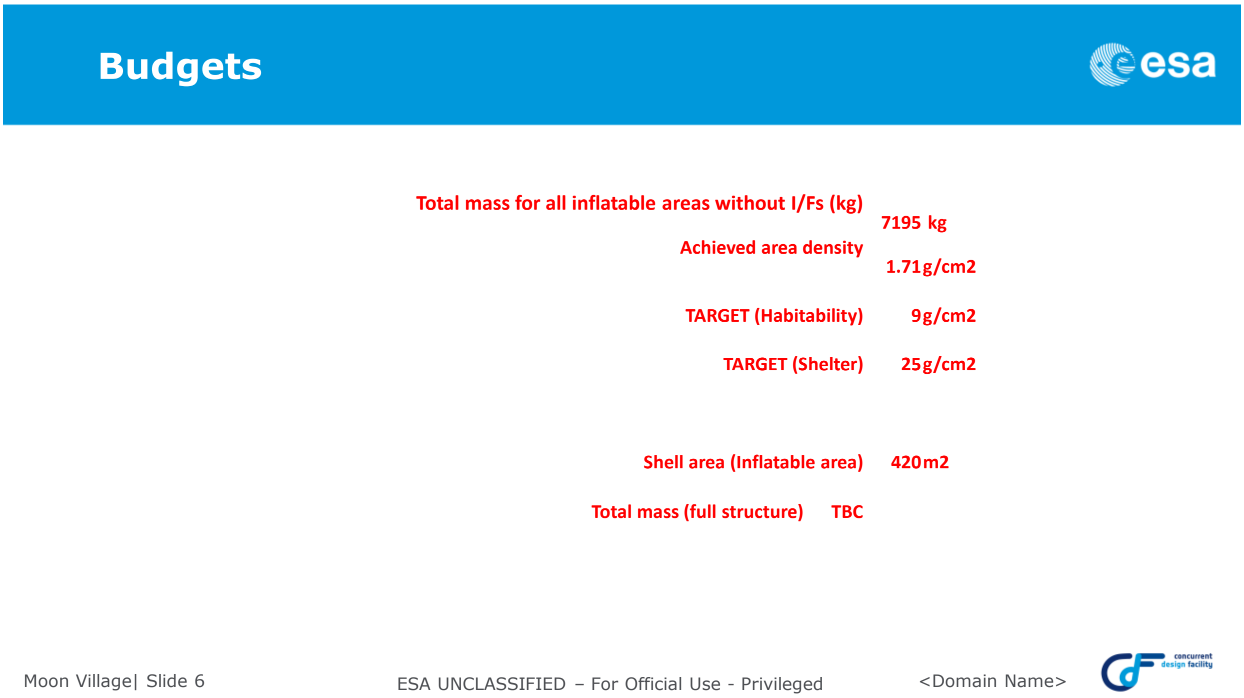

Figure 12-6: Materials Budgets

As can be seen from Figure 12-6 above, the total shell area from structural and design models for the inflatable shell area was chosen as 420m². Based upon this area a total mass of 7195 kg was derived for all internal and external layers (this excludes the external regolith (layer 0) and potential inner water protection layers (layer 10)). For the crew habitation zone, taking worse case radiation scenarios, an aerial density of 9 g/cm² was required to protect against radiation in the habitable zone and 25 g/cm² for the shelter locations. Radiation calculations based on the current design (excluding the external regolith (layer 0) and potential inner water protection layers (layer 10)) reach a value of 1.71 g/cm², which is far below the safety limits and hence the need to design using the additional external regolith or optional inner water layers for increased radiation protection. Designing with increased aerial density or increased layer thickness or number of layers would be problematic both from a materials manufacturing and processing perspective, a design and deployment perspective and a structural and weight consideration during launch and transportation- hence the design options chosen are optimised with this in mind.

12.4 Options

Figure 12-7 below list the additional materials options in terms of additional layers. The main radiation protection regolith layer and inner water layer are also functional options. All other materials presented are in RD[70] and are part of the budget baseline for materials. As an example of alternative materials, for the restraint layer choice, the use of high strength Zylon or Vectran can be used in place of Kevlar. For the bladder material other alternatives including Combitherm can be used which have been previously successfully used in Transhab.

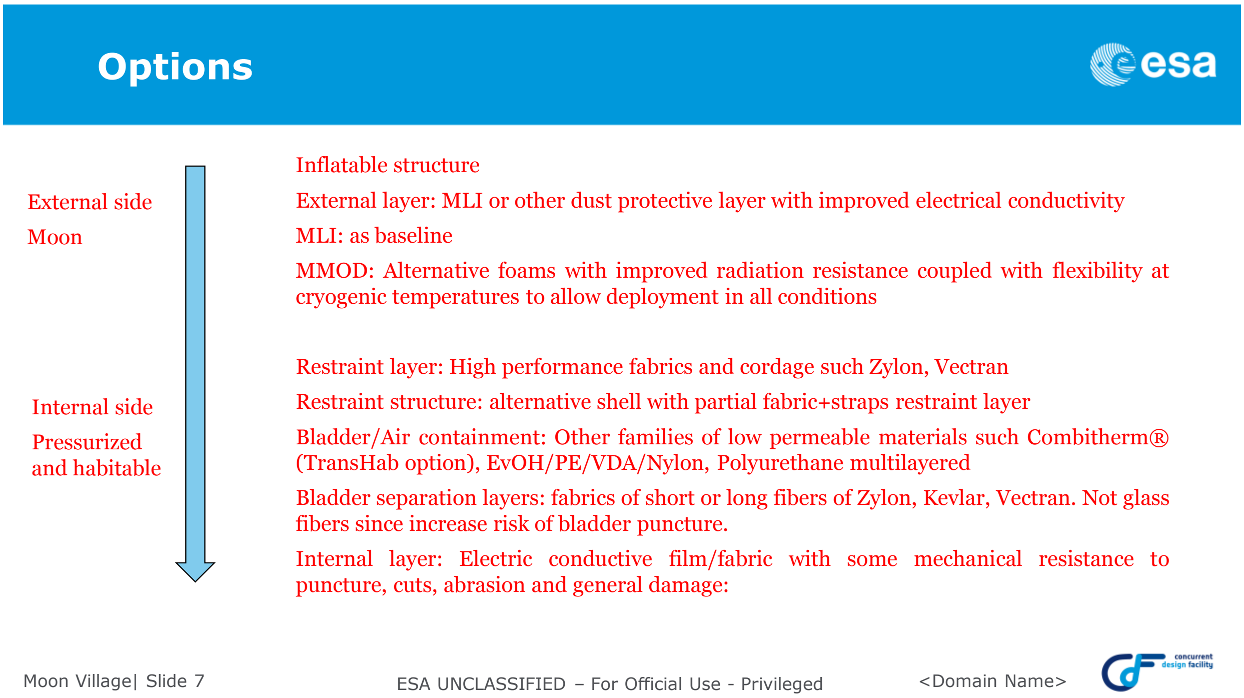

Figure 12-7: Materials options

12.5 Technology Needs

The Moon Village development will require extensive technological achievements in order to meet the safety requirements while coping with the extreme and specific environment present on the Moon as well as the CONOPS constraints. In terms of materials, the inflatable sections have been recognised as the most critical in terms of new technologies, (see Table 12-1 below) while the rigid structures are certainly more mature and thus potentially feasible using metallic light alloys or composite materials. In fact as there is a major difference in terms of heritage between these two categories of structures and materials since human rated and non-human rated satellites spacecraft are being build exploiting rigid structures.

12.5.1 Moon Village Challenges Related To Materials

External radiation shielding using in situ resources has been recognised by this study as a key technology for habitability on the Moon surface. The most available material is Moon regolith, therefore to be exploited in construction as an external radiation shielding wall (see Figure 12-8). Additive Manufacturing technology seems the most promising and marketed solution as being developed at European level.

Figure 12-8: Lunar Habitable Structure constructed in-situ from moon regolith

Habitat external layers (Figure 12-9 -in blue) will be robust to dust contamination, in particular materials and equipment with foldable and moving interfaces since potentially sensitive to abrasion, erosion, change in thermal optical properties. Furthermore external layers will exploit charging mitigation strategies to reduce risk of ESD and dust attraction.

Foldable micrometeoroid and debris protection (Figure 12-9 -light orange) will be implemented in the external side of the foldable structure, it will protect the Moon Village during the CONOPS and early use phases until a final radiation shielding is implemented. The main challenges are the installation phase where foams should expand even at cryogenic temperatures.

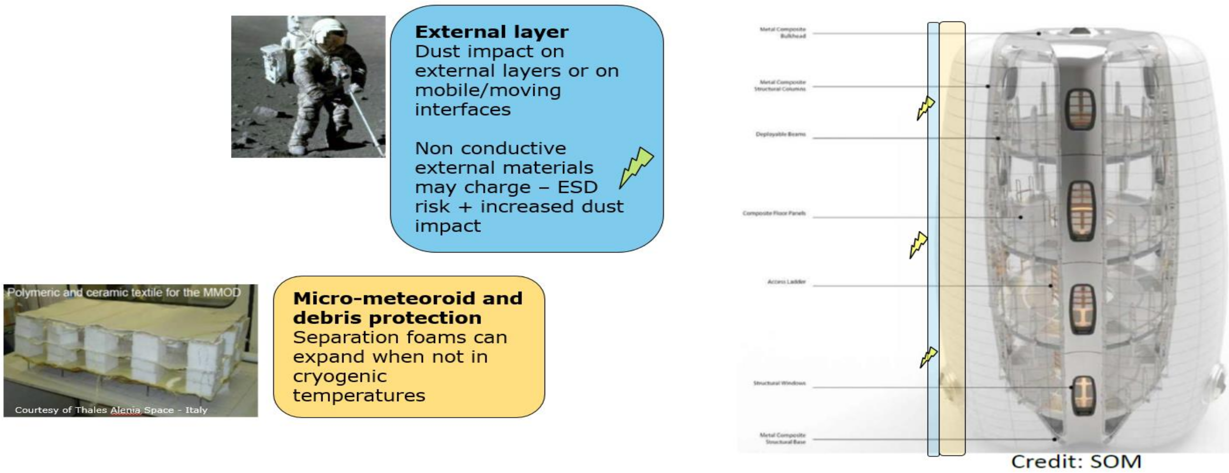

Figure 12-9: External and micro-meteoroid protection layers

The restraint layer (Figure 12-10) is considered a key technology to be mastered from a design, manufacturing and verification standpoint. The challenges cover a spectrum of details such as the complex interface with rigid structures, the lack of NDI to verify fabrics and seams acceptability, geometry and tolerance control, material constraints (viscous flow, UV sensitivity, etc.). At prototype level, TASI has manufactured and verified inflatable restraint modules and crew locks with advance fabric materials.

