AE-3200 Design Synthesis: Final Report

Lunar Helium-3 Mining

Faculty of Aerospace Engineering

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Challenge the future

Preface

The final step to obtain the Aerospace Engineering Bachelor at the Delft University of Technology is to successfully complete the Design Synthesis Exercise (DSE). In this exercise a group of ten students work together on a design project, in which they, aside from their technical knowledge, prove and further develop their personal and team-working skills, preparing them for work in a professional environment.

With the group we have assessed the feasibility and designed a mission to extract He-3 from the regolith on the Moon and transport it back to Earth to provide fuel to fusion power plants that will provide 10% of the global energy demand in 2040.

The progress we have made up to this date would not have been possible without the support, knowledge and contribution from our project tutor, Ir. Ron Noomen. Furthermore, we would like to thank Ir. Joost Ellerbroek, Dr.ir. Coen de Visser and Dr. Ranjita Bose, our project coaches, for their contribution and attendance to our progress meetings and discussions. Their tips and advice helped strengthen the report and design up to this date. Our gratitude goes towards Prof. Mosteshar and Sergiy Negoda, for their knowledge and opinion on legal matters related to mining He-3 in space. Furthermore, we would like to thank Ir. Zandbergen and Ir. Mooij for their support on general space system engineering issues, Prof. Dr. Lopez Cardozo for his insight on He-3 fusion, Prof. John Santarius for his knowledge on lunar He-3 mining, Prof. Dr. McInnes and Prof Dr. Dachwald for their insights on solar sailing, Matthijs Damen for information on fuel cells and Nico Golembiewski of AzurSpace for providing information on costs of solar cells. Finally, we would like to thank the client, Carlos Corral van Damme, who gave us the opportunity to work on this assignment and provided us with feedback throughout each stage.

Summary

With the current rate of depletion of common fossil fuels, the search for alternative clean energy sources has become an increasingly urgent matter. Alternative energy sources are being developed, but their scalability may not allow for full replacement of the energy today derived from fossil sources. Fusion energy is a possible alternative, but the technology is still in a very early phase, and for a radiation-free energy source, the Helium-3 isotope must be used as a fuel. This isotope is hard to obtain on Earth, but is abundant on the Moon. Together with the customer of this study, the following need statement was formulated:

"Due to an increasing demand for alternative energy sources, the feasibility of an end-to-end system for lunar He-3 mining, used for nuclear fusion to supply 10% of the global energy demand in 2040, must be assessed".

This report is the final product of the feasibility study. First, the general aspects of the mission were covered, including a market analysis concerning the energy market, space law aspects concerning the mission, and other preliminary subjects. It was found that 10% of the global energy demand in 2040 will require 200 tons per year of pure Helium-3. In the second phase, a conceptual design process was performed, trading off various concepts. The trade-offs eliminated the least optimal concepts, after which the very best elements of each concept were combined into a final end-to-end mission. This mission is defined as follows: a Space Plane carries cargo (empty Helium-3 canisters) to Earth orbit at 500 km height, where it will dock with a manned Space Dock. Between this Space Dock and lunar orbit, a Continuous-Thrust Transfer Vehicle (CTTV) will carry the cargo, powered by electric-propulsion VASIMR engines. In lunar orbit, the CTTV will dock with a Lunar Surface Access Module (LSAM), which will provide cargo transportation between lunar orbit and surface. On the Moon, a manned base with 2000 autonomous mining vehicles processes regolith volatiles, from which the Helium-3 is extracted and stored in Helium-3 canisters, which are then transported back to Earth.

Each segment of the end-to-end mission was designed at a conceptual level. These conceptual designs cover basic details including preliminary numbers on total mass, propellant, payload characteristics, and lunar operations, amongst others. It was found that the regolith will have to be processed at a rate of 640 tons per second to extract the required amount of Helium-3, due to its low abundance in the lunar regolith. Two elements of the mission were designed in detail: the CTTV and the LSAM.

The detailed designs cover all main subsystems and layout configurations of the segments. The subsystem design covers: trajectory design, propulsion, structural characteristics, thermal control, attitude determination and control, guidance and navigation, power supply, telemetry, tracking and command, command and data handling, and docking procedures, including payload handling. The results of the detailed design were two systems that consist of slightly modified off-the shelf components and technology with high Technology Readiness Levels (TRLs), complying with the requirements set at the beginning of the detailed design. Precursor missions and crew replacement mission were addressed on a conceptual level. It was found that for setting up the operation, 220 000 tons of equipment has to be brought into Earth orbit and onto the lunar surface, at an initial cost of 1500 B€.

Once the detailed designs were completed, the total cost budget of the entire mission was determined. A total annual cost of between 427 to 1347 B€ was derived, along with an annual expected revenue of 623 to 687 B€. The costs were compared with other energy sources, shown in Tab. 1. The cost per MWh is estimated to be lower than the selling price

Table 1: Comparison of cost per MWh.

Scale	**Cost [€/MWh]**		**Other sources**	**Selling price [€/MWh]**
0.1%	34.2 - 100.0		Solar	83 - 340
1.0%	20.3 - 68.4		Wind	80 - 255
10%	18.9 - 65.7		Natural Gas	30.4 - 115

of renewable energy. However,keep in mind, that the selling price indicated in the table, includes more components such as profit margin, company loans etc. The ten percent mission was shown to have an expected profit of -724 to 260 B€. It was concluded that only in the most optimal case, this mission will be economically interesting. With these numbers established, the feasibility of the study was assessed. It was concluded that the mission is only feasible under most optimal conditions. No unexpected costs may be incurred, no errors can be made, and no delays may occur. Scaling down the mission was found to reduce the profit. Once the feasibility was assessed, recommendations were made. These include increased investment in fusion technologies, development of more efficient Earth orbit access vehicles, and research on lunar mining operations. The final conclusion of the feasibility study found that lunar Helium-3 mining is unsuitable to provide 10% of the global energy demand in 2040.

Contents

Nomenclature

Abbreviation	Name
2D	Two-Dimensional
3D	Three-Dimensional
ACAS	Automatic Collision Avoidance System
ADCS	Attitude Determination Control System
ANS	Autonomous Navigation System
ASDS	Autonomous Satellite Docking System
ATHLETE	All-Terrain Hex-Limbed Extra-Terrestrial Explorer
ATV	Automated Transfer Vehicle
BER	Bit Error Rate
BFO	Blood-Forming Organs
BOL	Beginning-of-Life
BOP	Balance of Plant
C&DH	Command and Data Handling
CAD	Computer Aided Design
CBM	Common Berthing Mechanism
CBS	Cost Breakdown Structure
CEV	Crew Exploration Vehicle
CFD	Communication Flow Diagram
CMG	Control Momentum Gyroscope
CPU	Central Processing Unit
CTTV	Continuous-Thrust Transfer Vehicle
DEMO	Demonstration Power Plant
DET	Direct-Energy-Transfer
DOD	Depth-of-Discharge
DOF	Degrees of Freedom
DOS	Long Duration Orbital Station
DSA	Deep Space Antenna
DSE	Design Synthesis Exercise
DSN	Deep Space Network
EDRSS	European Data Relay Satellite System
EFDA	European Fusion Development Agreement
EIRP	Effective Isotropic Radiated Power
EO	Earth Orbit
EOL	End-of-Life
ERA	European Robotic Arm
ESA	European Space Agency
ESTRACK	ESA Tracking Station Network
EVA	Extra-Vehicular Activity
FBS	Functional Breakdown Structure
FCB	Functional Cargo Block
FFBD	Functional Flow Block Diagram
FSO	Free-Space Optical
GCR	Galactic Cosmic Rays
GEO	Geostationary Earth Orbit
GNS	Guidance and Navigation System
GNSS	Global Navigation Satellite System
HCS	Hard Capture System
He-3	Helium-3
HLLV	Heavy Lift Launch Vehicle
IC	International Cryogenics
IMU	Inertial Measurement Unit
ISRU	In-Situ-Resource-Utilisation
ISS	International Space Station
ITER	International Thermonuclear Experimental Reactor
ITU	International Communications Union
JAQ	Joint Airlock Quest
JEMRMS	Japanese Experiment Module Remote Manipulator System
LBCS	Lunar Base Communication Station
LCC	Life Cycle Cost
LEO	Low Earth Orbit
LEV	Lunar Excursion Vehicle
LL1	Earth-Lunar Lagrange Point 1
LL2	Earth-Lunar Lagrange Point 2
LLO	Low Lunar Orbit
LS	Lunar Segment
LSAM	Lunar Surface Access Module
LTR	Lunar Transfer Rover
LUBSIM	Lunar Base Simulator
LULOX	Lunar Liquid Oxygen
LV	Launch Vehicle
MANS	Microcosm Autonomous Navigation System
MCC	Mission Control Centre
MHD	Magneto Hydro Dynamic
MLI	Multi-Layered thermal Insulation
MMH	Mono Methyl Hydrazine
MMOI	Mass Moment of Inertia
MNS	Mission Need Statement
MSS	Mobile Servicing System
MTR	Mid Term Review
NASA	National Aeronautics and Space Administration
NDS	NASA Docking System
NEO	Near Earth Objects
NEP	Nuclear Electric Power plant
NTO	Nitrogen Tetroxide
OLC	Operations and Logistics Concept
OST	Outer Space Treaty
O&M	Operation and Maintenance
PCU	Power Control Unit
PERA	Payload Exchange Robotic Arm
PME	Proton Exchange Membrane
POS	Project Objective Statement
PPT	Peak-Power Tracker
PV	Photovoltaic
RAMS	Reliability, Availability, Maintainability and Safety
RCB	Reaction Control Block
R&D	Research and Development
R&DS	Rendezvous and Docking System
RDS	Russian Docking System
REID	Risk of Exposure Induced Death
RF	Radio Frequency
ROI	Return On Investment
SCS	Soft Capture System
SLS	Space Launch System
SMAD	Space Mission Analysis and Design
SNR	Signal to Noise Ratio
SOI	Sphere Of Influence
SPDM	Special Purpose Dexterous Manipulator
SPE	Solar Proton Events
SRB	Solid Rocket Booster
TDRSS	Tracking and Data Relay Satellite System
TFU	Theoretical First Unit
TGM	Telegoniometer
TRASIM	Transportation Simulator
TRL	Technology Readiness Level
TS	Transport Segment
TT	Transfer Time
TT&C	Telemetry, Tracking and Command
TU	Technical University
UDMH	Unsymmetrical Dimethyl Hydrazene
UHF	Ultra High Frequency
UKERC	UK Energy Research Company
UN	United Nations
USA	United States of America
VASIMR	Variable Specific Impulse Magneto-plasma Rocket
VDM	Videometer
V&V	Verification and Validation
WB	Wiffleball

Roman Symbols

Symbol	Name	Value	Unit
A	Area	-	m^2
a	Semi-major axis	-	m
a	Planetary albedo	-	-
AU	Astronomical Unit	$149.6 \cdot 10^9$	m
BC	Ballistic Coefficient	-	kg m^2
C	Battery capacity	-	Amp-hours
C	Coefficient	-	-
c	Two-dimensional coefficient	-	-
D	Diameter Antenna	-	m
DOD	Depth-of-Discharge	-	%
e	Eccentricity	-	-
E	Young's modulus	-	Pa
E	Energy	-	J, or dB
F	Visibility factor	-	-
f	Frequency	-	Hz
G	Gravitational constant	$6.67 \cdot 10^{-11}$	m^3 kg^{-1} s^{-2}
G	Gain	-	dBi
H	Height	-	m
H(t)	Horizontal displacement w.r.t. time	-	m
$\ddot{H}$(t)	Horizontal acceleration w.r.t. time	-	m s^{-2}
h	Altitude	-	m
I	Moment of Inertia	-	kg m^4
$I_{r,Earth}$	Radiation intensity Earth	237	W/m^2
$I_{r,Moon}$	Radiation intensity Moon	304.5	W/m^2
I	Impulse	-	s
i	Inclination	-	°
J_P	Intensity factor	-	-
k	Boltzmann constant	$1.381 \cdot 10^{-23}$	m^2 kg s^{-2} K^{-1}
k	Thermal conductivity	-	K·m/W
k	Spring constant	-	N m^{-1}
k	Lift parameter	-	kg m^{-1}
L	Length	-	m
L	Loss	-	dB
L_d	Fuel cell degradation	-	-
M	Molar mass	-	grams mol^{-1}
M	Mass	-	kg
N	Unit amount	-	-
N	Avogadro's number	$6.022 \cdot 10^{23}$	-
N_0	White signal noise	-	dB
n	Load transmission efficiency	-	%
n	Number	-	-
P_{moon}	Moon perigee	$362570 \cdot 10^8$	m
P_{sun}	Power output of the Sun	$3.856 \cdot 10^{26}$	W
P	Power	-	W
P	Normal force	-	N
P	Period	-	s
p	Pressure	-	Pa
p(t)	Dynamic load	-	N
Q	Heat flow	-	J/s
Q	Internally dissipated power	-	W
R	Orbit position	-	m
R	Data Rate	-	bps
R	Radius	-	m
R_{Earth}	Radius Earth	$6.378 \cdot 10^6$	m
R_{moon}	Radius Moon	$1.737 \cdot 10^6$	m
r	Radial distance between two points/bodies	-	m
S	Learning curve slope	-	-
S	Free space path length	-	km
S	Horizon distance	-	m
S	Surface area	-	m^2
s	Thickness of conducting material	-	m
s	Displacement	-	m

Roman Symbols

Symbol	Name	Value	Unit
T	Temperature	-	K
T	Time		sec
T	Thrust	-	N
T_{ES}	Period Earth-Sun	365.26·24·60·60	s
T_{moon}	Blackbody temperature Moon	270.7	K
T_s	System Noise Temperature	-	dB-K
t	Time	-	s
t	Thickness	-	m
t	Total pulse duration	-	s
V	Velocity	-	m s^{-1}
$\bar{x}, \bar{y}, \bar{z}$	Location centre of gravity X-, Y-, Z- -axis	-	m
x(t)	Displacement	-	m
$\dot{x}$(t)	Velocity	-	m s^{-1}
$\ddot{x}$(t)	Acceleration	-	m s^{-2}

Greek Symbols

Symbol	Name	Value	Unit
α	Specific mass	-	kg kW^{-1}
α	Absorptivity	-	-
β	Initial out-of-plane thrust angle	-	$^\circ$
Δ	Delta/Change	-	-
ϵ	Emissivity	-	-
η	Efficiency	-	-
Θ	True anomaly	-	$^\circ$
$\ddot{\Theta}$	Angular acceleration	-	rad s^{-2}
θ	Half-power Beam Width	-	arcsec
θ	Transmit Antenna Beamwidth	-	rad
λ	Wavelength	-	m
λ	Half-angle between shadowing body and shadowed body	-	$^\circ$
μ	Gravitational parameter	-	$m^3 s^{-2}$
μ_{earth}	Gravitational parameter Earth	398600	m^3 s^{-2}
μ_{moon}	Gravitational parameter Moon	4902	m^3 s^{-2}
ρ	Density	-	kg m^{-3}
σ	Stress	-	Pa
τ	Shear flow	-	Pa
Φ	Velocity angle w.r.t. x-axis	-	$^\circ$
ω	Angular velocity	-	rad s^{-1}
Ω	Ascending node location	-	$^\circ$

Subscripts

Subscript	Name
0	Initial, or sea level
avg	Average
Bat	Battery
b	Bit
c	Circular
cold	Relative to low temperatures
D	Drag
Damp	Damping
dry	Excluding propellants
Earth	Relative to Earth
ES	Relative to Earth and Sun
e	Relative to an eclipse
f	Final
i	Initial, or inclination
L	Lift

Subscript	Name
Moon	Relative to Moon
n	Natural
p	Perigee
path	Relative to patch point
pt	Payload Tank
req	Required
rev	Revolutions
SOI	Relative to SOI
s	Structural
sp	Specific
wet	Including propellants
x	Relative to x-axis
y	Relative to Y-axis, or year
yield	Relative to yielding
z	Relative to Z-axis

1. Introduction

The world will run out of fossil fuels, yielding a challenge to be faced by humanity: finding alternative energy sources. Among others, nuclear fusion is considered as a feasible option for an alternative energy source. World leading countries like the USA, Russia and China are considering to mine lunar He-3 for fusion, providing a clean alternative energy source.

The European Space Agency (ESA) asked us, a group of ten aerospace students, to evaluate the possibilities of lunar He-3 mining. The purpose of this project is to perform a feasibility study on an end-to-end lunar He-3 mining mission [1]. Before this final report, baseline work has been performed to establish the top-level requirements. A market analysis has been performed and Systems Engineering tools were defined. The work has been documented in the Baseline Report.

After the review of the baseline report, follow-up work has been performed on different design, operation and technological concepts, their respective risks and technical research budgets and contingency, along with the trade-off criteria and their rationales, multiple design trade-offs and a resulting proposed concept. This has been documented in the Midterm Report.

In this report, the final report, the proposed concept will be elaborated. The entire end-to-end mission will be designed conceptually and the transportation from lunar surface to Low Earth Orbit (LEO) will be designed in full engineering detail. First, the mission will be described in Ch. 2. It includes the mission need statement, top-level requirements, a market analysis and a sustainable development strategy. In Ch. 3 additional information will be provided, for example on He-3 and space law. In Ch. 4 every part of the end-to-end mission will be designed in conceptual level. After that, the focus will be on the transportation of He-3. Both the Continuous-Thrust Transfer Vehicle (CTTV) and the Lunar Surface Ascent Module (LSAM) will be designed in full engineering detail in Ch. 5 and 6 respectively. In Ch. 7 the mission will be evaluated and in Ch. 8 conclusions will be drawn on the feasibility of the mission. In Ch. 9 it will be addressed what future work is desired until the mission can start. Finally, in Ch. 10 conclusions will be drawn on the entire feasibility study and recommendations are given.

2. Mission Description

This chapter will elaborate on the mission description. First the goal of the project will be stated in the Mission Need Statement (MNS). Then the top-level requirements will be listed, the mission functions (including the Functional Flow Block Diagram (FFBD) and the Functional Breakdown Structure (FBS)) will be discussed, the work of the preliminary design phase will be discussed, a market analysis will be performed and a sustainable development strategy will be presented.

2.1 Mission Need Statement

According to the US Department of Defense, a Mission Need Statement (MNS) is a statement which defines the capability needs a program must satisfy, through a combination of both material and non-material solutions, to solve a mission deficiency [2]. Using this definition, along with the given Project Objective Statement (POS), the following MNS was derived:

"Due to an increasing demand for alternative energy sources, ESA will provide and assess the feasibility of an end-to-end system for lunar He-3 mining, used for nuclear fusion to supply 10% of the global energy demand in 2040."

2.2 Top Level Requirements

The aim of this project is the following: 'Perform a feasibility study on the use of He-3 and fusion to generate power for global usage. In particular, focus on a lunar mining and transportation system.' The top-level requirements, which form the foundation on which the feasibility study is based on, are written out below [1].

- Develop an end-to-end system at conceptual level. It has to be able to retrieve the amount of He-3 needed to supply 10% of the global population's energy demand.
- Develop a transportation system in full engineering detail. This system is to provide transportation in any direction (Earth → Moon and vice versa), and with the capability to fully meet all payload needs.
- Develop mining and extraction system at conceptual level.
- Scalable to satisfy 50% of world's energy demand.
- Capabilities and costs extrapolated to the situation in 2040.

The first top-level requirement applies to all involved mission segments, of which the conceptual design is shown in Ch. 4. It was decided to develop part of the transportation segment in full engineering detail, which were the CTTV and the LSAM. The detailed design is shown in Chs. 5 and 6. The scalability to 50% will be kept in mind for future developments. In Ch. 4 the system requirements will be given. For the two detailed designs the subsystem requirements are addressed in their respective sections.

2.3 Mission Functions

According to the requirements the mission functions can be discussed by providing the Functional Flow Block Diagram (FFBD) and the Functional Breakdown Structure (FBS). The top-level FFBD is given in Fig. 2.1. Note that these FFBDs have been made as an evaluation of the design to give a clear illustration of the mission profile.

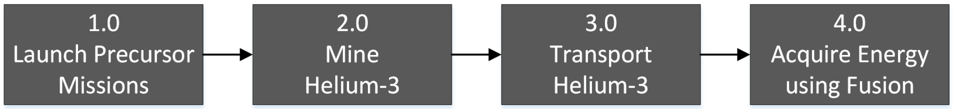


Figure 2.1: Top-level Functional Flow Block Diagram.

As this is a very general function describing the total end-to-end mission, the mining of He-3 (2.0) and the transport of He-3 (3.0) will be more elaborated. Their respective FFBD are illustrated in Figs 2.2 and 2.3.

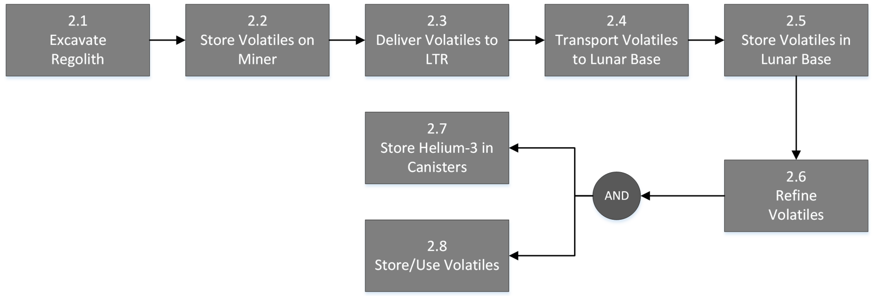


Figure 2.2: Second-level Functional Flow Block Diagram of He-3 mining.

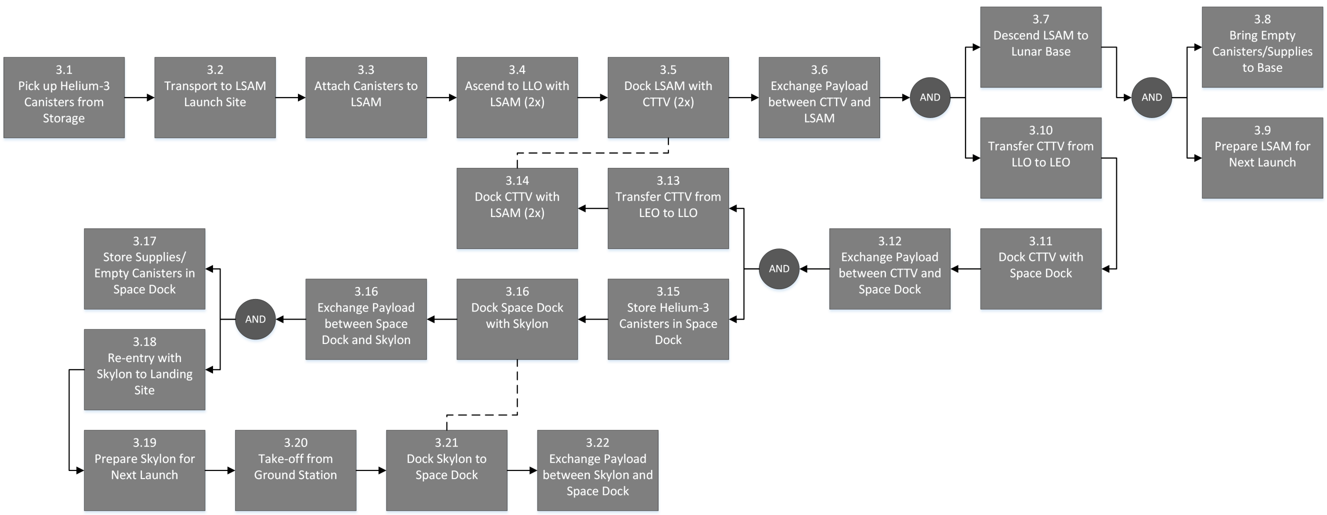


Figure 2.3: Second-level Functional Flow Block Diagram of He-3 transport.