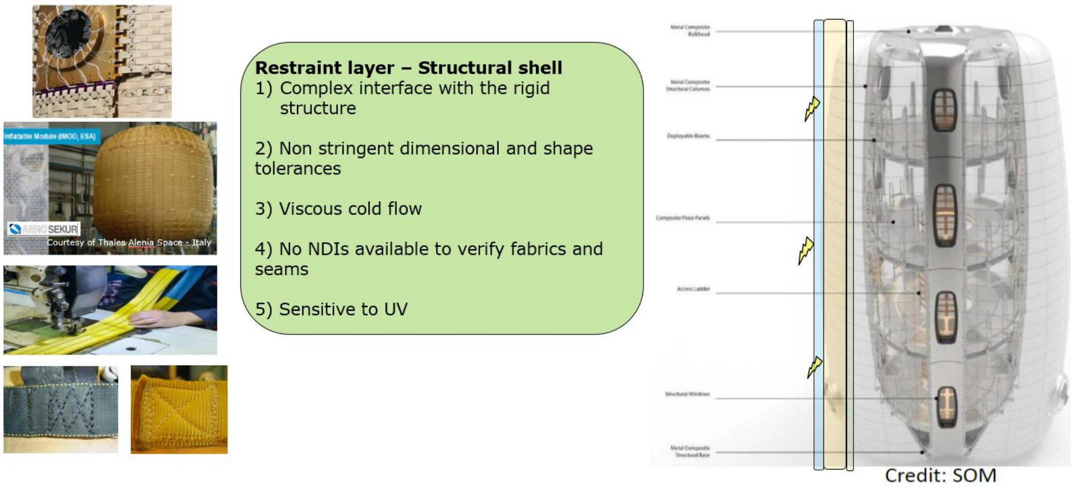

Figure 12-10: Restraint layer

The air containment barrier layers (Figure 12-11) are responsible to keep pressure inside the module, however the secondary functions are several and thus technology to produce lightweight air containments is unique. The multi-layered materials shall act as independent barriers for both air and humidity, maintaining flexibility under low temperatures for inflation while always remaining leak tight. The materials are mechanically weaker compared with the restraint layer resulting in complex interfaces. Safety is also a concern for what regards flammability, high temperature, puncture, biological induced damage and toxicity due to off-gassing.

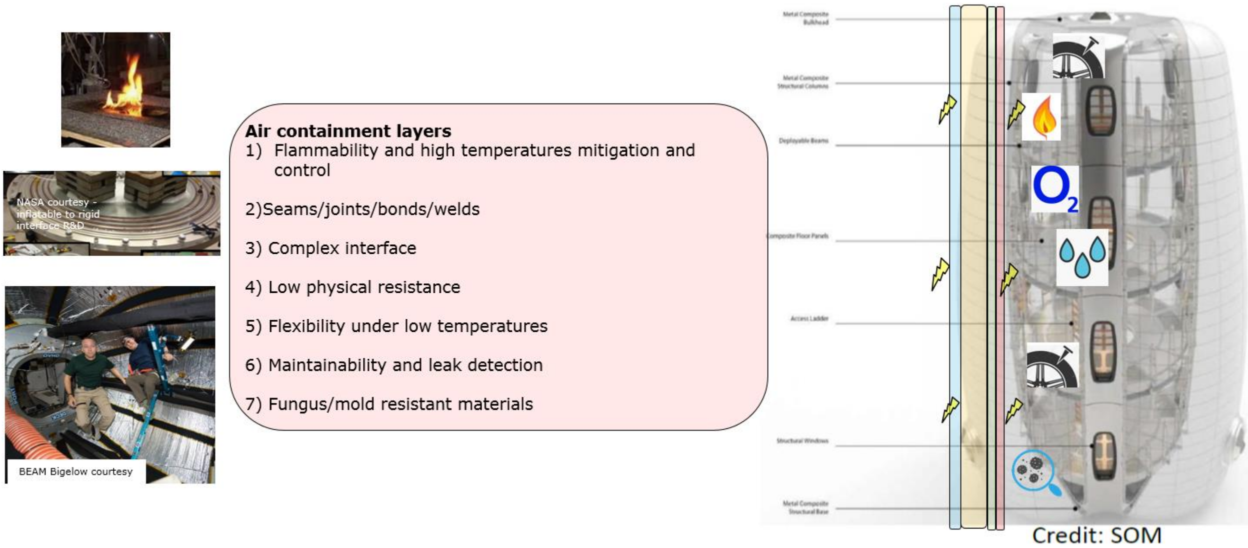

Figure 12-11: Air containment barrier layers

Table 12-1 below presents the technology needs for the moon village- the major technologies highlighted are the inflatable structures where some heritage exists already in terms of the BEAM module, flexible/ rigid materials interfaces where there is some heritage from TASI and other technology issues where development is required including; fire resistant air containment barrier layers, inflatable / foldable MMOD layers , NDI techniques for the flexible shell and the development of in-situ repair technologies for the inflatable sections.

Technology Needs						
*	Equipment Name & Text Reference	Technology	Supplier (Country)	TRL	Funded by	Additional Information
	Inflatable habitable module	inflatable habitable module	Bigelow (USA)	9	USA	BEAM is flight proven in LEO – ISS environment. conversion to moon may require some developments
	Inflatable Shell	Flexible to rigid interface for structural restraint, to air containment layers	TASI (IT)	>=5		
	Inflatable Shell	Large surface air containment bladder bonding joining	TASI (IT)	>=5		
	Inflatable Shell	In Loco repair technologies for inflatable sections				To be developed
	Inflatable Shell	Antibacterial coatings for bladder and fabrics at the habitable sections	TAS (IT)	04-May		

Technology Needs						
*	Equipment Name & Text Reference	Technology	Supplier (Country)	TRL	Funde d by	Additional Information
	Inflatable Shell	Air containment barrier with high flame resistance				To be developed
	Inflatable Shell	CFRP Structures with air and humidity containment properties				To be developed
	Inflatable Shell	Structural flexible shell NDI				To be developed
	Inflatable Shell	Inflatable/foldable MMOD for lunar application	TAS (IT)	04-May		
	Inflatable Shell	Inflation technology to successfully deploy the inflatable sections				To be developed
	Inflatable Shell	Packing technologies to reduce mass and support successful deployment				To be developed
	Regolith radiation shields - in loco resource utilisation	3D printing radiation shields				Several additive manufacturing technologies are under development in EU
	External moon MLI	Dust repellent low alfa, moon compatible MLI material				To be developed

Table 12-1: Moon Village Technology Needs

12.6 Additional Material

Section	Notes	Layers	Material	Function	Nº layers	Thickness (per layer) (cm)	Total thickness (cm)	Aerial density (kg/m2)	Density (kg/m3)	Total mass/sqm (kg/m2) (all layers)	Total mass for all inflatable areas without I/Fs (kg)
EXTERNAL SIDE OR IN VACUUM - INFLATABLE PART	Optional layer	Sintered regolith	Sintered regolith	Delta radiation protection based on external regolith shield	0	2	0	60	3000	0.0	0
		Deployment system (straps only)		For launch/transport only, keep stowed and shell tight	1	0.5	0.5	0	0	0.0	0
	Optional layer	External protective layer for dust resilience	Nextel AF-62 (Al_2O_3, SiO_2, and B_2O_3)	Dust and generic mechanical resistant layer	1	0.14	0.14	0.98	700	0.98	412
		MLI	DAM/DAK (double aluminised Mylar/Kapton), Black Kapton, Stamet	Thermal control	20	0.0025	0.05	0.0355	1420	0.7	298
		MMOD fabric layer	Nextel AF-62 (Al_2O_3, SiO_2, and B_2O_3)	Insulation/MMOD	4	0.14	0.56	0.98	700	3.9	1646
		MMOD Foam (Assumption: 3 layers, each covering 10% of the total deployable shell area)	Low Density Polyurethane Foam, open cell, intercalated between MMOD fabric layers	Insulation/MMOD	0.3	7.6	22.8	3.648	48	1.1	460
		Restraint Layer: shell with woven straps	Kevlar straps/cordage	Restraint structural layer	2	0.5	1	3.7	740	7.4	3108
INTERNAL PRESSURIZED SIDE - INFLATABLE PART, NOMINALLY PRESSURIZED, IN VACUUM	TransHAB optional barrier layer	Bladder - air and pressure containment	CombiTherm/Silicone/Polyurethane (3 layers barrier film + 3 separation layer)		0	0.02	0	0.26	1300	0.0	0
	Bigelow barrier layer	Bladder - air and pressure containment - BIGELOW material	CEPAC HD200 (PE/Al/EVOH/Nylon/etc)	Oxygen/Moisture/Air retention/physical protection/weldable	3	0.0254	0.076	0.33	1300	1.0	416
		Bladder separation layer	Aramid Kevlar fabric/felt	separation support for the bladder layers, venting path.	3	0.07	0.21	0.518	740	1.6	653
		Inner layer in contact with habitat	Nomex - meta aramid	Bladder protection Layer	2	0.03	0.05	0.24	962	0.48	202
	Optional layer	Water	H_2O	Delta radiation protection based on H_2O	0	6	0	60	1000	0.0	0
				TOTALS:	36	17	25	131	11910	17.1	7195

Shell area (Inflatable area)	420.00	m^2
Achieved area density	1.71	g/cm^2
TARGET (Hab)	9	g/cm^2
TARGET (Shelter)	25	g/cm^2
Shell Mass (current)	7194.50	kg

Table 12-2: Deployable Shell details

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13 ADVANCED CONCEPTS AND INFLATABLES

The history of space inflatables reaches back to the mid 1950's when engineers and scientist started to conduct studies on using softgoods in space structures. One of the first concepts of an inflatable space station was developed by Wernher von Braun, which envisioned a rotating toroidal wheel-shaped station in LEO accommodating several hundred crew members (Figure 13-1, left). The design that was presented to the American public in 1951 at the First Symposium on Space Flight unfortunately never became a reality. Nevertheless, the ideas were picked up by NASA researchers in the following years who saw the potential of inflatable technology for structures in space. The first realized application, however, was not in a habitat, but in passive communication satellites – satelloons. The first the Echo satellite (Figure 13-1, right), a 31-meter-diameter balloon, was put into an orbit in 1960. RD[72]

Figure 13-1: W. von Braun space station (left) and Echo satelloon (right) RD[74] RD[75]

The success with Echo satelloons proved to NASA researchers that one of the most promising technologies for an orbiting space station would be a self-deploying inflatable. Since 1960's NASA worked together with Goodyear to conduct the first in depth studies about inflatable habitats for human spaceflight. In addition to the toroidal demonstrator (Figure 13-2, left), they were able to manufacture a number of full-scale prototypes for surface habitats (Figure 13-2, right) and an airlock. These designs allowed to test the necessary hardware for inflatable habitats, addressing various issues related to multi-layered softgoods structure that would be resistant to space environment. Unfortunately, none of these designs were sent into space. At the same time, the Russians were able to manufacture their own inflatable airlock, which was sent to space in 1965. Due to a couple of life threatening events the airlock was later discarded, and remained the only human-rated inflatable sent to space until 2016. RD[73] Throughout the 1970's the focus on inflatable technology at NASA died out, not because of the doubt in technology, but rather due to lack of support. The studies were briefly started up again in 1980's when the US administration talked about returning to the Moon. In 1989 one of the concepts that NASA proposed was the Inflatable Habitat Concept for a Lunar Base. At the same year also the Lawrence Livermore National Laboratory started to investigate the inflatable concepts and develop prototypes, however the activities gradually ended due to the lack of financial and political support. RD[74] RD[75]