In order to go more in depth regarding the transfer of the CTTV from LEO to LLO (3.13) the 3^{rd} level is given in Fig. 2.4. The same is done for the docking of the LSAM to the CTTV (3.5 & 3.14) in Fig. 2.5 to elaborate more on the docking process.

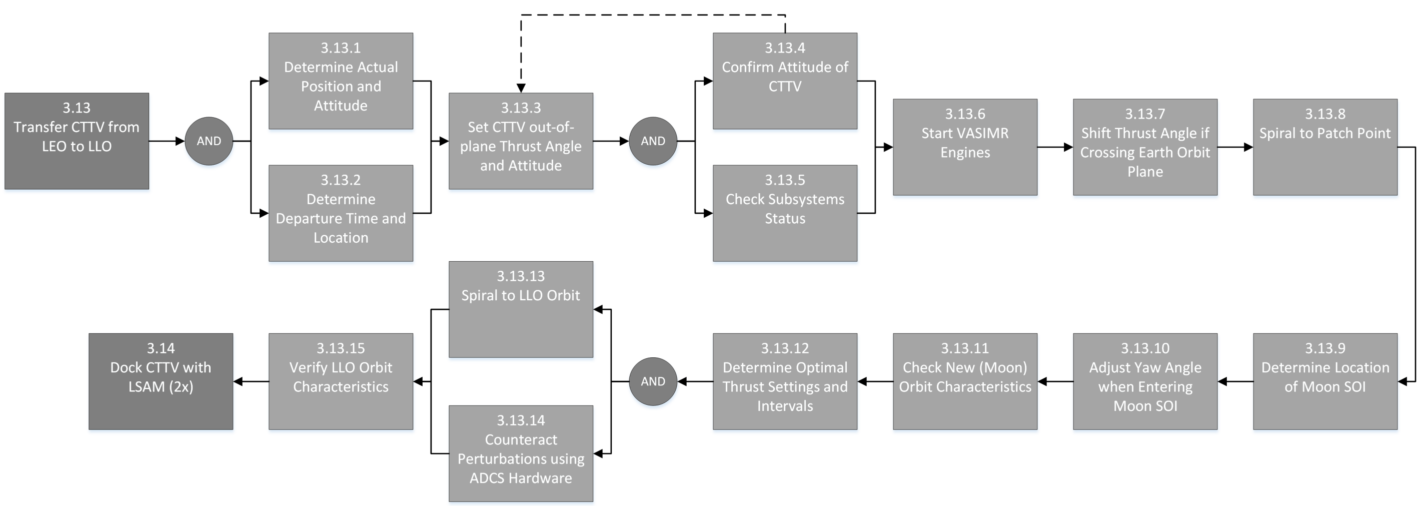


Figure 2.4: Third-level Functional Flow Block Diagram of CTTV transfer.

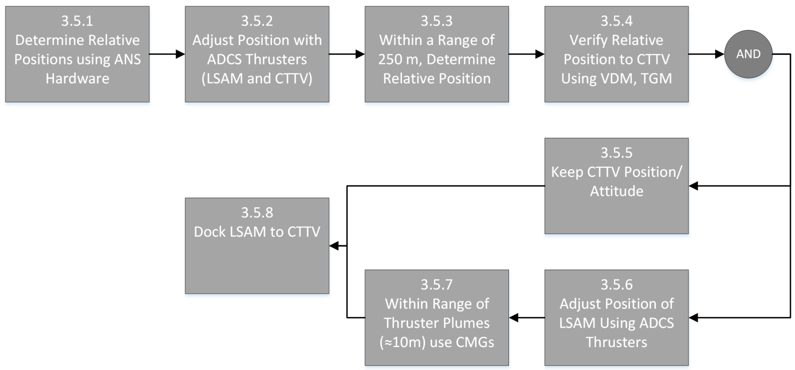


Figure 2.5: Third-level Functional Flow Block Diagram of LSAM docking.

In addition to these FFBDs a FBS can be made, see Fig. 2.6. As this is very similar to the FFBD, the FBS is kept very general. This FBS gives an overview of all the functions in the end-to-end mission, for the detailed explanation of the functions see the FFBS and the Baseline Report [3].

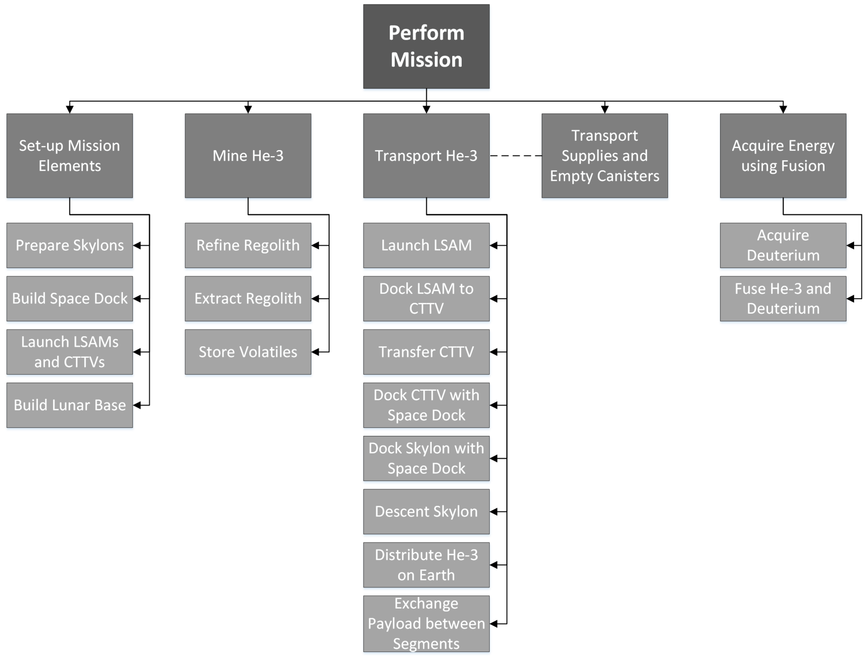


Figure 2.6: General Functional Breakdown Structure of the end-to-end mission.

2.4 Previous Work

The work of the previous reports will be briefly mentioned to get an overview on what has been developed prior to this report and to establish a clear precedent. The steps taken to come up with the final concept and the applied trade-off technique will be discussed.

2.4.1 Extensive Literature Study

Before brainstorming, a literature study had to be performed. To list all of the possibilities that could affect the mission elements, a collaborative spreadsheet was created. All the ideas that were found were documented in their respective category. These categories ranged from lunar operations (different types of excavators, bases, rovers etc.) to different transport options (Earth take-off, low-thrust transports, LSAMs etc.). Part of the proposed elements in the spreadsheet were discarded based on their TRLs (Technology Readiness Levels). These TRL indicate how mature a certain concept or theory is. Once the concepts that were not mature were eliminated, brainstorming could start.

2.4.2 Brainstorm

With a clear overview of the possibilities for each segment of the mission, the group members brainstormed about what the mission potentially could look like. Each project member was asked to individually come up with two possible mission lay-outs, then present them to the group. This resulted in 20 concepts, though most of them showed similarities. From these 20 preliminary concepts, seven concepts were derived and evaluated in the Mid-Term Report [4]. Based on the strong and weak points of the concepts a logical trade-off was performed and three feasible concepts were selected combining the strong elements of the seven concepts.

2.4.3 Three Concepts

The three concepts left were the Helium-3 Shuttle, Helium-3 Solar Sailing and Helium-3 Conventional. A brief description of the three feasible concepts will be provided. The He-3 Shuttle provides LEO access using a Space Plane. From LEO to LLO a CTTV is used, with a VASIMR engine using electromagnetic technology. Transport from LLO to the lunar service was performed by a LSAM. The payload ascends from the Moon with the LSAM and transports back to Earth with the CTTV. On the lunar surface a fully autonomous base is chosen, with no humans present.

The second concept uses a magnetic levitation assisted launch to LEO. The rocket docks to a space station and transfers the payload. Payload from the Moon is put into capsules and re-enter to Earth surface. The transfer from LEO to LL1 is performed by solar sails. From LL1 to the lunar surface, payload is transported by a lunar elevator. Mining operations use multiple bases and miners to refine the He-3.

The conventional concept uses mostly conventional technologies, techniques that have already been proven by the Apollo missions, a direct rocket launch and Hohmann or free-return trajectories. The rocket flies to LLO where it is docked to a lunar space station. The payload gets interchanged and a shuttle brings the payload to lunar surface. A mass driver launches the payload back to Earth using capsules.

For all concepts humans could be involved, but advantages (i.e. physically and mentally flexible, possibility of instant action) should be traded off with disadvantages (i.e. life-support systems, protection from radiation).

2.4.4 Final Trade-off & Final Concept

Once the three concepts were established, they were researched in a more profound manner than happened for the first seven concepts. More detailed estimations of the budgets were made, allowing for a more precise trade-off. This trade-off combined all of the findings into four criteria (scalability, risks, cost and sustainability). Weights of relative importance were assigned and the three concepts were rated on their performance for each of the criteria.

After that, strong and weak points of the concepts could be determined, criteria have been defined and the final concept is proposed. This final concept will be designed in depth in this report. It will consist of six segments that will be designed either on conceptual level or in full engineering detail. A schematic overview of the final concept is illustrated in Fig. 2.7.

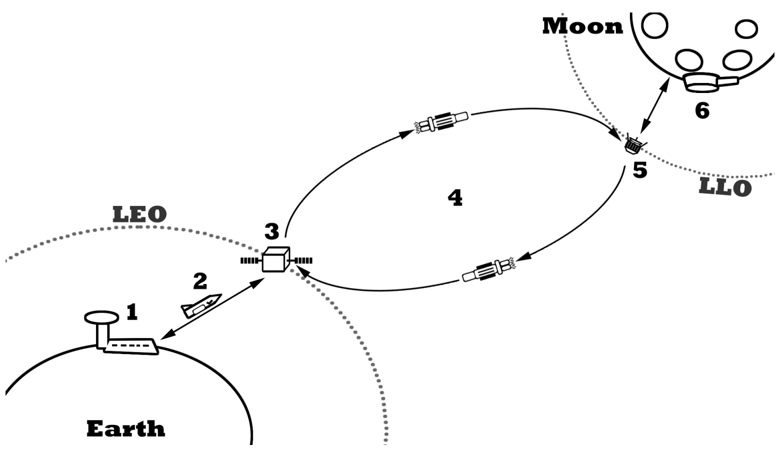


Figure 2.7: Final concept schematic overview.

Segment 1 deals with all the ground operations for the mission, including the launch site, distribution of the He-3 etc. Segment 2 consists of a Space Plane to the Space Dock (3). The CTTV (4) will transfer to LLO. There it interchanges payload with the LSAM (5). The lunar base (6) will be a fixed base with multiple miners. It will process all the regolith and refine the regolith. Segment 4 and 5 will be designed in full detail, the other segments on conceptual level. More detailed information on the final trade-off can be found in the Midterm Report [4].

2.5 Market Analysis

A market analysis must be performed in order to assess the feasibility of a lunar He-3 mining concept. The current and future energy market will be evaluated, the space market will be evaluated and the economic value of He-3 will be evaluated. Also, the expected cost of the mission will be placed into perspective with respect to the current market costs of other energies.

2.5.1 Current Market

To be able to scale the mission, the required amount of He-3 per year to satisfy 10% of the energy need should be determined for 2040 and subsequent years. Using the estimation of different sources, a range of possible demand predictions will be made. This range will be based on the best and worst case estimates for the future. The values obtained from the research done by ExxonMobil and Shell, consists of predicted values up to the year 2050 [5][6]. Since no predictions have been made about the subsequent years, the numbers have been extrapolated to find an equation that covers these estimations. The different estimations of Exxon and Shell and their extrapolations are shown in Fig. 2.8 [5][6]. Although the energy demand does not have a linear relation in the long run, given the time-span for which this approximation is yielded, it is a plausible prediction for the starting years of the mission [6]. It is assumed that the energy demand will increase at a lower rate in the future due to the stimulation of efficient energy usage and a decreasing rate in global population growth [7].

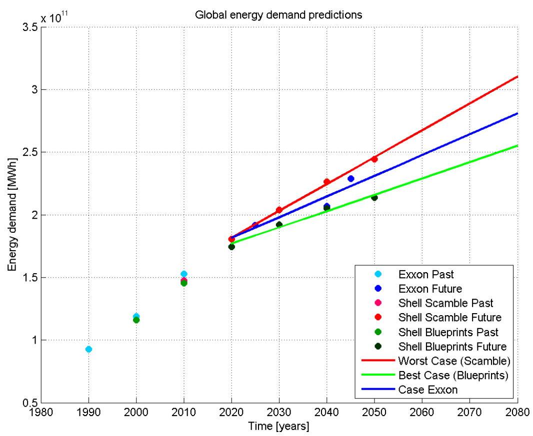


Figure 2.8: Total global annual energy demand.

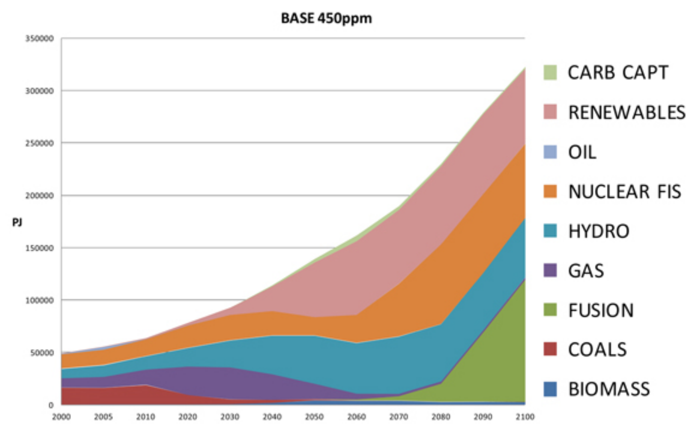


Figure 2.9: Energy allocation prediction up to 2100 [8].

According to Fig. 2.8, the total energy demand in 2040 will most likely be between $2.05 \cdot 10^{11}$ and $2.26 \cdot 10^{11}$ MWh. Alternative energy sources other than He-3 have to be discussed since they might be cheaper or more feasible for the situation in 2040. In the following sections, alternative energy sources will be discussed and the space market is analysed for 2040, to address the expectations concerning for example competition or cost reductions. The European Fusion Development Agreement (EFDA) has conducted research on predictions for energy allocation. Assuming a baseline regulation of 450 ppm for carbon dioxide, the energy prediction is shown in Fig. 2.9.

It has to be noted that energy generation from fusion might not start until 2050. However, fusion has a promising future according to EFDA. It is not guaranteed that fusion technology will be available in 2040, but it could have a large market share in the future. In 2040, the fusion market share is negligible compared to the other energy sources as EFDA expects nuclear fusion to penetrate the market towards the end of the century (Fig. 2.9) [9].

2.5.2 Possible Alternative Energy Sources

Concerning other alternative energy sources, solar and wind energy seem to be the most feasible options that will gain market share as renewable sources of energy [10]. To address their potential, a cost and quantity estimation based on statistical data of resources is made, in order to compare it to the lunar mission for He-3.

2.5.2.1 Solar Energy

Currently, technologies for capturing solar energy are being improved and costs are reducing rapidly. It is expected that costs for solar photovoltaic (PV) technology will drop approximately by two-third before 2030. However, this expectation assumes a significant growth in the solar market and does not include shortages of silicon which might play a role. An alternative to silicon is the use of thin film, which according to Sark *et al* will possibly replace silicon and get a market share of up to 25% of the global energy market [11]. To be able to see if solar energy is a feasible option, a rough cost estimation for 2040, based on several resources, is made. For this estimation, several assumptions are made:

- **A1** Solar intensity on panel = 220 W/m^2, which complies with the solar intensity at proposed sites for large scale solar farms [10].
- **A2** Estimated conversion efficiency is 0.2 (worst case) or 0.3 (best case) [12].
- **A3** The available Sun hours equal 10 hours a day.
- **A4** Fluctuations due to seasons, solar cycles, weather etc. are not taken into account.

Since the situation in 2040 is not known four scenarios are used for the calculations. The first two scenarios (a best and worst case energy demand) assume a high economical growth with low population growth. The world will be more interconnected which makes energy transportation easier [12]. The other two scenarios focus on regional development with low economic growth and high population growth [12]. Using the related values found by De Vries *et al*, a rough cost estimation and solar array area are computed. The results are summarised in Tab. 2.1 [12].

Table 2.1: Rough costs and area estimations for solar energy (10% global energy demand) [10].

	Scenario 1	**Scenario 2**	**Scenario 3**	**Scenario 4**
Total area needed [km^2]	85160	127740	93981	140971
Costs per W [€]	1.05	2.7	1.05	2.7
Total costs in 2040 [T€]	2.432	6.305	2.634	6.958
Costs [€/MWh]	83 - 91	279 - 307	119 - 131	307 - 340

The required area to support 10% of the global energy demand in 2040 will be approximately twice the size of the Netherlands. It must be kept in mind that these areas are only valid with the assumptions made in this section. The costs in Table 2.1 vary depending on the energy demand in 2040 (worst or best case).

The total costs are based on 10% of the energy demand in 2040 and the total costs per Watt [12]. Comparing this with the value found by the International Energy Agency; 2.341 T€ (1€/W multiplied by Watts needed) it can be stated that the investments needed will be in the order of trillions of euros [13]. For the costs per MWh, a range is given because of the two different energy predictions. Please note that the cost predictions do not take the growth of the solar market (and consequently the price reductions) stated before into account. The two major drawbacks of using solar energy are finding locations and transportation of the energy. It will be difficult to find locations with a very large area, close to big cities (where the energy is needed). Next to that, the most efficient locations for solar energy are not close to areas with a dense population. Creating a large transportation system will increase costs, decrease efficiency and will be more difficult to maintain. Also, the Sun is not always available (clouds, night) and a storage system is required (which will decrease overall efficiency).

2.5.2.2 Wind Energy

Similar to solar energy, wind energy is clean, renewable and may be sufficient to supply the entire global energy demand in 2040 [10]. Using statistical data, a rough estimate for the cost of using wind energy to supply 10% of global energy demand in 2040 can be made. The following section will describe forms of wind energy and how a rough cost estimate was performed.

Cost of Wind Energy: In order to estimate the costs of wind energy in 2040, statistical data and online studies have been used. Research performed by the UK Energy Research Centre (UKERC) estimated that by the end of 2030, in the worst case, prices for turbine, foundation, operations, and maintenance (O&M) with respect to 2009 will increase by 10, 20, 25 and 10% respectively [14]. In the best case, costs will be reduced by 40, 30 and 25% for turbine, foundation and O&M respectively. Using the cost per power values in 2009 (172 €/MWh), and the worst (243 €/MWh) and best case (113 €/MWh) values in 2030, the cost in 2040 can be extrapolated by assuming a linear relationship. Using a 5% margin, the worst and best case cost per unit of power in 2040 is 255 €/MWh and 80 €/MWh respectively. For the worst and best case energy demand it yields the total costs are 6.73 and 1.65 T€ respectively.

Table 2.2: Cost wind energy for energy demand 2040 (10% global energy demand).

Global Energy Demand	**Price [€/MWh]**	**Cost [T€]**
Worst Case	80	2.12
	255	6.73
Best Case	80	1.65
	255	5.23

2.5.2.3 Other Alternatives

According to Fronius [15], solar and wind power will be the most probable renewable energy sources. Other possibilities can be considered, but they will act on a smaller scale [15]. Those options are difficult to predict or will not be scalable to the level required for the predicted energy demand. There might be a breakthrough in another segment of renewable energy (i.e. hydro power) but this can not be foreseen for the period up until 2040.

2.5.2.4 Conclusion on Alternatives

The alternative energy sources discussed are feasible options, but they will most likely not be able to support the global energy demand completely. According to predictions, nuclear fusion (see Fig. 2.9) might become the main energy source in the next century [16]. The most probable solution for a world without the use of fossil fuels would be a combination of wind, solar, fusion and possible other renewable energy sources.

2.5.3 Space Market Analysis

In the coming centuries, the space market is likely to change to a more commercial, mature market [17]. The areas in which it develops are broad with many different applications (i.e. navigation, tourism etc.). Commercial companies are growing and are becoming more competitive. In addition, multiple countries such as China, Russia, Britain, Europe, and the USA are aiming to explore the Moon in the coming years [16][18][19]. Due to this, and due to the increasing interest in exploring the Moon and its resources, the costs for space travel are expected to decrease significantly. Innovative ideas, new technologies and optimisation of current processes are expected to be the main driver to lower costs. The following section will outline what major changes in the space market can be expected.

In the recent years, experiencing zero gravity in space has become possible for civilians with *Virgin Galactic* and other companies [20]. Private space companies such as *SpaceX*, *Deep Space Industries*, *Planetary Resources* and *XCOR Aerospace* are competing to operate in space at low costs [21]. The space sector has become highly competitive, leading to investments for research and development of new techniques and methods to optimise space transport and operation. A few upcoming technologies:

- **Vehicle Design:** Many private space agencies are currently working on new vehicle designs for cargo transportation (i.e. Dragon), exit and entry capsules (i.e. KANKOH-MARU), rockets, and shuttles (i.e. Spacebus and Kelly Eclipse, Skylon) [22]. These new designs aim to increase efficiency, minimise costs and reduce the current limitations with respect to space-related missions in the future.
- **New Technology:** New technologies will allow new concepts to arise. An example is the development of a fusion and anti-matter rocket [23][24]. If fusion were feasible in 2040, using a fusion rocket for propulsion would be a possible alternative to conventional propulsion systems.
- **Earth Orbit Access:** What makes or breaks the space market and space exploration of the future is easy and cheap access to LEO, or even Earth Orbit (EO) in general. Once a spacecraft escapes the Earth's "gravity well" and atmosphere, it can begin highly specialised missions. Separating the ascend-to-EO phase from the rest of the mission gives a new perspective to space missions. Numerous concepts to reach EO are under development, from Skylon to the famous Space Elevator. These concepts are still at an early stage, with no guarantee that they will be available in 2040.

2.5.4 Economic Value of Helium-3

As will be determined in Sec. 3.1.2.1, the energy provided per kilogram of He-3 is calculated to be $1.635 \cdot 10^5$ MWh/kg for second generation fusion, assuming a 100% efficiency. To put a value on one kilogram of He-3 the price per MWh can be determined using statistics from other energy sources. For this the values determined in Sec. 2.5.2 and predictions from Exxon and Enerstrat a range for the price (€/MWh) for renewable energy in 2040 can be determined [5][25]. These ranges are listed in Tab. 2.3.

Table 2.3: Prediction of the energy price for renewables in 2040 [5][25].

Source	Range [€/MWh]
Wind/solar energy	80 - 350
Exxon	46 - 115
Enerstrat	30.4 - 68.4

As these are price ranges for the renewable energies, which are a competitive energy source for fusion, the price per MWh of He-3 can be estimated. In the worst case, it is estimated that the He-3 price per MWh will be 30.4 €, the target value to be competitive with fossil fuels. For the 10% global energy demand it is estimated that annually 200 tons of He-3 is needed, see Sec. 3.1.4. This will indicate that yearly minimum revenue (income) is approximately 600-700 B€.

3. Additional Information

Before getting into the design of the mission, some additional information is given. Starting with information about He-3, followed by an elaboration on the space laws which can influence our mission. In Sec. 3.3 the radiation effects on the crew will be explained. The cost conversion method used to get a consistent number for the cost is discussed thereafter. At the end there will be a general explanation of the Verification and Validation (V&V) procedures.