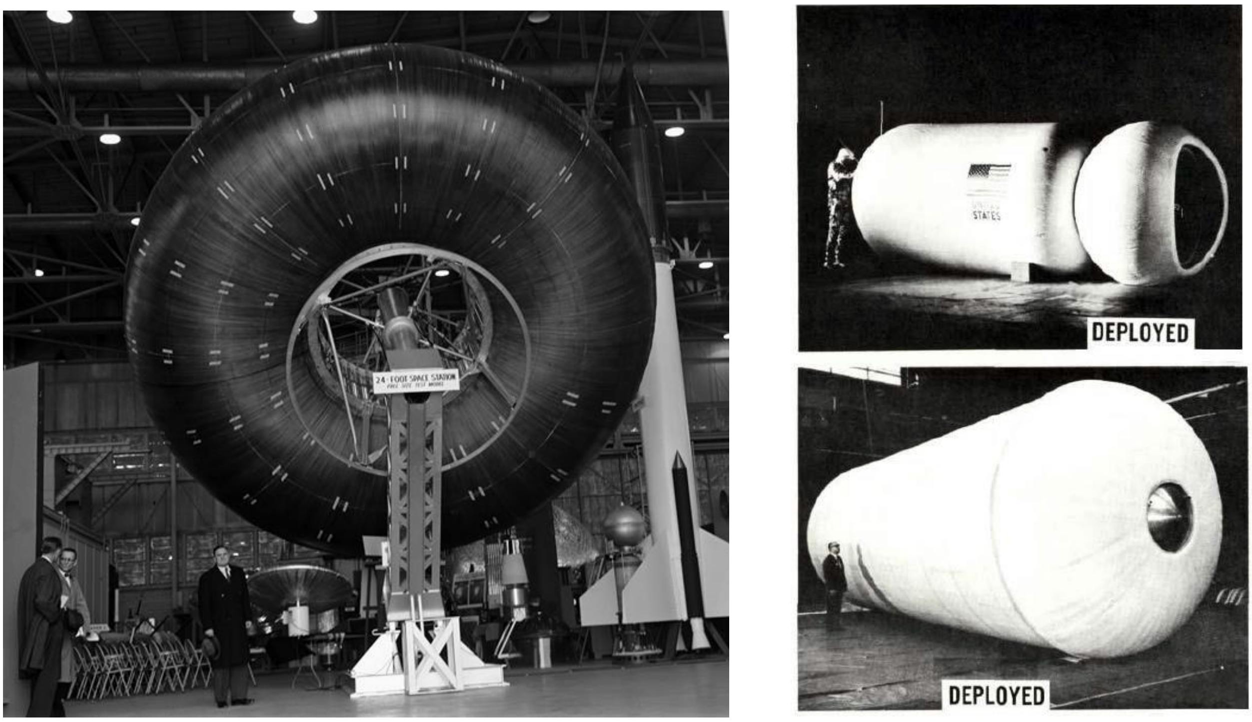

Figure 13-2: NASA/ Goodyear Aerospace inflatable demonstrators RD[75]

In 1996 NASA, together with ILC Dover, began looking into using an inflatable habitat for a Moon mission for the checkout activities before establishment of a permanent habitat. Soon after that, in 1997, the focus was set on the development of an inflatable habitat module, a possible addition to ISS. The inflatable was initially considered as an interplanetary transit habitat (TransHab). TransHab (Figure 13-3) had a rigid aluminium and graphite-composite core which was surrounded by an inflatable shell. The hab was able to provide three times more habitable volume than traditional aluminium modules based on the same mass and costs due to its ability to be folded for launch and deployed in orbit. The full-scale version of this three level lightweight habitat was actually built and tested to four times ambient pressure, for folding/ inflation and micrometeorites protection. It was for the first time that the inflatable passed all the necessary tests successfully proving its suitability for human spaceflight. One of the biggest problems was solved under the direction of William Schneider who helped to develop an adequate MMOD protection for TransHab that had remained one of the weak points of flexible textile based structures until then. Unfortunately, TransHab project was cancelled due to financial concerns for many years until the technology was bought by Bigelow Aerospace. RD[76] RD[77]

Due to many benefits inflatables have over traditional rigid modules, and developments in softgoods technology, the TransHab concept has remained one of the most followed examples in inflatable habitat design until today. To that end, over the years a number of studies have been conducted proposing inflatables as orbital, deep space or planetary habitats. Only a few, however, have been able to develop full scale prototypes since TransHab for testing and development of the technology.

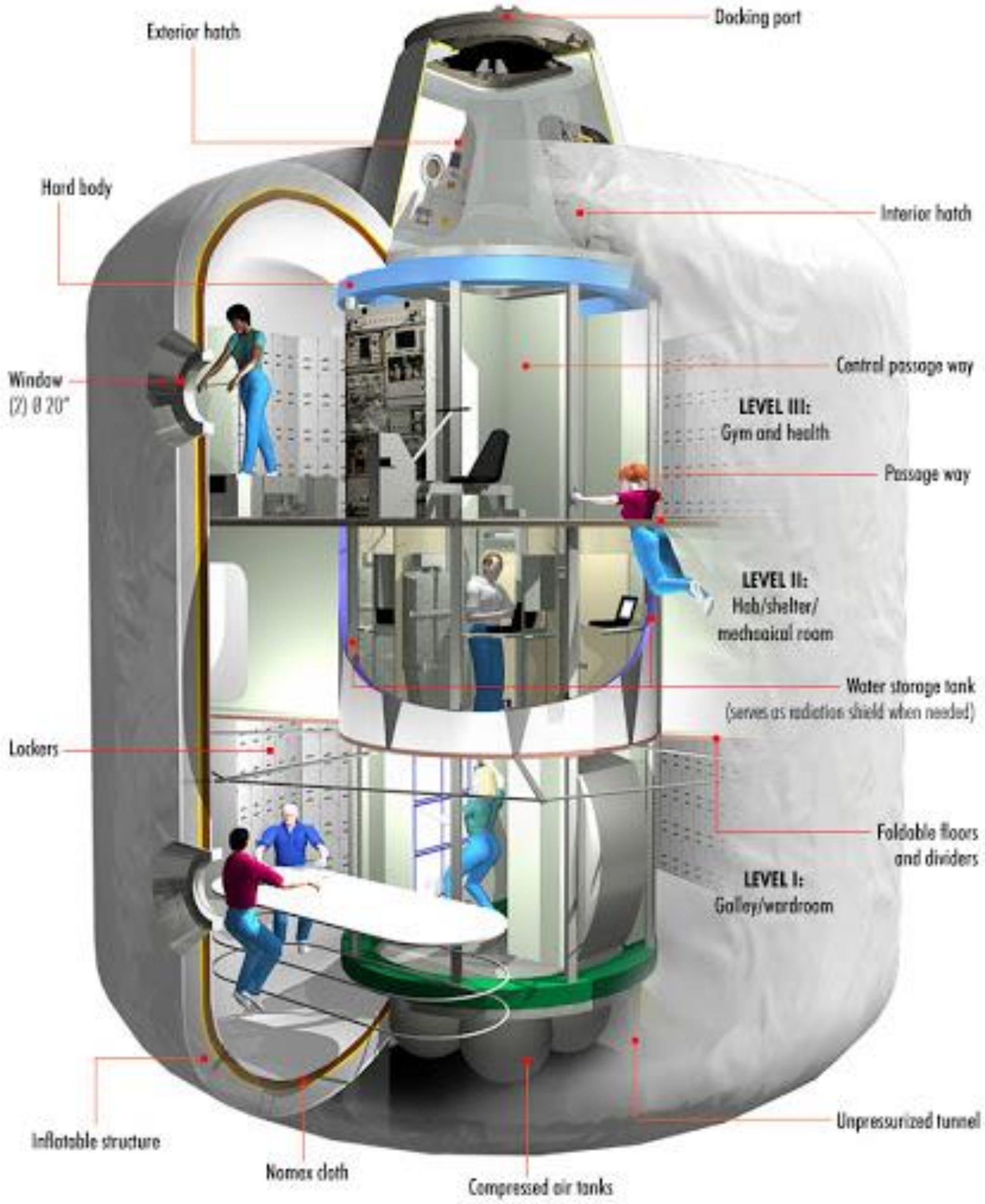

Figure 13-3: TransHab configuration (Source: russianspaceweb.com)

Currently, the leaders on the field of developing the inflatable technology are the US based private companies, such as Bigelow Aerospace, ILC Dover and Sierra Nevada. Only one of these, Bigelow Aerospace, has been able to send their habitats into space so far. In 2006, their first habitat, Genesis I, was launch into orbit on board Russian Dnepr rocket. The habitat served as a testbed for inflatable technology and remained without crew, like its follower, Genesis II, launched in 2007. After Genesis programme, Bigelow developed three new modules, Galaxy, Sundancer and BA-330, but none of them flew due to delays in developing commercial vehicles needed to fly people to orbit. Fortunately, at the same time NASA decided to test the technology on ISS and planned to send the BA-330 into orbit. However, first a smaller version, called BEAM, needed to be tested. RD[74] BEAM (Figure 13-4), of 16 m^3 pressurised volume, was launched into orbit in 2016 and was attached to ISS. BEAM has been successfully fulfilling its purpose to date, and will remain in service until 2028.

Figure 13-4: BEAM module (Source: abcnews)

In Europe, ESA has studied inflatables for space applications with its industry partners. Some of the projects are Inflatable Capture Mechanism (ICM), Inflatable manned module (IMOD), Flexible Expandable Commercial Module (FLECS) and SpaceHaven.

13.1 Challenges with Human-Rated Inflatables

The main benefits of using an inflatable space habitat is its ability to package the structure into various compact shapes for launch and that way save a lot of volume during transport. In addition, inflatables ensure a significant mass reduction, when compared to traditional rigid modules, in three ways: (1) the use of very high specific strength filamentary materials; (2) reduced design loads since the design is driven by in-space loads following deployment, instead of loads induced during launch, which are typically higher; and (3) reduction in the mass of the launch shroud and launch system due to the compactness of the packaged inflatable structure. RD[76] However, at the moment, the use of fibrous inflatable structures in space applications is approached with a lot of hesitation due to the of use non-traditional materials, non-traditional manufacturing techniques, and load paths in structures that are not precisely defined. Although high-performance fibrous components such as straps, cords, and fabrics possess extremely high specific strengths, their use is challenging because they are highly nonlinear, subject to creep, and subject to imperfections created during the fabrication process. In order to make the most efficient use of the specific properties and benefits of these materials structural concepts must be developed with rationality and robustness. RD[78]

13.1.1 Softgood Properties

Jones (2018) summarises the most important factors in softgoods properties causing the complex non-linear behaviour. High-performance fibre based materials are hierarchical by nature. Generally, five levels of structural hierarchy is present in a typical human-rated inflatable module (Figure 13-5). To that end, factors that affect the textile structure and properties are present at each level, from manufacturing processes and sizings used at the fibre and yarn level, weave type and resin selection at the webbing level and weave, friction, to stitch properties at the inflatable module level.