3.1 Helium-3

Scientists have been doing research on the use of He-3 as an energy source since the early 1970s. The Apollo missions brought lunar soil back to Earth, and it was discovered that the lunar soil contains a significant amount of He-3. This section will discuss some additional information on He-3. First, it is discussed what He-3 actually is and where it can be found. Secondly, it is described how to obtain energy from He-3 through fusion processes. Lastly, the current development on these fusion processes is outlined, and the annual demand is briefly covered.

3.1.1 Sources of Helium-3

He-3 is an isotope of Helium. It contains two protons and one neutron. This isotope is mostly produced by the Sun. The Sun fuses Hydrogen atoms together and forms normal Helium, but about one in every 10000 fused atoms is missing a neutron, resulting in He-3 [26]. This process has been ongoing for billions of years.

The He-3 that is generated by the Sun is transported throughout the galaxy. However, due to the Earth's magnetic field and atmosphere, these particles are almost completely deflected. He-3 is available on Earth in small amounts, but not readily accessible as it is stored in the Earth crust. He-3 is currently extracted from natural gas. The Moon only has a very weak magnetic field, and no atmosphere. Therefore, He-3 has been steadily deposited in the lunar regolith for the past billion years, as confirmed by the Apollo missions. However, the concentrations are very low, as can be seen in Tab. 3.1. He-3 can be extracted from the regolith through heat treatment, while other volatiles are released in the process that can be used to sustain human life.

Table 3.1: Estimation of probable He-3 reserves in the lunar regolith [27]. Note: parts per billion (ppb) is a mass fraction [28]. For example: 1 ppb of He-3 indicates 1 kg of He-3 per 10^9 kg regolith.

Category	**TiO_2 [%]**	**Area S_{TiO_2} [km^2]**	**He-3 abundance [ppb]**	**Regolith depth [m]**	**Density [kg/m^3]**	**He-3 reserves [tons]**	**He-3 [%]**
I	5-10	$487{\cdot}10^3$	15.1	4.4	1900	$615{\cdot}10^2$	2
II	3-5	$152{\cdot}10^4$	8	4.8	1900	$111{\cdot}10^3$	4
III	1-3	$159{\cdot}10^4$	5.7	8.1	2000	$146{\cdot}10^3$	6
IV	0-1	$343{\cdot}10^5$	3.1	10.1	2000	$215{\cdot}10^4$	87
Sum						$244{\cdot}10^4$	100

Other Sources: While the Moon might seem the most attainable source of He-3 at first glance, both in matter of distance and experience, there are other potential sources of He-3 in the solar system. These other sources include bodies similar to the Moon, like Jupiter's moons, or gas giants, who still contain large amounts of He-3 from the formation of the galaxy. However, reaching these far-out bodies is far more complex and difficult than a consistent mission to the Moon already is. Other possible sources include Near Earth Objects (NEOs) and direct extraction from the solar wind. Both are dismissed at this time, as NEOs are far too small in surface and have a too low He-3 concentration to warrant an entire mission infrastructure on it. Direct extraction from the solar wind would take far too long, and be a nearly impossible endeavour because of the required size of the "solar net", and many other complications. This option would not be feasible by 2040.

3.1.2 Fusion Processes

Obtaining He-3 is one aspect of the mission objective, using it to produce energy is another. As mentioned, He-3 can be used for nuclear fusion processes, similar to the processes taking place within stars. Multiple alternative reactions are available that produce energy from fusion. There are three generations of fusion processes, only the second- and third-generation fusion will be discussed in the following sections.

Second-generation Fusion: The second-generation fusion fuels, Deuterium and He-3, are the most interesting for this mission. The reaction is illustrated in Fig. 3.2. This reaction will result in an alpha particle (He-4) and a high energy proton [29]. It should be noted that the temperatures required for this reaction are about five times higher than the temperatures required for first-generation fusion (D-D and D-T), as can be seen in Fig. 3.1. The issue of the D-He3 reaction is that it is possible (and, at these temperatures, very probable) that D-D reactions will occur, which will result in neutrons. This indicates that D-He3 fusion is not clean in terms of radiation. However, the neutron flux of D-He3 fusion is only 1% compared to D-T fusion [30].

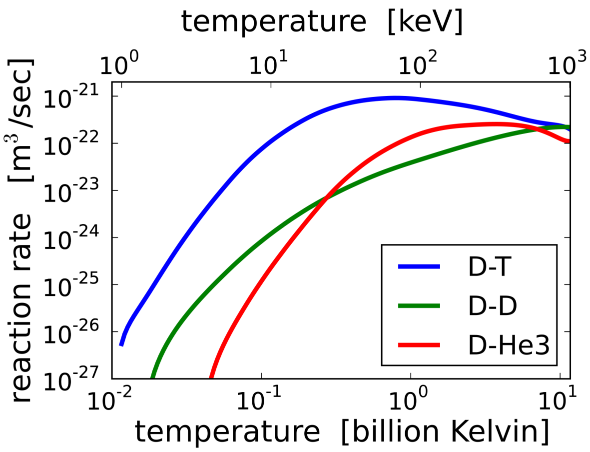


Figure 3.1: Reaction rate vs. temperature [31].

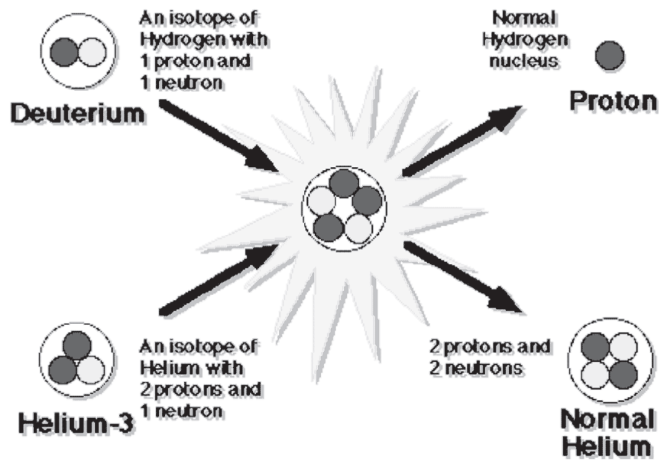


Figure 3.2: Illustration of D-He3 fusion [32].

Third-generation Fusion: The third-generation fusion fuels are He-3 and Boron. The first possibility is a clean reaction with only He-3 (He3-He3). This reaction results in an alpha particle, two protons and no neutrons [31]. This is the cleanest fusion reaction because the produced Helium is non-polluting and has no radioactive by-product. The required temperature to obtain a sufficient reaction rate is is about 200 keV [30] (corresponding to 2.3 GK), due to the opposing +2 charge of the two He-3 molecules. Fusion temperatures in Kelvin are obtained by dividing the energy in Joule by the Boltzmann constant, k.

The alternative third-generation fusion fuel is Boron, which has the same advantages compared to He3-He3. Boron and a Hydrogen-1 atom (i.e. a proton) fuse together, a reaction abbreviated as p-B11, resulting in three Helium atoms [33]. An important aspect is that p-B11 fusion is already further in development than D-He3 fusion. It must be noted that p-B11 has no radioactive by-products, and above all it is readily available on Earth itself. The disadvantage is that to achieve the maximum and most optimal reaction rate, the temperature should be around 600 keV ($\approx 10^{15}$ J$\approx$6.95 GK) [34]. This is about 18 times higher than the temperature needed for D-T fusion and 12 times higher than the temperature needed for D-He3.

3.1.2.1 Summary

Table 3.2: Overview of the fusion processes [30][33][29].

	1st-gen		**2nd-gen**	**3rd-gen**	
Reactants	D + D	D + T	D + He3	He3 + He3	p + B11
Product(s)	n + He-3 or p + T	n + He-4	p + He-4	2·p + He-4	3·He-4
Energy [MeV]	3.27 or 4.03	17.59	18.35	12.86	8.68
Neutron flux [n/MeV]	11	5	0.04-0.2	0	0
Temperature needed [keV]	34.5	3.9	50	≥200	600

Using the energy values for every generation, the energy in J/kg can be determined using Eq. 3.1. Here, N is the Avogrado's number of $6.022 \cdot 10^{23}$ and M is the molar mass. The value for eV is converted to Joule by multiplying it by $1.6 \cdot 10^{-19}$, and a factor 1000 is used to convert from grams to kilograms. An example calculation for second-generation fusion is given in Eq. 3.2. Here the molar mass (M) is 3 g/mol and the energy is $18.35 \cdot 10^6$ eV. The result is $1.635 \cdot 10^5$ MWh/kg of He-3. It must be noted that this assumes a 100% efficiency in energy conversion.

$$Joule\ per\ kg = \frac{(N)}{(M) \cdot (1000)} \cdot (eV) \cdot (1.6 \cdot 10^{-19}) \tag{3.1}$$

$$\frac{(6.022 \cdot 10^{23})}{(3) \cdot (1000)} \cdot (18.35 \cdot 10^6) \cdot (1.60 \cdot 10^{-19}) = 5.90 \cdot 10^{14}\ J/kg = 1.637 \cdot 10^5\ MWh/kg \tag{3.2}$$

3.1.3 Current Development

Several research projects have been called into life ever since fusion has been discovered as a possible solution for energy production. The most important project would be the International Thermonuclear Experimental Reactor (ITER) in Southern France, an enormous facility that would make fusion feasible by 2040, though it has encountered a significant delay. Other initiatives include the "Wiffleball" running on p-B11 fusion, a Tritium-Deuterium reactor developed by Skunkworks and other small start-ups such as Tri Alpha Energy and Lawrenceville Plasma Physics [35][36][37].

3.1.4 Annual Helium-3 Demand

A minimum value of required He-3 was previously established to provide 10% of the global energy demand by 2040. This was found to be between 120 and 400 tons, depending on the energy demand scenario, and the kind of fusion used. For now, 200 tons of annual He-3 demand is determined by Eq. 3.3. This is based upon three assumptions:

1. Worst case energy demand in 2040 of $2.25{\cdot}10^{11}$ MWh is assumed.
2. Second-generation (D-He) fusion is performed, requiring much less He-3 per unit of energy than third-generation fusion. The most important downside to this reaction is the fact that short-lived radiation is released. This fusion process yields $1.637{\cdot}10^{5}$ MWh/kg of He-3.
3. A total efficiency of the end-to-end system of about 67.5% is assumed.

$$(10\%) \cdot (2.25 \cdot 10^{11}) \cdot \frac{1}{(1.635 \cdot 10^{5})} \cdot \frac{1}{(67.5\%)} \approx 200\, tons \tag{3.3}$$

3.2 Space Law

In the Mid-Term Report the juridical aspects of this entire mission have already been discussed [4]. The main points of this discussion and the conclusion will be briefly mentioned in this section.

There is a large amount of discussion going on about who is actually allowed to "own the Moon". The answer to this is simple, nobody does [38]. Although many people and organisations claim to have ownership right, this is far from true. To have ownership of something, there needs to be some higher authority to recognise this. This simply means that as long as countries do not have sovereignty over extraterrestrial areas, nobody can own bits of outer space. This brings up the question who is actually allowed to exploit the Moon. There are several international UN treaties, some of which are widely ratified, some of which are not (Moon Treaty) [39][40]. The most leading treaty is the 1967 Outer Space Treaty (OST), which mainly includes points to ensure that outer space resources (like the Moon) are preserved and made available to all mankind. In addition to that, it is not just treaties which establish international law. Also, when countries act in a certain way, for example with respect to satellites freely flying over their territory, it is said to be just as legally binding as a treaty. This concept is called jurisprudence.

With respect to outer space resources having to be made available to everybody, one can take the Intelsat organisation to elaborate on this. It was the first project actually making use of the benefit of geostationary orbits. Although only a few nations financed it, and could have claimed the right to use this for their own, as they did not ratify the Moon Treaty. However, they made it readily available for everybody who was willing to pay a reasonable price. In other words, it is not required to offer resources for free, as long as they are offered to all countries. This could be done in the same way concerning mining He-3.

In addition to all that has been stated above, the OST and Moon Treaty include an article which allows for revision or addition of clauses if the current global situation requires so, for example when outer space resources become more reachable. Assuming that the He-3 will be used to benefit mankind in general, there will be no issues from a juridical point of view for this feasibility study.

3.3 Radiation and Crew Replacement

Radiation is an important aspect to consider for manned missions. The radiation exposure determines the time interval between crew replacement missions. Tab. 3.3 contains the overall results of this section. Current radiation limits are only defined for LEO activities, in principle, NASA generally operates under the ALARA ("As Low As Reasonably Achievable") principle.

Galactic Cosmic Rays (GCR) are the most prominent radiation in outer space and constantly flood the solar system [41]. In the short run, limited exposure to GCR has little health risks, however, accumulating exposure over lifetime leads to later radiation-induced illnesses such as cancer. Solar Proton Events (SPE), also referred to as Solar Particle Events, are another source of radiation. Their occurrence is not well understood yet, but large SPEs appear to be limited to one or two occurrence per eleven-year solar cycle [42][43]. Shielding from SPE, for example using water tanks in the walls, is effective [43]. The last significant source of radiation is caused by GCR hitting the surface, which causes neutron back-scatter. This neutron albedo exposes astronauts on the surface to increased radiation as compared to free space [44].

For now, no consensus has been reached as to what an acceptable level of radiation exposure for extra-LEO activities is. For a mission architecture involving a habitat, water-tank-shielded rovers, and 240 hours of EVA activity, a 70 day surface mission would create an effective dose of 57.1 mSv [43]. This accounts for GCR and neutron albedo, but not for SPEs. According to Adamczyk et al, the design of the Orion Crew Exploration Vehicle (CEV) has a radiation exposure limit of 150 mSv as a reference[43]. This limit would be reached after 184 days (excluding transfer time). For LEO activities, a 3% Risk of Exposure Induced Death (REID) from fatal cancer constitutes NASA's career exposure limit [45]. A 30-year old woman (worst case) has a career limit of 470 mSv, a 55-year old man (best case) has a limit of 1470 mSv [45]. ESA dose limit is 1000 mSv for the total career (no gender or age differences considered). Note that blood-forming organs (BFO), eyes, and skin have lower limits (between 250 and 1500 mSv for 30-day missions, and between 500 and 3000 mSv for year-long missions) [45]. BFOs set the limit, to 500 mSv in one year [43]. The maximum tolerated exposure in one month is 250 mSv.

No clear conclusion can be drawn with respect to radiation and mission duration. However, astronauts should reach the career limit only in multiple subsequent missions. A mission duration of 180 days appears reasonable[43]. More research and measurements must be conducted. Mitigation of radiation, especially of SPEs, should be developed (such as burying a structure under regolith [41]).

Table 3.3: Radiation limits summary.

Type	**Exposure [mSv]**	**Period [days]**
30 days on lunar surface	24.6	30
1 year on lunar surface	299.3	365
Blood-forming organ limit 1	250	30
Blood-forming organ limit 2	500	365
NASA (30y women)	470	576
NASA (55y men)	1470	1802
ESA limit	1000	1219

Note: exposure lunar surface, 57.1 mSv/70 days = 0.82 mSv/day

3.4 Converting Cost

Research and calculations done to estimate cost often yield values in different currencies and base years. To make such values comparable, it is important to define a general procedure for converting cost to the same fiscal year. The procedure consists of two steps; converting the value to the present value (fiscal year 2012) by taking inflation into account, and converting the present value to euros. For the procedure a spreadsheet was developed. The first step required a literature study on inflation rates in Europe and the United States. Yearly inflation rates were found from 1960 onwards (US) and 1997 onwards (Europe) [46][47]. Yearly averages are computed ranging from -2.0% to 14.5% and using these inflation rates, a value in a certain year can be converted to the present value in 2012, which is 2.22% for Europe (step 1). The value, given that it is not in euros, then has to be converted with an exchange rate (step 2). Over the fiscal year 2012 the average exchange rate is found to be $1.2848 for 1 € [48]. This rate will be used for further calculations in the report. The same method was applied for the conversion of pounds to euro, with an average exchange rate found of £0.811 for 1 €. Both steps were combined in an Excel file, making sure all the cost can be documented with the same base to make them comparable.

Since the mission operations for 10% of the energy demand are on a very large scale, for the production costs of different systems a learning curve should be applied to account for the improvement in productivity as the number of units increase. The approach from Space Mission Analysis and Design (SMAD) [49] has been used, with which the total production costs for N units can be given by the equations below.

$$Production\ cost = TFU \times N^B \tag{3.4}$$

with

$$B \equiv 1 - \frac{\ln((100\%)/S)}{\ln 2} \tag{3.5}$$

Here TFU denotes the theoretical first unit cost and S is the learning curve slope in percent. The latter shows the reduction in percentage of cost for every time the amount of produced units is doubled. This variable S depends on the number of units produced. For the production of <10 units, $S = 95\%$ is recommended. With the production of between 10 and 50 units, $S = 90\%$. When more than 50 units are produced, $S = 85\%$. This approach has been applied to the cost estimations made on the mission. All the cost estimates in this report make use of the conversion procedure and rates stated above.

3.5 Verification and Validation Procedures

In this section, the general procedure for Verification and Validation (V&V) of a numerical and/or computer model used during the design phase is outlined. The V&V results of every model are discussed in the "Verification and Validation" section of the respective chapter or section.

Both the lunar and ground segment are designed at a conceptual level. It is important that one knows whether depth and accuracy of the claims made for the concepts suffice for further and more detailed design stages and analysis. In order to serve the appropriate level of detail along the design process, the results will be updated. In particular, this is done to remedy the low amount of concrete numbers and parameters which are part of the concepts. For this final phase of the project, the V&V will be performed very generally on the lunar and ground segment, with the focus on verification. For the transport segment, which is to be designed in full engineering detail, the validation will be performed if concrete numbers and parameters are available.

To prevent confusion between the two terms, a quick review of the meaning of both terms is given first: *"Verification and validation is the process of ensuring that a design meets requirements. Verification confirms that products properly reflect the requirements specified for them, ensuring that it was correctly built. Validation confirms that the product, as provided, will fulfil its intended use, ensuring that the right thing was built. Typical inputs are requirements and design representations such as 2D/3D mechanical CAD[1] drawings and models, electrical schematics, and software. Typical outputs are a either a determination of whether the design component or system met requirements, descriptions of failure modes, summarised test results, and recommendations for design improvement."* [50]

Firstly, all used numerical and/or computational models for the design of the system concepts will need to be verified for their accuracy, credibility, consistency and usability. This will be mainly used for the transportation phase and launch/re-entry phase. For example, an astrodynamic model for the orbital characteristics of the transportation system will have to be verified and validated for the above-mentioned criteria. A model which is not credible enough or usable enough for the chosen concept (i.e. overly simplified, wrong assumptions) will result in a non-valid concept. Secondly, several general steps will need to be taken [51]:

- Identify testing methods by which the requirements are validated.
- Obtain facilities and resources to develop models, run simulations. (here the group only considers simulation models which may be developed in MatLab)
- Prepare test cases and test configurations that translate requirements into loads and constraints that can be digitally and/or physically measured. (e.g. set up a model transfer segment for the concept, with its requirements)
- Execute tests (digital or physical) across all functional domains.
- Document results, reporting on the fulfilment of requirements and specifications, and recommendations for failed conditions.

While these steps are more oriented towards product validation, for this project, validation of the system as a whole will be applied. This means that real-time testing is ruled out completely. Focus is put on validating the final concept. The validating methods which apply to the proposed system/concept have been identified. These are:

- Analysis.
- Simulation (Astrodynamics).
- Similarity with previous/existing missions (re entry/launch phase, mining activities, etc.).
- Review of design (documentation).

The basis for the validation process consists of the full set of requirements for the specific segments together with supplementary design tools and representations (end-to-end system sketch, orbit prediction/calculation scheme, accurate launch/re-entry phase scheme, extraction and process scheme, and other). Using the above steps, these will be used to assess the validity of the concept and its segments. As the requirements on lower levels will become more refined/defined, the validation process will show whether they are met. If validation reveals flaws, there will be an obvious need of redesigning system elements.

[1] CAD: Computer Aided Design

4. Conceptual Designs

As the first of five top-level requirements states and end-to-end mission has to be designed on a conceptual level, this chapter will show designs of several mission segments to this extent. Firstly, the system requirements will be listed. Consequently, the conceptual design is started with the payload design. After this, a conceptual design of the general communications of the total mission will be shown. In addition, there will be presented some designs of the transportation segment, the LEO access vehicle, the Space Dock, the CTTV and the LSAM. The final four sections will then contain the conceptual design of the lunar base, followed by the design of the docking systems between transportation segments. The additional precursor and manned missions are discussed thereafter. The latter two are to be designed as a side track to the operational segments.

4.1 System Requirements

In Tab. 4.1 an overview is given of all the system requirements that were identified for the different mission segments treated in this chapter. The requirements for the LEO dock, CTTV and LSAM will also be used as baseline for their respective detailed design.

Table 4.1: System requirements for the conceptual designs.

Requirement ID	**Requirement description**
Payload-1	The design shall comply with all payload constraints set by the other mission elements.
Payload-2	The amount of He-3 carried per unit of structural mass shall be maximised.
Payload-3	The structure shall be able to withstand hostile space conditions.
Payload-4	The structure shall be equipped with a system compatible with other docking systems.
SpaceP-1	The system shall allow for a high frequency of flights between Earth and Space Dock.
SpaceP-2	The system shall allow a payload capacity of at least 12 tons both ways.
SpaceP-3	The system shall have a service ceiling of at least 525 km.
SpaceP-4	The system shall be able to attach to the Space Dock.
SpaceP-5	The system shall allow for horizontal take-off.
SpDock-1	The Space Dock shall allow for exchange of payloads between Space Plane and CTTV.
SpDock-2	The Space Dock shall facilitate maintenance and refuelling of the CTTV.
SpDock-3	The Space Dock shall allow for manned support.
SpDock-4	The Space Dock shall allow for storage of empty and full He-3 canisters.
SpDock-5	The Space Dock shall be able to keep track of the location of the transfer vehicles (CTTV and Skylon).
SpDock-6	The Space Dock shall be able to maintain orbit and prevent collisions with orbital debris or satellites.
SpDock-7	The Space Dock shall have enough propellant for an end-of-life boost.
SpDock-8	The Space Dock shall be able to communicate with the ground segment.
CTTV-1	The CTTV shall be able to autonomously pick up cargo in lunar orbit.
CTTV-2	The CTTV shall be able to autonomously dock with the Space Dock.
CTTV-3	The CTTVs shall be able to transport 1000 tons of cargo to Earth orbit per year.
CTTV-4	The CTTVs shall be able to transport 800-1000 tons of cargo to lunar orbit per year.
CTTV-5	The technology for the CTTV shall be ready by 2040.
CTTV-6	The CTTV shall not produce any space debris.
LSAM-1	The module shall bring up 12 tons of payload (He-3 including its vessel) per trip.
LSAM-2	The module shall be able to descend again from LLO for reuse.
LSAM-3	The module shall be propelled only on in-situ resources.
LSAM-4	The module shall be able to accommodate for rendezvous manoeuvres with the CTTV.
LSAM-5	The module shall have a 100 m, 95% confidence bound, landing accuracy.
LSAM-6	The module shall be able to perform at least 150 ascent and descent cycles.
LSAM-7	The module shall be able to perform all its tasks autonomously.
LSAM-8	The module shall deliver at least 2.5 m/s^2 of acceleration.
LOps-1	200 tons of He-3 shall be retained over the course of 1 year.
LOps-2	Multiple volatiles shall be mined (He-3, H_2, H_2O, CO, CO_2, N_2, CH_4, He-4).
LOps-3	Mining of rough/inaccessible terrain shall be avoided.
LOps-4	Mining operations will be carried out at all available time.
LOps-5	The base and all machinery shall remain operational during the entire mission duration.
LOps-6	The base shall provide space for machinery and spare parts.
LOps-7	Redundant equipment shall be provided.
LOps-8	Redundant equipment shall be shielded from lunar dust.
LOps-9	Volatile storage and volatile refining facility shall be provided.
LOps-10	Volatiles shall be stored in gaseous form.
LOps-11	Redundant machinery shall be provided to allow continuous operations.