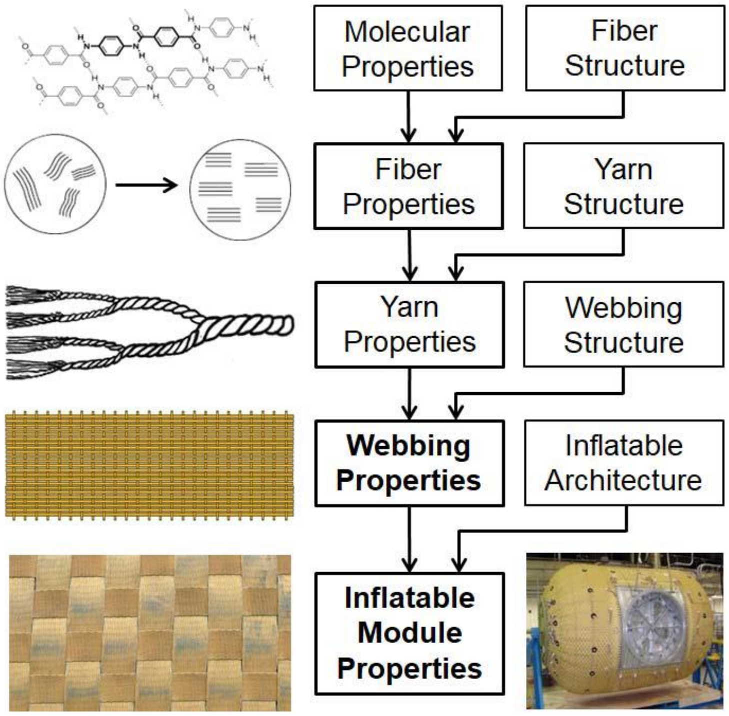

Figure 13-5: Structural hierarchy in high-strength softgoods structures RD[75]

Another challenging aspect of fibrous materials is their load versus strain behaviour (Figure 13-6). The strain of the material is dependent on the number and frequency of the load cycles as the fibres get stiffer when the materials is continuously exposed to loads. When the material is first loaded the relatively loose fibres still contain twists and crimps which results in non-linear and soft strain behaviour until the fibre-lock is reached (Figure 13-6, a). After load-cycling the material becomes stiffer and consistent (preconditioned) (Figure 13-6, b) causing the stress in the fibres to be distributed more evenly. If the material is left unloaded for a period of time, the behaviour of the fibres starts to move back to non-linear strain (Figure 13-6, c) due to partial relaxation and recovery of crimp and twist. As generally preconditioned curve is used in the modelling of softgoods, this changing behaviour makes tracking the load distributions challenging.

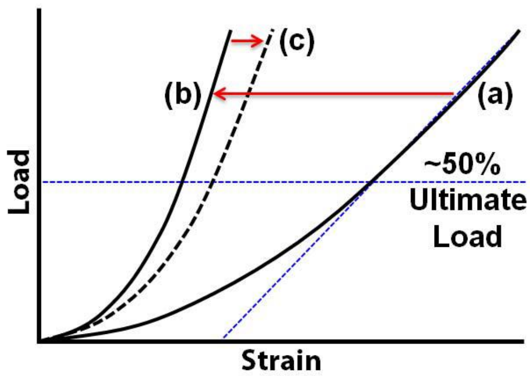

Figure 13-6: Load vs strain behaviour RD[75]

One of the most difficult material properties to test is the long-term behaviour of the restraint layer (Figure 13-7) because of high costs and a need for a large, environmentally controlled test facility that can accommodate a large number of specimens for many years. There are two main factors that influence the creep time to failure. First, large variations in strength properties of tested materials reduce the structural efficiency of an article by requiring using a larger margin on the design load due to the higher uncertainty. Second, a non-uniform load distribution on these large strength variations broadens the time to failure range even further.

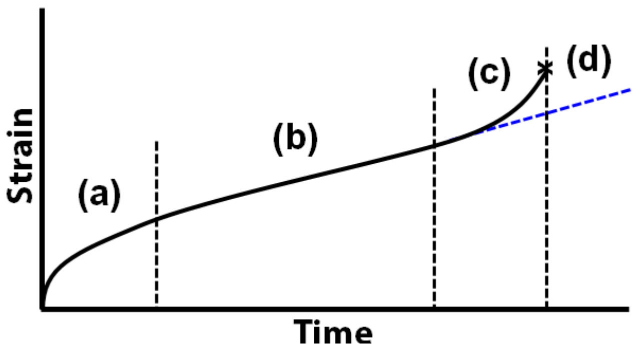

Figure 13-7: Viscoelastic creep RD[75]

13.1.2 Inflatable Shell Structure

Typically, the shell of the inflatable contains a number of layers of softgoods with each its own function (Figure 13-8). The inner layer is a liner, usually from Nomex, that protects the bladders from punctures, is durable and flame resistant. Behind the liner, there are three layers of bladders from a flexible and low permeable material, such as Combitherm, to contain the pressure. The outward working force from the internal pressure on the bladders is taken up by a restraint layer, which provides the habitat with a structural support. The material used is generally Kevlar or Vectran. The restraint layer is protected from outside by a MMOD shielding. The shielding consists of several layers of Nextel fabric, separated by low-density cored polyurethane foam. For thermal regulation, the MMOD is covered with MLI, made from a number of layers of Nylon reinforced double aluminised Mylar, sandwiched by an inner and outer layer of Kapton.

In some cases, also an outer liner is used for the protection against atomic oxygen. RD[77]

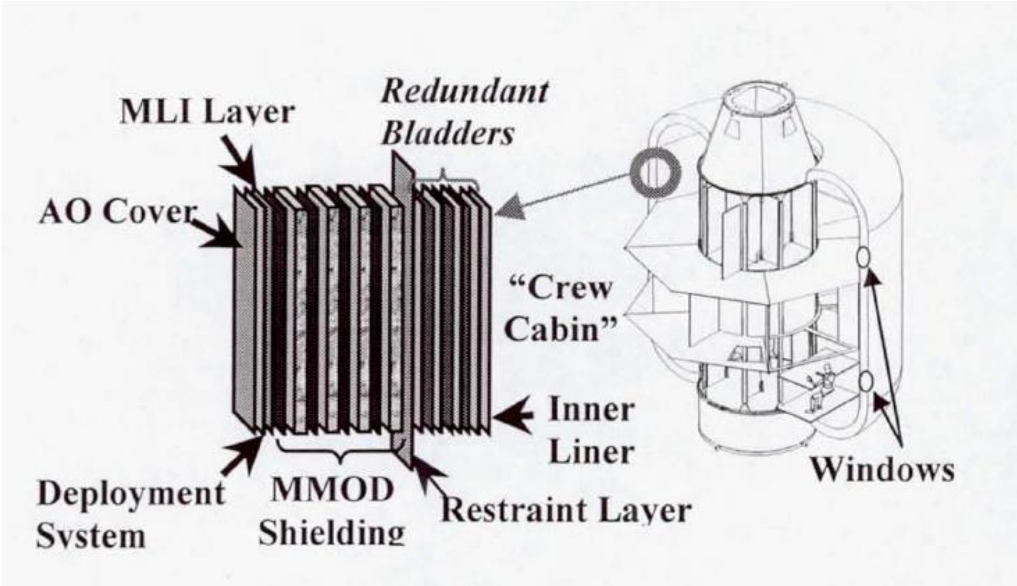

Figure 13-8: Inflatable shell structure RD[77]

The most critical of all is the design of the restraint layer, as it needs to take up all the loads present in inflatable and give the habitat its proper shape. NASA has been focussing most of its research on different types of restraint (Figure 13-9) based on webbing, cordage and broadcloth fabric. RD[75] The most common designs are compared below:

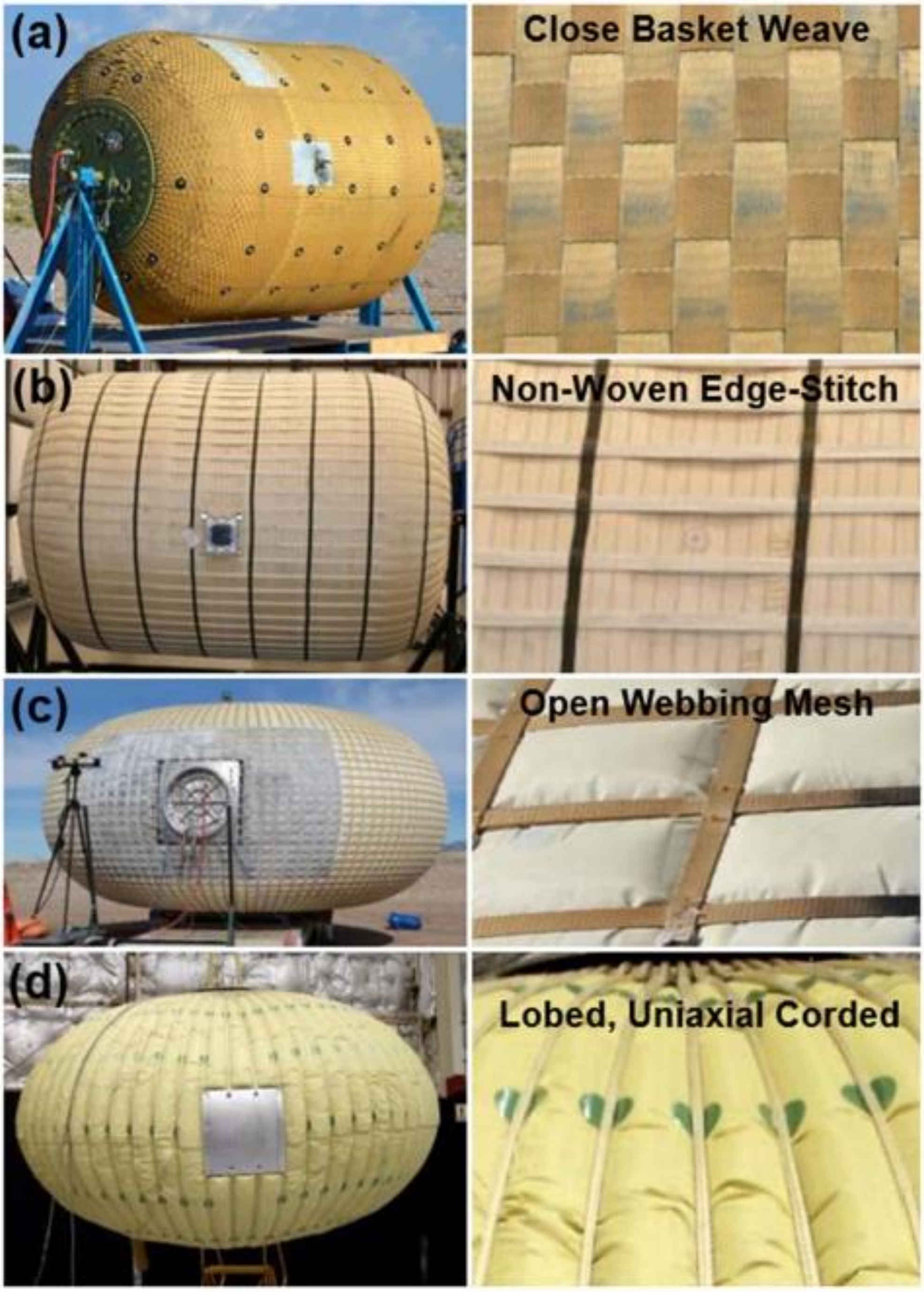

Figure 13-9: Four most common restraint layer architectures RD[75]

a) Closed basket weave (TransHab): webbings are woven into a tight mat

   + Highly robust to damage due to the high friction between webbings, and allows the use of a non-structural bladder layer

   - Load distribution is highly non-deterministic

b) Non-woven edge stitch: the hoop webbings stitched together along their adjacent edges, and with axial webbing overlaid

+ Simplified fabrication process

- Complex local stresses in the stitched webbings, challenging load determination, seam design especially critical

c) Open webbing mesh: open webbing over a structural fabric

+ Lighter construction but less robust than a and b

- Less robust than the first two approaches, subject to indeterminate loads due to the cross-overs and interaction between the hoop and axial webbings

d) Lobed, uniaxially corded: low hoop stress geometry, the principle loads are unidirectional. The low hoop loads are carried by lobes formed between the cordage

+ One of the lightest approaches possible, the principle load paths are determinate, but not redundant

- Design relies on a properly offloaded and seamed fabric layer designed to take the dynamic load of a cord failure, the cords must be sized to carry the additional load from a failure to add robustness

13.1.3 Approach for Structural Certification

Currently, no clear certification method exists for human-rated space inflatables. The standards that are in place offer little guidance for the design and testing. The only criteria defined is the Factor of Safety (FOS) derived from airship design standards.