LOps-12	Protective habitat for humans shall be provided.
LOps-13	Crew shall stay no longer than 180 days on the Moon.
LOps-14	Dedicated launch and landing site shall be located near the base.
LOps-15	Launch and landing sites shall not be located near vital elements of the base.
LOps-16	Transportation infrastructure shall provide for coverage of mineable area.
LOps-17	Communications, supervision and control shall be provided between MCC and lunar base.

4.2 Payload Design

In this mission 200 tons for of He-3 will be transported to Earth on an annual basis. He-3 must be stored and transported in every element in this mission (i.e. lunar base, Skylon, Space Dock, CTTV and LSAM). Thus, the design of the Helium canister (payload) is essential. The following section will outline the driving factors influencing the payload design and the method used to come up with the design. Finally, a summary of the He-3 canister is given.

4.2.1 Driving Factors

In order to come up with a proper design it is essential to identify the driving factors and needs. Firstly, per year, 200 tons of He-3 must be transported back to Earth. Secondly, the design must comply with the respective payload constraints (mass and volume) set by all elements. The density of He-3 at room temperature is 0.1650 kg/m^3 [52]. If He-3 were to be transported in this state, a total volume of 1.2 km^3 would be required, thus it is beneficial to cool and pressurise the He-3 to decrease the total volume. Furthermore, during transportation, the payload is directly exposed to the space environment (vacuum and radiation). Therefore, the structure of the canister (payload) must withstand high pressures, extreme temperatures and radiation. For transportation purposes, multiple payloads must be able to assemble and disassemble from each other. The structure must thus contain a coupling mechanism.

4.2.2 Payload Design Method

The process used to determine how the final payload design (Sec. 4.2.5) was chosen is shown below. First, the state of the Helium is determined, followed by the method used to determine the payload size and mass. The payload size and mass depend on the He-3 container size, the space thermal/debris shield and re-liquefier system, which are described below.

As shown on Earth, transporting Helium, among other gasses, is most efficient when in liquid form. Using cryogenics, He-3 can be liquefied, increasing its density from 0.1650 to 75.9 kg/m^3 [52]. This reduces the total volume from $1.2 \cdot 10^6$ to 2600 m^3 [53]. International Cryogenics (IC) and Wessington Cryogenics manufacture containers that can be used for transportation of liquid Helium (He-3 and He-4) [54][55]. The containers are equipped with a vent, withdrawal, secondary relief, road and in-transit relief valve, and a pressure gauge for filling, maintaining and monitoring the He-3 during transportation and loading/unloading.

Both companies published the specifications of their Helium (He-3 and He-4) canisters. Fig. 4.1 shows that there is a linear relationship between the total mass and Helium mass. The figure also shows that Wessington Cryogenics is busy manufacturing large He-3 storage vessels, whereas IC focuses on smaller sized canisters. For the following mission, Skylon's maximum return payload mass of 12 tons is the limiting factor. Since this is a large storage vessel, canisters from Wessington Cryogenics are interpolated to determine the amount of Helium that can be transferred using 10.2 tons, which is Skylon's maximum return payload (12 tons) minus a 15% safety margin (also taken into account additional devices such as a docking system will have to be added). The margin is introduced to account for sub-elements such as a thermal/space debris shield. If aforementioned relationship is extrapolated, one can deviate that a Skylon is capable of carrying 2.28 tons of He-3 meaning the structure of the canister is 7.92 tons.

The interpolation is also shown in Fig. 4.1. A ±10% margin was used to determine the amount of He-3 the canister with a total mass of 10.2 tons (including Helium) could carry. The HC-10k deviates far from the rest because of additional features like a staircase to access the top of the tank.

If volume is extrapolated the same way as the structure to payload ratio was found, one yields a required volume of 54.4 m^3. Taking into account that it is a cylinder with semi-ellipsoidal ends, with a diameter of 3 m, this yields a canister height of 7.94 m, which allows the tank to be stored in the front of Skylon' payload bay.

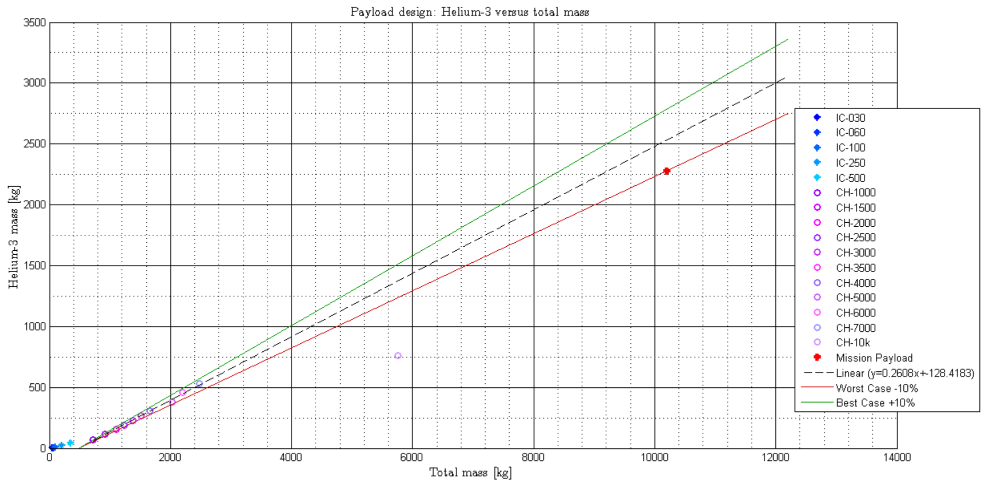


Figure 4.1: Interpolation of total mass to Helium-mass for canisters made available by Wessington Cryogenics [55].

Wessington Cryogenics canister boil-off rates decrease with increasing size. The HC-10k has a boil-off rate of 0.2% per day on Earth (1 atm and 298 K) [55]. In space, the pressure and temperature will be significantly lower, thus the expected boil-off rate of the canister in space is expected to be 0.15% per day. A boil-off rate of 0.15% is also achievable on Earth, depending on the canister's seal. For now, it is assumed that the boil-off rate on Earth is 0.2% (same as the HC-10k), and 0.15% for space. These rates correspond to 4.56 and 3.42 kg of He-3 per day for Earth and space respectively. If no counter measures are taken, a large amount of He-3 would be lost during the transfer. Therefore, the boil-off He-3 must be re-liquefied using a re-liquefier. Cryomech manufactures re-liquefier that can be directly mounted to the canister [56]. Currently their PT415 re-liquefier is the most efficient gas re-liquefier. In combination with the CP1010 compressor, it is capable of re-liquefying more than 27 litres per day (2.05 kg/day) . To capture all evaporated He-3 during space transport, two PT415 re-liquefiers (39.5 kg each) must be installed in combination with two CP1010 compressors (214 kg each). The entire re-liquefying process will require 22 kW (11 kW per re-liquefier and compressor). The width of the compressor (0.61 m) adds to that of the final payload structure.

To withstand the hostile space environment, such as micrometeorites and extreme temperature changes, the He-3 canister, CP1010 compressors and PT415 re-liquefier have to be covered with 12 layers of Multi-Layered thermal Insulation (MLI) sheets, as will be elaborated upon in the next section. For redundancy, it has been decided to use 15 layers, each having a thickness of 3 mm and a density of 249 kg/m^3 [57]. This would yield a total mass of 65 kg to cover the complete casing. Additional tank specifications such as maximum pressure and evaporation rates are derived from the already existing canisters (see Sec. 4.2.5).

4.2.3 Thermal Subsystem

The payload has to be maintained at the right temperature to remain a cryogenic liquid. The compressor needs to be maintained at its operational temperature (280 - 311 K) throughout the whole mission, due to water being used in the system. It is assumed that the payload-carrying vehicles provide power for thermal control for the payload. During ascent to LEO and re-entry, the payload bay area of Skylon will provide a suitable environment for the payload. The Space Dock and CTTV will provide heating or cooling power to the payload to maintain the temperature. Finally the LSAM provides the operational temperature during ascent and descent and maintains a sufficient temperature while on the lunar surface. For the thermal analysis of the payload the same method is applied as for the detailed design in Sec. 5.5.

4.2.3.1 Assumptions

Below, the assumptions for the thermal subsystem are listed. They are evaluated for LEO and LLO orbit.

- The payload will be treated as a single node
- Earth albedo factor equals 0.35 [58].
- Moon albedo factor equals 0.07 [58].
- Surface area of the canister exposed to albedo and solar/planetary radiation is 33% of the total surface area.
- Earth radiates with an intensity of 237 W/m^2.
- The surface area of the payload canister equals 89 m^2.
- Distance Moon-Sun is assumed to be constant at 1 AU.
- The effective radiating surface is assumed to be equal to the radius of Earth/Moon.

- Solar radiation and Albedo will be equal to 0 in the shadow cases.
- Other celestial body radiation is neglected (only Moon and Earth considered).
- The payload canisters will be stored in a suitable. environment on the Moon (no extreme temperature cases).
- Total dissipated power is assumed to be 2200 W (10% of total power).

4.2.3.2 Thermal Subsystem Design

Using the method as it will be described in Sec. 5.5 the equilibrium temperature of the payload can be calculated, with the results presented in Tab. 4.2. The surface is assumed to have the outer layer of an MLI which has an average absorptivity of 0.40 and an emissivity of 0.70.

Table 4.2: Equilibrium temperature of payload.

Case	Equilibrium temperature [K]
I: Hot Earth	300.9
II: Cold Earth	188.0
III: Hot Moon	285.9
IV: Cold Moon	201.4

Table 4.3: Power requirement thermal subsystem.

Case	Power required [kW]
I: Hot Earth	0
II: Cold Earth	11.7 (heating)
III: Hot Moon	0
IV: Cold Moon	9.7 (heating)

The temperature required for operations is between 280 and 311 K for the compressors. Based on the results of the equilibrium temperature it can be stated that a cooler will not be needed, while active thermal control will be required for maintaining the temperature. Using the formula's stated in Sec. 5.5 the power requirement can be calculated for the heater, the results are given in Tab. 4.3. The heating power required equals 11.7 kW for maintaining the temperature within the operational range throughout the whole mission. The first thermal control will be MLI of 15 layers. This will have an effective emissivity of approximately 0.01, resulting in a 357 W radiation to the outside in the cold case. This means 1843 Watt of the internally dissipated power will be maintained in the spacecraft and will passively heat the craft. The remaining 9.9 kW will be provided by active thermal control. The commonly used option is by use of a heater. Patch heaters can be used, as well as a radio-isotope heater units. Since radio-isotope heater units are less reliable and weigh more, patch heaters will be used for the payload design. A patch heater of 25 by 25 cm can provide a power of 1000 Watt [59], which means 11 patches will be installed (assuming 95% efficiency). This results in a power requirement of 11 kW per He-3 canister delivered by the transfer vehicle (CTTV or LSAM).

4.2.3.3 Budgets Thermal Subsystem

The power requirement of the thermal subsystem follows from the worst case temperatures encountered and the power needed to maintain the spacecraft in its operational temperature. The requirement is found to be 11.7 kW as shown in Tab. 4.3. For the mass the MLI provides the largest mass contribution with a density of 0.73 kg/m^2 [60]. This results in a mass contribution of approximately 65 kg. The MLI also acts as a protection shield for the internal components. The patch heaters require a total area of $14 \cdot 0.25 \cdot 0.25$ m^2 = 0.875 m^2. Since the thickness is in the order of less than a millimetre per sheet, the mass required for heater is assumed to be negligible compared to the overall payload mass. The cost for the heaters is 68.5 € per heater, yielding a total cost of 753.5 € [61]. The MLI layers will cost 65.21 €/m^2 per layer [62]. Using 15 layers this results in a cost of 87055 €. Total cost of the thermal subsystem is estimated to be 88 k€.