Jones (2018) proposes an updated approach to defining a design factor for these structures, taking into account material strength and load variability, number and type of tests, and effects of damage and degradation over time. The design factor is calculated using the FOS in combination with the loss and load factors and an uncertainty factor based on the combined risks listed under additional factors, outlined below:

Factor of Safety (FOS) for softgoods = 4

The FOS specifies the actual failure load versus the intended design load of the structure and is set based on the level of criticality of the structure, its application, and the relative risk assessed from the level of testing and analysis that has been performed on similar structural architectures and materials. With appropriate investigation and characterization of the derating factors below, however, a baseline FOS of 2 to 3 should be feasible.

Knockdown / Derating Factors:

Knockdown factors are all architecture and material dependent at both the component and system level. It is therefore not possible to create a universal design factor for these structures.

1. **Loss factors (LsF) -** These are damage and degradation factors that reduce the effective strength or lifetime of the inflatable article.
   - Thermal-vacuum effects
   - Ultraviolet (UV) light exposure

- Radiation
- Folding from packaging
- Abrasion
- Tear strength
- Joining efficiency (seams, stitches, and splices)
- Fatigue
- Creep

2. **Load factors (LdF)** - These are factors that increase the effective loads beyond the statically determined pressure loading based on the geometry of the inflatable article.
   - High variance in material strength properties
   - Non-uniform load distribution
   - Dynamic loading
   - Built-up Inflatable architecture effects

3. **Additional factors** - These are influence factors that may affect the overall design factor but are not classified as a load or loss factor. These elements are assessed for each design and their combined effect is an Uncertainty Factor (UF) that is an additional multiplier of the design factor.
   - Number and type of samples tested
   - Ability to verify full-scale, as-built restraint layer
   - Structural redundancy
   - Ability to repair restraint layer
   - Programmatic controls on fabrication and ground handling
   - Experience with design and materials, and level of testing and analysis performed
   - Level of Uncertainty in loss and load factors

Design Factor = FOS * [(LdF1 * LdF2 *… LdFn) / (LsF1 * LsF1 * … LsFn)] * UF

Where: FOS = Factor of Safety, UF = Uncertainty Factor, LdFn = Load Factors and LsFn = Loss Factors.

The calculated design factor multiplied by the limit loads of the softgoods restraint layer elements gives the minimum ultimate tensile strength (UTS) of each component to meet the certification requirement.

The best approach to obtain these values is to use systematic method of testing multiple factors at once and a statistically based analysis to best approximate the as-built article behaviour.

13.2 Rigidization of Softgoods

It is known that the inflation gas can be relied upon for only a finite period of time to provide post-deployment structural rigidity. That is because the inflation gas will escape through tiny imperfections in the inflatable skin such as pinholes that have appeared during manufacture, folding or deployment. The higher the inflation pressure, the faster this process will occur. Therefore, it is suggested to use methods for rigidization of the structure to ensure the structural stability even after a loss of pressure. RD[80].

When choosing a suitable rigidization method, the following factors should be considered:

- Testing (is it reversible)
- Stowage and handling (ease of handling, storage life)
- Energy needed for rigidization process
- Outgassing
- Performance (how uniform is the final result)
- Structural performance (can it handle the loads)

Schenk et al (2014) propose the following methods for rigidization in space applications. Below an overview is presented of the methods that could be applied to inflatable habitats:

1. **UV-Setting Resins**

   UV-setting resins can be cured in two ways, using either environmental or lamp-based sources. The requirement for a sufficient curing is the right choice of material and the thickness of the structure, as UV-radiation needs to reach through the material. The advantage of this method is long storage life, low outgassing and possibility of using variety of shapes of structures. The solar based curing enables passive curing, although there is a chance for uneven curing. The lamp-based curing provides the process with more control, at the expense of more power and complex system. UV-rigidization is irreversible.

2. **Thermosetting Resins**

   Thermally cured resins are compatible with a large number of fibres providing the structure with high strength and stiffness. These resins have low outgassing, good space resilience and low CTE properties. The rigidization happens either by using solar heat or embedded heaters. In case of solar heating the process is passive, whereas with embedded heaters it can be localised and controlled. There is almost no limit for the shape of the structure where this method can be used. The process is irreversible.

3. **Glass Transition Resins**

   Glass transition resins become rubbery at the glass transition temperature allowing the material to change shape. This property can be used in rigidization process by creating a structure that solidifies below a certain temperature. These types of resins are known to be less rigid than thermosets, but allow a reversible process by making multi-deployment possible.

4. **Stretched Metal Laminates**

   Stretched metal laminates are made of thin layers of metals bonded to thin layers of polymers. The laminates are used in the skin of inflatables. When inflating the structure, the yield stress of metals is exceeded slightly. After deflation, the metals gain their pre-stress state and exposed to compression, while the polymer is in tension. The rigidity in the metals is gained through strain hardening locally, and removal of imperfections globally. The laminates are simple to manufacture and handle, rigidize predictably, have extremely low outgassing, can be stored almost indefinitely and suffer few radiation effects, depending on the choice of polymer. The process is also reversible, with some degradation in structural performance with each subsequent deployment. The laminates can only be used in structures with low structural loads.

5. **Gas and Vapour Cured Resins**

   Gas and vapour cures resins are increasingly less used in space application due to potential outgassing of hazardous catalyst. Experiments have been done, however, with water-setting resin impregnated fibreglass, polyurethane polymers rigidized by volatile peroxide vapour and polyurethane foam that rigidizes in a self-propagating reaction initiated by an aerosol delivered catalyst. The process allows passive curing and large number of resin-fibre combinations.

6. **Solvent Boil-Off Rigidization**

   Solvent boil-off method is very limited for space application due to large amount of outgassing and high probability for uneven rigidization. In this method a solvent is used to cover resin to make it flexible for folding and stowage. When the solvent is allowed to evaporate the resin will reach it rigidity. Tests have been done with hydrogels, polyvinyl alcohol and gelatin.

7. **Foam Rigidization**

   Foam rigidization can happen in two ways: 1. A structure is filled with foam which serves as a driver for deployment, or 2. A structure can be coated with material that starts to foam with heat, vacuum or catalyst. It is complicated, however, to ensure a uniform filling and keep the outgassing low during foaming.

13.3 Fibre-Based Composites for Space Structures

Fibre-based composites are already increasingly used in wind energy, transportation, building industry, sports, leisure, and machinery building. The fibres, which have high tenacity and high Young's modulus are combined with matrix resulting in products with high stiffness and tenacity, high vibration damping, high crash energy dissipation, and fatigue resistance. The low density and the adjustability of the mechanical properties via proper orientation of fibres offer a weight advantage up to 60% in comparison to steel and up to 25% in comparison to alumina. In addition, fibre-based composites are noncorrosive, have a high geometrical design freedom, low heat elongation, and low die costs. RD[81]

Two-dimensional (2D) laminated composites are characterized by their in-plane high specific stiffness and strength, but are not suitable for out-of-plane loading conditions.

Instead three-dimensional (3D) composites are used where out-of-plane loading conditions occur, such as wind turbine blades, stringers and stiffeners in aircraft, pressure vessels and construction applications, etc. When in 2D composites fibres are placed only in x and y direction, then in 3D composites additional binder fibres are added in z-direction. This ensures that these composites are delamination resistant and can produce near-net-shapes, compared to 2D composites.

Due to these advantages, 3D composites are suitable candidates for aerospace applications and could offer interesting design solutions in substituting metallic components for weight reduction.

13.3.1 3D Composites and Suggestions for Applications

The following methods could be considered for replacing the metallic parts in habitat design with lightweight composite components:

1. **3D Weaving**
   3D woven structures can be divided into four categories: (a) orthogonal, (b) through-the-thickness angle interlock, (c) layer-to-layer angle interlock, and (d) fully interlaced (Figure 13-10). They have a high formability, which means they can easily take the shape of the mold in case of complex composite designs and a highly porous structure, which decreases resin infusion time.
   The benefit of 3D orthogonal woven fabrics is that they exhibit less or no yarn crimp - the difference in length of yarn, before and after weaving. Therefore, mechanical properties of fibres are optimally used in warp and weft directions.
   The shape of 3D woven fabrics can be controlled in all three directions during the weaving process, producing near net shape fabrics such as I-beams and stiffeners. This ensures that the fabricated preforms could be placed directly in the mold without any additional work. The layering of the laminates is no longer needed, like with 2D composites, because the single fabric has a considerable thickness that provides the full three-dimensional reinforcement. RD[82]

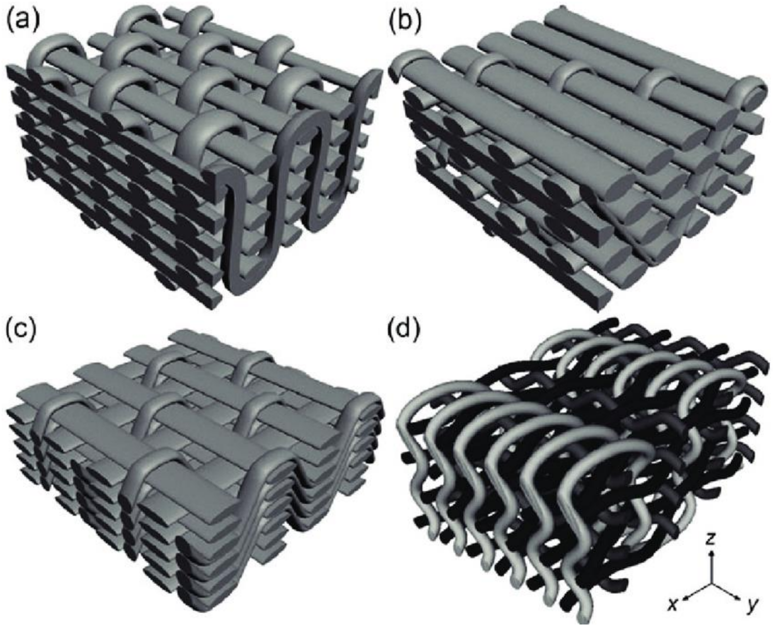

Figure 13-10: 3d woven architectures (Stig, Hallström, 2012)

2. **Braiding**
   Braided component is manufactured by intertwining of two or more yarn systems to form an integral structure (Figure 13-11). When woven fabrics have orthogonal interlacement then the braids can be constructed using a wide range of different angles. These composites have superior toughness and fatigue strength in comparison to filament wound composites. An additional set of axial yarns can be integrated to the braiding process to fabricate triaxial braids which are more stable and exhibit nearly isotropic properties.
   Braids can be fabricated either as seamless tubes or flat fabrics with a continuous selvedge. Composites manufactured by using the braided preforms exhibit superior strength and crack resistance in comparison to broadcloth composites, due to fibre continuity, i.e. composites with braided holes exhibit about 1.8 times the strength in comparison to drilled holes. RD[83].

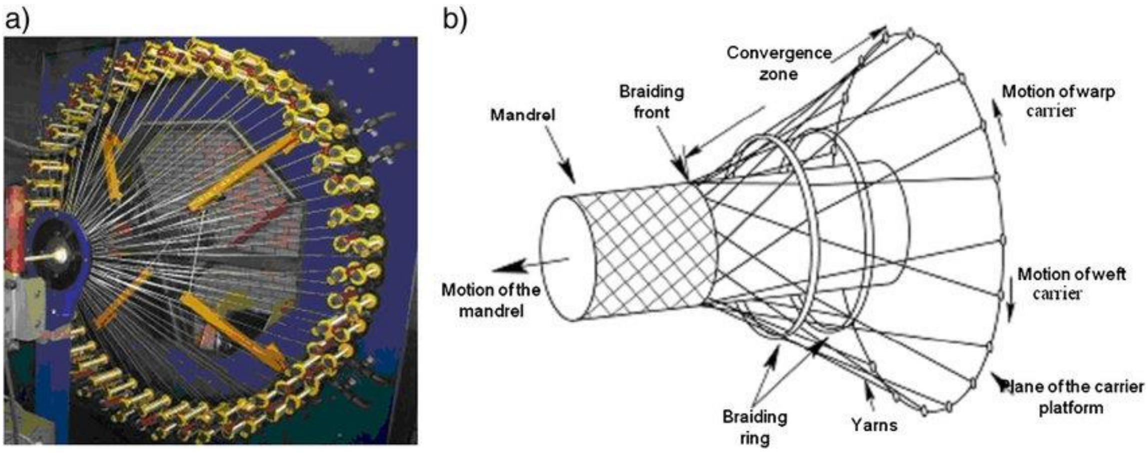

Figure 13-11: Composite braiding (Guyader, Gabor, Hamelin, 2013)

3. **Pultrusion**
Pultrusion allows manufacturing of fibre composite materials with constant cross section. In this process resin-impregnated fibres are pulled through a die, in contrary to extrusion process, where the material is pushed through a die (Figure 13-12). The reinforcing fibres, rovings, or textiles are stored on a bobbin creel and are pulled through a resin-impregnation bath and a heated pultrusion tool (die). The resin of the impregnated fibres is cured while being led through the tool. The finished profile leaves the tool continuously by means of reciprocating pullers or a caterpillar and is cut to length by a saw. Because of many possible options for size and cross sections of the profiles, pultrusion products are increasingly used in the building industry (e.g. window frames, tubes, strips), in sporting goods (sailing masts, skis, walking sticks), transportation (train, bus, airplane cover panels), and in the furniture industry (slatted frame).
The process exhibits high process stability and a high output. The fibres are mostly applied as low-cost uni-directionally oriented reinforcing fibres/ rovings. In case of biaxial or multiaxial mechanical loads, additional nonwovens (so-called surface veils), woven fabrics, braids, or non-crimp fabrics are added into the die.
The combination of braiding and pultrusion is called braid-pultrusion. In this respect, the braiding technique is integrated into the pultrusion process to give the otherwise unidirectional pultruded profiles a better torsional stability. RD[84].

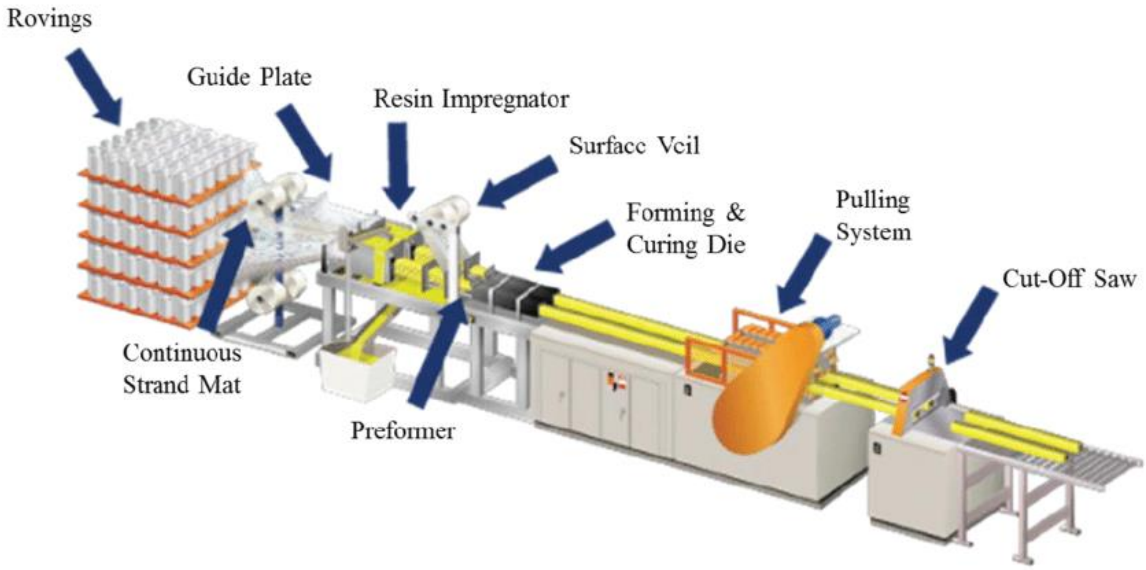

Figure 13-12: Pultrusion process (Source: Creative Pultrusions)

4. **Sandwich panels**
A sandwich-structured composite (Figure 13-13) is a special class of composite materials that is fabricated by attaching two thin but stiff skins to a lightweight but thick core. The core material is normally low strength material, but its higher thickness provides the sandwich composite with high bending stiffness with overall low density. Open- and closed-cell-structured foams, like

polyethersulfone polyvinylchloride, polyurethane, polyethylene or polystyrene foams, balsa wood, syntactic foams, and honeycombs are commonly used core materials. Sometimes, the honeycomb structure is filled with other foams for added strength .Open- and closed-cell metal foam can also be used as core materials. Laminates of glass or carbon fibre-reinforced thermoplastics or mainly thermoset polymers are widely used as skin materials. Sheet metal is used as skin material in some cases.

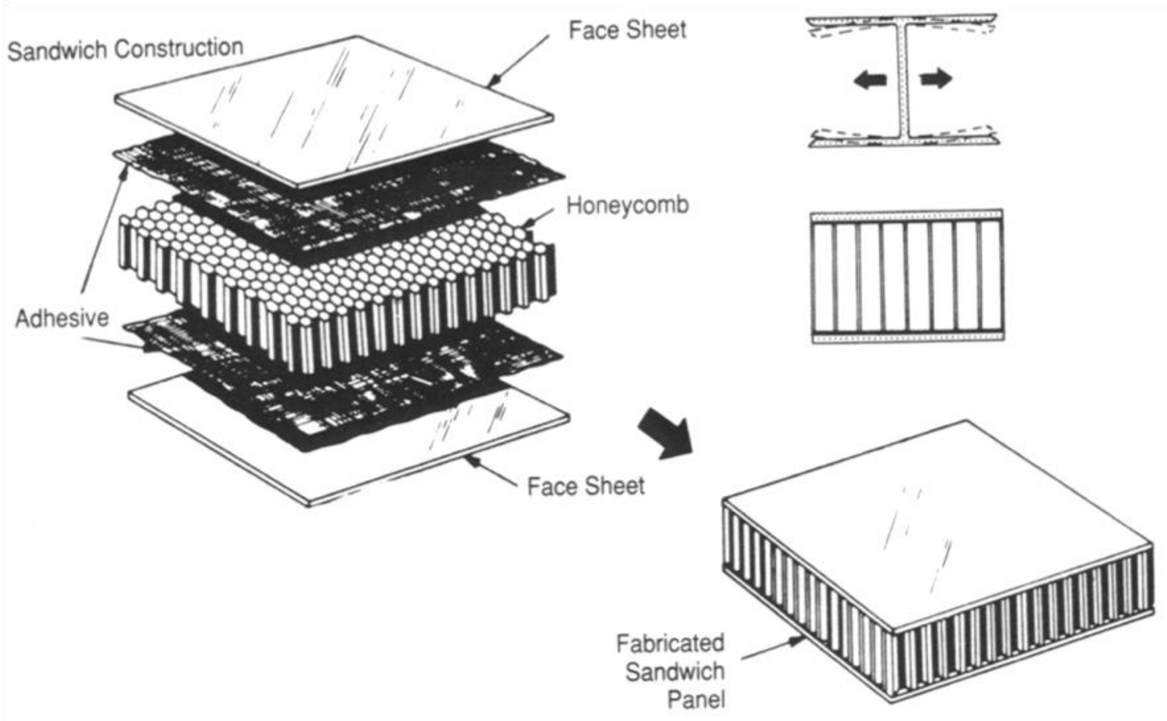

Figure 13-13: Composition of sandwich panel (Kesarwani, 2017)

14 CONCLUSIONS

14.1 Satisfaction of Requirements

The SOM habitat design has been reviewed in terms of structures / configuration / radiation / thermal and power design, as well as requirements for internal architecture and life support.

An analysis has been done on all logistics related to using the habitat on the Moon, i.e. transfer and thermal maintenance, dismounting from lander (power connection, mechanisms, etc.) and placement into the village, as well as normal usage of the habitat.

Small solar panels are proposed to be mounted between windows for transfer thermal heating.

A set of Use Cases has been defined showing both challenges and requirements/needs for each use case.

Crew accommodation features have been proposed such as work, exercise, EVA suit donning, medical, etc.

Illumination condition assumptions and power requirements (e.g. 59kW assumption) will need more detailed work to be confirmed e.g. possibility to be provided by service module in case of multi-kW thermal requirements during transfer, either by solar power or ELHS. ISS/Gateway standards for voltage were applied.

Note that currently emerging standards are based on ORION and Lunar Gateway, but will need to be expanded for high power designs.

A first proposal for the material selection for the inflatables was given, and also used in the radiation analysis. Radiation ray-tracing analysis has been done.

Mass estimates for interior design, life support, thermal and power support and structures and hinges design, were redone taking into account ESA standard margins for pre-assessment design. Mass improvements such as the use of 3D composites were proposed. Radiation tolerance standards are based on LEO dose limits.

14.2 Compliance Matrix

Primary Objectives:	Objectives met?	Explanation
Review the boundary conditions of the performed SOM-ESA-MIT Moon Village concept study	Yes	Review included identification of several launch strategies
Identify requirements of the habitat module w.r.t. lunar environment	Yes	As part of the CONOPS analysis, needs were identified for each sub-system
Deliver habitat functional design features	Yes	Sizing was done for structures, power, thermal, life support, materials, mechanisms and radiation
Define habitat interior design	Yes	Crew accommodation needs and sizing

Primary Objectives:	Objectives met?	Explanation
features		was performed
Identify applicable standards and interfaces	Partially	Boundary conditions for habitat airlock interfaces were identified, as well as power standards
Secondary Objectives:		
Define a rough Concept of Operations and ROM running costs for the habitation module	Partially	A full concept of operations was created with needs and challenges for each use case. No running costs were established
Propose a baseline for launch and delivery to the lunar surface	Yes	SLS and StarShip identified as potential launchers
Assess potential In-Situ Resource Utilisation (ISRU)	Partially	No sizing was done however the need for water or regolith based radiation protection, potentially created using ISRU, was established

Table 14-1: Compliance matrix

14.3 Conclusions and Further Study Areas

The proposed habitat mass by SOM is too high for SLS launched lander; an investigation is proposed to study 2-floor modules of maximum 28.6T Mass (possibly to be integrated on lunar surface), or alternatively use the StarShip launcher to launch the 4-floor habitat. SOM's habitat size, however, is compatible with SLS type fairing.

Launch mass and logistics can be optimised by e.g. launching crew accommodation and life support facilities (such as redundant ones, or nitrogen/oxygen/water tanks) could need to be launched separately due to its high total mass; this would have a cascading effect since launching a high mass is also increasing the mass/size of other building blocks such as moveable crane.

Feasibility of RDV (possibly with uncooperative target) in LTO is to be investigated. As this is un-crewed, the RDV could be performed at an optimal time (for example at apogee) or timed over several orbits.

Other modules such as the high mass lander, orbit service module for the habitat, and long lifetime lunar polar power station (e.g. fission reactor) and radiator plants, lunar crane, were not part of the scope of this study, and could be designed in the future.

Inflatable structures are found to be insufficient for radiation protection; a larger contribution is expected from internal equipment. Protection against solar particle events will need to come from additional layers of e.g. water or regolith, as well as optimization of internal equipment placing (e.g. move beds to lower floor), or moving the ground floor below the lunar surface.

There is a low European heritage for using inflatable technologies in space; several advancements will need to be done (bladder design, coatings, shell, MMOD, estimation of leak rate etc.); as well as investigation into alternative methods such as 3D printing for complementing the inflatable structures for MMOD and Radiation protection.

Crew accommodation facilities, as well as life support facilities, as well as possible updates (e.g. launcher adapter fitting, precise inflatable stowing, as well as estimation of spare parts, is proposed to be designed in a follow-up study.

14.4 Final Considerations

The CDF would like to thank SOM for their active participation during the study, and being part of the team.

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15 REFERENCES

15.1.1.1 Chapter 3 System References

RD[1] Moon Village – ESA-SOM-MIT outcomes (Study Kick-Off slides), Presentation, Skidmore, Owings & Merrill (SOM), 23/02/2020

RD[2] Review and Analysis of (European) Building Blocks for a Future Moon Village, Conference Paper, David Binns et al., 68th International Astronautical Congress (IAC), Adelaide (Australia), 25-29/09/ 2017

RD[3] Encyclopedia Astornautica, http://www.astronautix.com/index.html

RD[4] M. Braun, N.G. Veronica Trivino, S. Hosseini, R. Schonenborg, M. Landgraf, Human lunar return: An analysis of human lunar exploration scenarios within the upcoming decade, Acta Astronautica (2020), doi: https://doi.org/10.1016/j.actaastro.2020.03.037

RD[5] Cis-Lunar Transfer Vehicle (CLTV) CDF Study, ESA-ESTEC Concurrent Design Facility Team, CDF Study External Final Presentation, 18/11/2019

RD[6] Lunar and Interplanetary Trajectories, Book, Robin Biesbroek, ISBN 978-3-319-26981-8, Springer International Publishing, 2016

RD[7] MAS Working Paper No. 603 "Human Lunar Exploration Precursor Program" - Mission Analysis Guidelines - ESA HSO-K-TN-0008, F. Renk, R. Jehn, W. Martens, L. Bucci, B. De Vogeleer, Issue 0, Revision 3

RD[8] Space Launch System (SLS) Mission Planner's Guide, David Alan Smith, NASA Marshall Space Flight Center, Huntsville, AL, United States, Revision A, 19/12/2018, available at https://ntrs.nasa.gov/search.jsp?R=20170005323

RD[9] Crew replace pane on ISS Cupola window, Chris Bergin, NASASpaceflight.com, April 15, 2017, https://www.nasaspaceflight.com/2017/04/crew-replace-cupola-window-iss/ (accessed on Feb 2020)

RD[10] ISS Utilization: Cupola Observation Module of the ISS, available at: https://directory.eoportal.org/web/eoportal/satellite-missions/i/iss-cupola (accessed on Feb 2020)

RD[11] Joint Quest Airlock Overview, National Aeronautics and Space Administration, 21/02/2019,https://www.nasa.gov/mission_pages/station/structure/elements/joint-quest-airlock (accessed on Feb 2020)

RD[12] Starship Users Guide, SpaceX, Revision 1.0, March 2020, available at https://www.spacex.com/sites/spacex/files/starship_users_guide_v1.pdf

15.1.1.2 Chapter 4 Crew Accommodation References

RD[13] Designing, Sizing, and Integrating a Surface Base, Chapter 13 by P. Eckart, Y. Ishikawa, K.J. Kennedy in: Human Spaceflight: Mission Analysis and Design, Ed. W.J. Larson, L.K. Pranke, New York, NY McGraw-Hill 2007, ISBN-13: 9780072368116

RD[14] Minimum Acceptable Net Habitable Volume for Long-Duration Exploration Missions, Subject Matter Expert Consensus Session Report NASA Human Research Program, NASA/TM-2015-218564, A. Whitmire, L. Leveton, H. Broughton, M. Basner, A. Kearney, L. Ikuma, M. Morris, April 2015, https://ston.jsc.nasa.gov/collections/trs/_techrep/TM-2015-218564.pdf

RD[15] Partial Gravity Habitat Study with Application to Lunar Base Design. Technical Report NASA-CR-186048. S. Capps, J. Lorandos, E. Akhidime, M. Bunch, D. Lund, N. Moore, K. Murakawa, L. Bell, G. Trotti, D. Neubek, Sisakawa International Center for Space Architecture, January 01, 1989

RD[16] Energetics and mechanics for partial gravity locomotion. Aviation, space, and environmental medicine. 65. 815-23. Newman, Dava & Alexander, H & Webbon, Bruce. 1994

RD[17] Habitat Design and Assessment at Varying Gravity Levels. 44th International Conference on Environmental Systems, 13-17 July 2014, Tucson, Arizona, ICES-2014-264. D.L. Akin, K. McBryan, N. Limparis, N. D'Amore, C. Carlsen

RD[18] Human Integration Design Handbook (HIDH), NASA Special Publication SP-2010-3407/REV1, D.R. Liskowsky, W.W. Seitz et al., Revision 1 - June 2014, https://www.nasa.gov/sites/default/files/atoms/files/human_integration_design_handbook_revision_1.pdf

RD[19] Crew Accommodations, Chapter 18 by D. Stilwell, R. Boutros, J. H. Connolly in: Human Spaceflight: Mission Analysis and Design, Ed. W.J. Larson, L.K. Pranke, New York, NY McGraw-Hill 2007, ISBN-13: 9780072368116

RD[20] Extravehicular Activity (EVA) Systems, Chapter 22 by B.N. Griffin, P. Spampinato, R.C. Wilde in: Human Spaceflight: Mission Analysis and Design, Ed. W.J. Larson, L.K. Pranke, New York, NY McGraw-Hill 2007, ISBN-13: 9780072368116

RD[21] NASA - Food for Space Flight, Online Feature, Feb. 26, 2004, Website - retrieved 2020-02-02, https://www.nasa.gov/audience/forstudents/postsecondary/features/F_Food_for_Space_Flight.html

RD[22] Advanced Microgravity Compatible, Integrated Laundry (AMCIL), Umpqua Research Company, NASA SBIR 2011 Proposal Summary, Online - retrieved 2020-02-16, https://sbir.gsfc.nasa.gov/SBIR/abstracts/11/sbir/phase1/SBIR-11-1-X3.02-9012.html?solicitationId=SBIR_11_P1

15.1.1.3 Chapter 5 Config and Structures References

RD[23] SLS Mission Planners Guide, ESD 30000, 19 December 2018

15.1.1.4 Chapter 6 Mechanisms References

RD[24] Space Station Berthing Mechanisms – Attaching Large Structures On-Orbit that were Never Mated on the Ground, Aerospace Mechanisms Symposium (AMS), Robert M. Foster et al., 2004

RD[25] Advanced Docking Berthing System, NASA JPL, James L. Lewis, 2006

RD[26] Robotic Lunar Surface Operations 2, NASA JPL, Austin et al., 21-25 October 2019

RD[27] Human Spaceflight Vision – European Man-Tended Moon Base, CDF Study Report, January 2004

15.1.1.5 Chapter 7 Power References

RD[28] P. Gläser et al., "Illumination conditions at the lunar south pole using high resolution Digital Terrain Models from LOLA". *Icarus* 243 (2014), p 78-90.

RD[29] E. Marazico et al., "Illumination conditions of the lunar polar regions using LOLA topography". *Icarus* 211 (2011), p 1066-1081.

RD[30] R. M. Cataldo and J. M. Bozek, "Power Requirements for the First Lunar Outpost", 10th Symposium on Space Nuclear Power and Propulsion, Albuquerque, New Mexico, 1993, NASA TM 105925.

RD[31] L.S. Mason. "A Comparison of Fission Power System Options for Lunar and Mars Surface Applications", Space Technology and Applications International Forum, Albuquerque, New Mexico, February 12–16, 2006, NASA/TM—2006-214120.

RD[32] M.A. Gibson at al. "NASA's Kilopower Reactor Development and the Path to Higher Power Missions", IEEE Aerospace Conference, Big Sky, Montana, March 4, 2017, NASA/TM—2017-219467.

RD[33] L. S. Mason and D.I. Poston, "A Summary of NASA Architecture Studies Utilizing Fission Surface Power Technology, NASA/TM—2011-216819, AIAA–2010–6599.

15.1.1.6 Chapter 8 GNC References

RD[34] S. Theil and L. Bora, "Multiple beacons for supporting lunar landing navigation". *CEAS Space Journal* 10, 295-305(2018).

RD[35] D. Neveu et al., " Achieving Lunar Landing Hazard Detection and Avoidance in the Context of PILOT". Proceedings 10th International ESA Conference on Guidance, Navigation & Control Systems (ESAGNC2017); Salzburg, Austria, 2017.

RD[36] Hormigo et al. , "Flight Testing a Vision-Based Navigation and Hazard Detection and Avoidance (VN&HDA) Experiment over a Mars-Representative Terrain", 15th International Planetary Probe Workshop (2018).

RD[37] J. Huesing et al., "Engineering the Earth Return Orbiter Concept for A Potential Mars Sample Return Campaign". Ninth International Conference on Mars, Pasadena, USA (2019).

15.1.1.7 Chapter 9 Thermal References

RD[38] Vasavada, A. R., Paige, D. A., and Wood, S. E., 'Near-Surface Temperatures on Mercury and the Moon and the Stability of Polar Ice Deposits', Icarus, 141/2: 179–193 <doi:10.1006/icar.1999.6175>. (1999)

RD[39] Hayne, P. O., et al., "Global regolith thermophysical properties of the Moon from the Diviner Lunar Radiometer Experiment. Journal of Geophysical Research: Planets, 122, 2371–2400. https://doi.org/10.1002/2017JE005387, (2017).

RD[40] Keihm, S. J., Peters, K., Langseth, M. G., and Chute Jr., J. L., 'Apollo 15 measurement of lunar surface brightness temperatures. Thermal conductivity of the upper 1 1/2 meters of lunar Regolith', Earth and Planetary Science Letters, 19: 337–351. (1973)

RD[41] Horai, K.-I., Simmons, G., Kanamori, H., and Wones, D., 'Thermal diffusivity, conductivity and thermal inertia of Apollo 11 lunar material', Proceedings of the Lunar Science Conference, 3: 2243–2249. (1970)

RD[42] Cremers, C. J., and Birkebak, R. C., 'Thermal conductivity of fines from Apollo 12', Proceedings of the Lunar Science Conference, 3: 2311–2315, (1971).

RD[43] R. Birkebak, C. Cremers, "Thermophysical Properties of Lunar Materials: Part I", Advances in Heat Transfer, Vol. 10. (1974)

RD[44] Williams, J.-P., Paige, D. A. , Greenhagen, B. T., Sefton-Nash , E. "The global surface temperature of the Moon as measured by the Diviner Lunar Radiometer Experiment", Icarus, Vol. 283, p. 300-325, 2017. https://doi.org/10.1016/j.icarus.2016.08.012

RD[45] Williams, J.- P., Greenhagen, B. T., Paige, D. A., Schorghofer, N., Sefton- Nash, E., Hayne, P. O., et al. "Seasonal polar temperatures on the Moon. Journal of Geophysical Research: Planets, 124, p. 2505-2521, 2019. https://doi.org/10.1029/2019JE006028

RD[46] https://doi.org/10.1029/2019JE006028Gaier, J. R.; Jaworske, D. A., "Lunar Dust on Heat Rejection System Surfaces: Problems and Prospects", Space Technology and applications international forum 2007, New Mexico, Feb. 11-15, NASA/TM—2007-214814, 2007.

15.1.1.8 Chapter 10 Radiation References

RD[47] "Feasibility study of astronaut standardized career dose limits in LEO and the outlook for BLEO", Susan McKenna-Lawlor, http://dx.doi.org/10.1016/j.actaastro.2014.07.011

RD[48] "Cancer risk from exposure to galactic cosmic rays: implications for space exploration by human beings",F.A.Cucinotta, M.Durante, https://doi.org/10.1016/S1470-2045(06)70695-7

RD[49] OLTARIS On-Line Tool for the Assessment of Radiation in Space: https://oltaris.nasa.gov/

RD[50] "Evaluating shielding approaches to reduce space radiation cancer risks", F.A.Cucinotta et al., NASA TM-2012-217361, 2012

RD[51] "Space Radiation and Astronaut Safety", Erik Seedhouse, Springer International Publishing AG, 2018

RD[52] "A ready-to-use galactic cosmic ray model", Matthiä et al., Advances in Space research 51, 329-338, 2013

RD[53] "Badhwar-O'Neill 2010 Galactic Cosmic Ray flux model - Revised", P.M. O'Neill, IEEE Transactions on Nuclear Science 57(6), 3148-3153, 2010

RD[54] Tylka, A. J., et al., CREME96: A Revision of the Cosmic Ray Effects on Micro-Electronics Code, IEEE Trans. Nucl. Sci., 44, 2150-2160, 1997a

RD[55] L. Townsend et al., Carrington Flare of 1859 as a Prototypical Worst-Case Solar Energetic Particle Event; IEEE Transactions on Nuclear Science, vol. 50, NO. 6, Dec. 2003

RD[56] "A Novel Forecasting System for Solar Particle Events and Flares (FORSPEF) " A. Papaioannou, P.Jiggens https://iopscience.iop.org/article/10.1088/1742-6596/632/1/012075

RD[57] Simonsen, L. C., (1997) "Analysis of Lunar And Mars Habitation Modules for the Space Exploration Initiative (SEI)," in "Shielding Strategies for Human Space Exploration," Edited by J. W. Wilson, J. Miller, A. Konradi, and F. A. Cucinotta. NASA Conference Publication 360 https://ntrs.nasa.gov/archive/nasa/casi.ntrs.nasa.gov/19980137598.pdf

15.1.1.9 Chapter 11 Life Support References

RD[58] Limnospira indica PCC8005 growth in photobioreactor: model and simulation of the ISS and ground experiments,Life Sciences in Space Research, Poughon L. and all, vol 25, May 2020, p 53-65, https://doi.org/10.1016/j.lssr.2020.03.002

RD[59] MEliSSA Adaptation for Space, Phase II, ESA contract 20104/06/NL/CP, Technical Note 6

RD[60] ANITA2 Trace Gas Analyser for the ISS-Flight Model Finalization and ground test results, 49th International Conference on Environmental Systems, ICES-2019-67, July 2019

RD[61] Space Mission Utility and Requirements for a Heat Melt Compactor, John W. Fisher, 46th International Conference on Environmental Systems, ICES-2016-377, July 2016

RD[62] Life Support Baseline Values and Assumptions Document, NASA/TP-2015-218570/Rev1, January 2018

RD[63] Would Current International Space Station (ISS) Recycling Life Support Systems Save Mass on a Mars Transit?, Harry W. Jones, 47th International Conference on Environmental Systems, ICES-2017-85, July 2017

15.1.1.10 Chapter 12 Materials References

RD[64] Gerard Valle, Doug Litteken. Review of Habitable Softgoods Inflatable Design, Analysis, Testing, and Potential Space Applications (https://ntrs.nasa.gov/archive/nasa/casi.ntrs.nasa.gov/20190000847.pdf)

RD[65] Antonia Simone, Marco Nebiolo - Lightweight materials for advanced space structures. NanoInnovation 2016

RD[66] https://www.nasa.gov/feature/beam-update-expandable-habitat-reveals-important-early-performance-data

RD[67] https://twitter.com/bigelowspace/status/821771133066186752

RD[68] https://twitter.com/BigelowSpace/status/981966659933028352

RD[69] https://en.wikipedia.org/wiki/Bigelow_Expandable_Activity_Module

RD[70] Annex 1 – layers inflatable shell-options 1.3.xls

RD[71] Thomas C Jones - Structural Certification of Human-Rated Inflatable Space Structures

15.1.1.11 Chapter 13 Advanced Concepts & Inflatables References

RD[72] Inflatable composite habitat structures for lunar and mars exploration, Journal Paper, D. Cadogan, J. Stein, M. Grahne, 1999, DOI: 10.1016/S0094-5765(99)00103-4

RD[73] Inflatable technology: using flexible materials to make large structures, Conference Paper, D. A. Litteken, 2019, DOI: 10.1117/12.2500091

RD[74] Bigelow Aerospace. Colonizing Space One Module at a Time, Book, E. Seedhouse, 2015, ISBN 978-3-319-05197-0

RD[75] Structural Certification of Human-Rated Inflatable Space Structures, Paper, T. C. Jones, 2018, DOI: 10.1109/AERO.2018.8396616

RD[76] ISS TransHab: Architecture Description, Conference Paper, K. J. Kennedy, 1999, DOI: 10.4271/1999-01-2143

RD[77] TransHab : NASA's Large-Scale Inflatable Spacecraft, Conference Paper, H. de la Fuente, J. L. Raboin, G. R. Spexarth, G. D. Valle, 2000, DOI: 10.2514/6.2000-1822

RD[78] Non-Axisymmetric Inflatable Pressure Structure (NAIPS) Concept that Enables Mass Efficient Packageable Pressure Vessels with Sealable Openings, Technical Report, W. R. Doggett, T. C. Jones, W. S. Kenner, *et al*, 2016, NASA/TM-2016-219331

RD[79] Non-Axisymmetric Inflatable Pressure Structure (NAIPS) Full-Scale Pressure Test, Conference Paper, T. C. Jones, W. R. Doggett, J. E. Warren, *et al*, 2017, DOI: 10.2514/6.2017-0854

RD[80] Review of Inflatable Booms for Deployable Space Structures: Packing and Rigidization, Journal Paper, M. Schenk, A. D. Viquerat, K. A. Seffen, S. D. Guest, 2014, DOI: 10.2514/1.A32598

RD[81] Manufacturing techniques of composites for aerospace applications, Chapter in Book, K. Balasubramanian , Mohamed T.H. Sultan , N. Rajeswari, 2018, DOI:10.1016/b978-0-08-102131-6.00004-9

RD[82] 3D Woven Composites: From Weaving to Manufacturing, Chapter in Book, H. M. El-Dessouky, M. N. Saleh, 2018, DOI: 10.5772/intechopen.74311

RD[83] Three-dimensional braiding for composites: A review, Paper, K. Bilisik, 2013, DOI: 10.1177/0040517512450766

RD[84] The pultrusion process for polymer matrix composites, Chapter in Book, S. C. Joshi, 2012, DOI:10.1533/9780857096258.3.381

RD[85] Manufacturing of thermoplastic composite sandwich structures: A review of literature, Paper, J. Grünewald, P. Parlevliet, V. Altstädt, 2015, DOI: 10.1177/0892705715604681

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16 ACRONYMS

Acronym	Definition
ACLS	Advanced Closed Loop System
ADBS	Advanced Docking Berthing System
ANITA	Analysing Interferometer for Ambient Air
ArtEMISS	Arthrospira B space flight experiment
ATV	Automated Transfer Vehicle
BEAM	Bigelow Expandable Activity Module
BFO	Blood Forming Organs
CDF	Concurrent Design Facility
CFRP	Carbon Fibre Reinforced Polymer
CFU	Colony Forming Unit
CLTV	Cis-Lunar Transfer Vehicle
CM	Crew Member
CME	Coronal Mass Ejection
CoM	Centre of Mass
CoO	Concept of Operations
DC	Direct Current
DNA	Deoxyribonucleaic Acid
ECH	Electrochromics
ECLSS	Environmental Control and Life Support System
ECSS	European Cooperation for Space Standardisation
ELHS	European Large Heat Source
EOL	End Of Life
ESA	European Space Agency
ESM	European Service Module
EVA	Extra Vehicular Activity
GCR	Galactic Cosmic Rays
GTO	Geostationary Transfer Orbit
H2O	Water
HDA	Hazard Detection and Avoidance

Acronym	Definition
HITL	Human-In-The-Loop
HZE	High Energy and Charge Ions
IBDM	International Berthing and Docking Mechanism
ISP	Specific Impulse
ISRU	In-Situ Resource Utilisation
ISS	International Space Station
LANL	Los Alamos National Laboratory
LEO	Low Earth Orbit
LH2	Liquid Hydrogen
LLO	Low Lunar Orbit
LOI	Lunar Orbit Insertion
LOX	Liquid Oxygen
LPSS	Low Shock Separation System
LSS	Life Support System
LTO	Lunar Transfer Orbit
MCI	Mass Centering Inertia
MELiSSA	Micro-Ecological Life Support System Alternative
MIT	Massachusetts Institute of Technology
MLI	Multi Layer Insulation
MMOD	Micro-Meteoroid And Orbit Debris Protection
MPCV	Multi-Purpose Crew Vehicle
MPLM	Multi-Purpose Logistics Module
MSR	Mars Sample Return
NASA	National Aeronautics and Space Administration
NDI	Non Destructive Inspection
OLTARIS	On-Line Tool for the Assessment of Radiation In Space
OSR	Optical Solar Reflector
PDU	Power Distribution Unit
PMM	Permanent Multipurpose Module
PSR	Permanently Shadowed Region
RDV	Rendezvous

Acronym	Definition
RFC	Regenerative Fuel Cell
RGPS	Relative GPS
RHU	Radioisotope Heater Unit
ROM	Rough Order of Merit
RTG	Radioisotope Thermoelectric Generator
S/W	Software
SEP	Solar Energetic Particles
SLS	Space Launch System
SOM	Skidmore, Owings & Merrill
SPE	Solar Particle Event
SRMS	Shuttle Remote Manipulator System
SSM	Second Surface Mirror
SSRMS	Space Station Remote Manipulator System
SSVP	Sistema Stykovki Vnutrennego Perekhoda docking system
TASI	Thales Alenia Space Italy
TBC	To Be Confirmed
TCH	Thermochromics
TCS	Thermal Control Subsystem
TLI	Trans-Lunar Injection
UPA	Urine Processing Assembly
UT	Utility Truck
VDA	Vapour Deposited Aluminium
WPA	Water Processing Assembly
WRS	Water Recovery System
WSB	Weak Stability Boundary