Table 4.4: Summary thermal subsystem budgets per canister.

Budget	Value	Units
Power	11.7	kW
Mass	65	kg
Cost	88	k€

4.2.4 Payload Docking

For the payload docking to CTTV, LSAM, Space Dock, and Skylon, the type of docking matters very little. Only a very basic exchange of data is required, and besides the canister itself no transfer of materials is involved. This means that a simple male-female docking system can be used. One light-weight docking system that is in development right now by the Michigan Aerospace Company is the Autonomous Satellite Docking System (ASDS). This system is a commercial, small, light-weight docking system. Fig. 4.2 shows its expected layout.

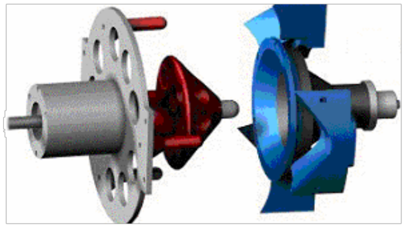


Figure 4.2: Layout of ASDS [63].

Note that the most recent article on ASDS was released in 2003. While results looked very promising, and successful docking had been performed in microgravity, no more information has been released since then, as ASDS was classified for military purposes [64].

ASDS was designed for satellites much smaller than the CTTV or the LSAM. Therefore, the docking mechanisms would have to be redesigned to withstand the torque loads that the vehicles will be subjected to during operations.

The above considerations resulted in the following characteristics. Each payload bay of the involved vehicles will have four male docking systems per canister for safety (ensuring that the canister is able to withstand for example torques) and redundancy. Each payload canister has four female systems. Each docking system weighs approximately 50 kg. The docking system is also inactive in either docked or undocked state, therefore only expected to consume approximately 100 W each during locking and unlocking. Note that mass and power of the docking system is only accounted for in the detailed vehicle designs. The values are not added to the overall payload canister design.

4.2.5 Summary Helium 3 Canister

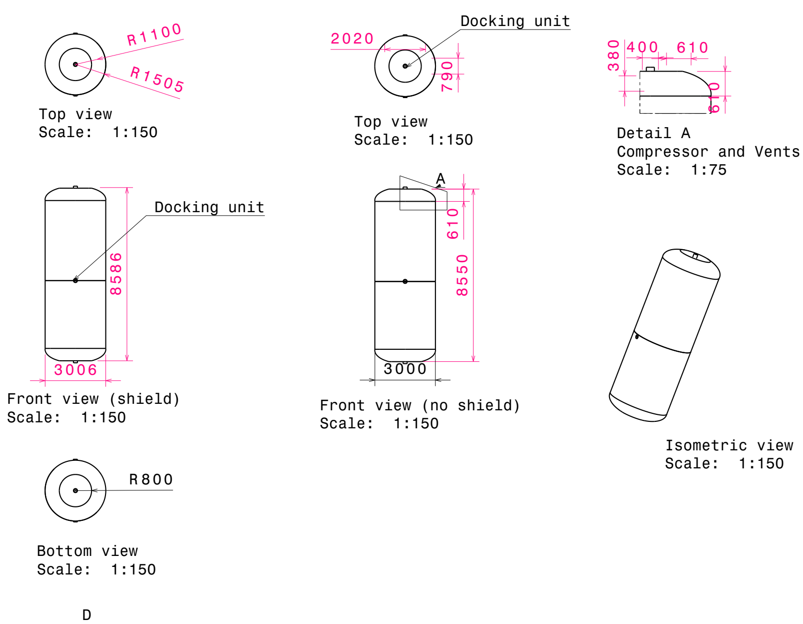


Figure 4.3: Overview of payload design (image not to scale).

Fig. 4.3 gives a general overview of the dimensions and components of the final design. Additionally, the dimensions, masses and power of each structural component are shown in Tab. 4.5. It must be noted that the structural mass contains the liquid He-3 and additional elements (labeled "2" in Fig. 4.3): vent, withdrawal, secondary relief, road and in-transit relief valve and a pressure gauge (see Sec. 4.2.2). The re-liquefier systems volume has been accounted for in the total presented volume.

Table 4.5: Summary of the payload design.

Contribution	Value	Unit	Contribution	Value	Unit
Mass			**Dimensions**		
...Shield	0.06	tons	...Length Canister	7.94	m
...Structure	7.92	tons	...Length Compressor	0.61	m
...Helium-3	2.28	tons	...**Total Length**	**8.55**	m
...Re-liquefier system	0.51	tons	...Payload Radius	1.5	m
...**Total Mass**	**10.77**	tons	**Volumes**		
Power			...Volume Canister	54.4	m^3
...Re-liquefier system	22	kW	...Volume Shield	9.4	m^3
...Thermal Subsystem	11	kW	...**Total Volume**	**63.8**	m^3
...Total	33	kW			
Pressure					
...Maximum He-3 pressure	0.5	bar			

Other considerations: It is important to mention that the payload canisters are lighter that the 12 tons that the Skylon can carry to an altitude of 500 km, as will be shown in Sec. 4.4. This is done to ensure that the design is still compatible with the rest of the system if small changes are made. On the other hand though, much improvement could be made on the canister structure and the re-liquefiers, as this mission will create a new demand for more mass efficient systems. By 2040 the dual re-liquefier system could be replaced with a smaller, lighter and more efficient one. Lastly, it was assumed that the cryogenic structures for storing liquid He-4 are similar to the structures holding He-3 (the density difference of He-3 and He-4 has been accounted for), because reference companies have only been designing canisters for He-4. This has been incorporated in the design by using a 10% safety margin on the structural mass.

4.3 General Communications

This section presents an overview of the most important communication flows within the mission. Firstly, a general overview of all this data flow is given, after which the lunar base, ground segment and Space Dock communication will will be treated. The relay satellite communication is included in the lunar segment, whereas the communication with LSAM and CTTV is treated in their respective detailed designs.

4.3.1 General Overview

In current days, space communication is mainly based on Radio Frequency (RF). Additionally, Free-Space Optical communication (FSO) is becoming more available for spacecraft [65][66]. Inherent disadvantages of RF include a limited bandwidth, low data rates, need for official regulation of frequencies and significant noise for transmission over large distances. Disadvantages of FSO include beam dispersion, absorption, higher power requirements and higher mass for exact pointing [60]. However, as RF is a fully mature technology which proved its capabilities, it is assumed that RF will be used to communicate between the various segments. FSO will only be used for communication between Earth and the lunar base, due to its capabilities of linking high data rates. It is assumed that for this direct link between Earth and Moon high data rates are involved.

Due to the size of the mission, it is assumed that continuous communication is required between Earth and Space Dock, CTTV and lunar base respectively. Three ground stations equally spaced around the equator would be needed, whereas more could be added for redundancy. Next to that, relay satellites are needed to ensure communication between Earth and the lunar base if the base is on the far side of the Moon. Also for communication between the lunar base and the miners, relay satellites might be needed. To ensure coverage of the entire Moon surface and continuous communications allowance, the relay satellites are placed in a halo (or Lissajous) orbit around Earth-Moon Lagrange point 1 and 2 (LL1 and LL2 respectively). The general communication architecture is shown in Fig. 4.4. The complete Communication Flow Diagram (CFD) between every segment of the mission is shown in Fig. 4.5.

Values for the required uplink and downlink capabilities of every link carry a high uncertainty. The possible grade of automation in the future as well as development of new methods to compress data are not taken into account in defining required data rates.

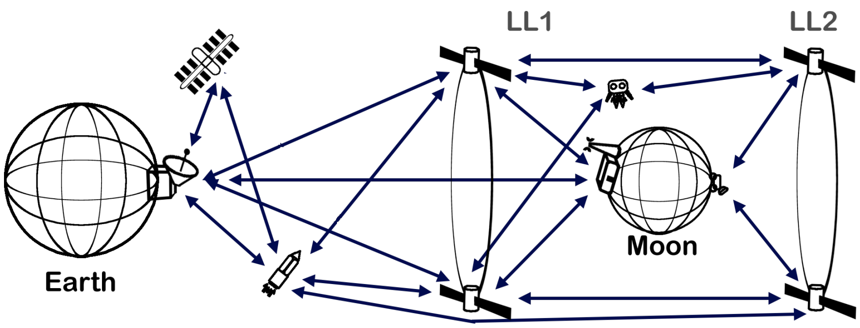


Figure 4.4: Overall communication architecture of the mission.

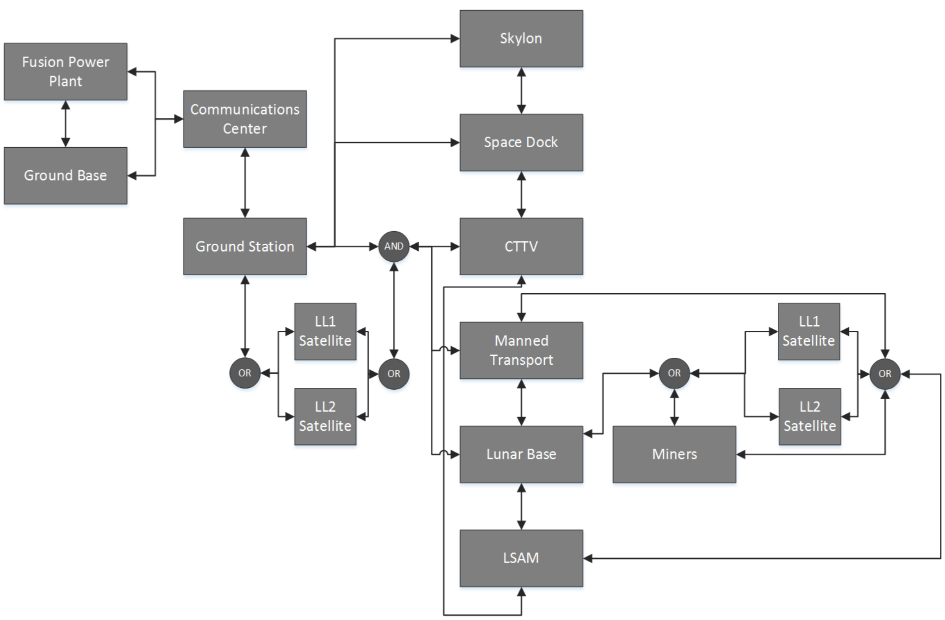


Figure 4.5: Communication Flow Diagram of the mission.

4.3.2 Lunar Base Communications

The lunar segment consists of links between the lunar base and the LSAM, miners, crew replacement vehicles, Earth and (one of the) relay satellites respectively. These will be addressed in their respective sections. For the lunar based communications between base and miners, the main challenge is to provide a line-of-sight network between base and miners. Two options have been considered for this, namely a network of communication towers on the lunar surface or a small relay satellite network. Communication satellites in LLO are not an option, due to short contact times and mostly unstable orbits. The relay satellite was chosen due to high mass and considerable construction efforts required by a communication tower network.

For the relay satellite option, only a satellite in LL1 will be elaborated as the base and mining operations initially take place on the near side of the Moon. Satellites in an orbit around LL1 can provide coverage for almost the entire near side of the Moon. LL2 would work the same, if in the future for example the the lunar base and mining activities will happen at the far side of the Moon. Additionally, LL2 ensures that continous Earth-CTTV communications is possible.

Rheinmetall and EADS Astrium proposed a communication system for the Lagrange points (using RF). The principles of this proposal can be applied to the mission, but larger spacecraft will be needed to manage the large number of miners simultaneously. The expanded version of Intelsat 8, a GEO relay satellite, has a maximum launch mass of 6700 kg. Power requirements are 12 to 18 kW, with 10 kW transmitting power (RF Watt) [67]. NASA's third-generation TDRS-K has a total mass of about 3.5 tons [68]. A detailed design is left for other concept studies. Based on this, it is assumed that the relay satellite masses are about 3-6 tons, realising that it depends on the intensity of the involved data rates. Corresponding dry mass is expected to be about 2-5 tons. High-frequency bandwidth seems suitable for the RF link. Ka, K, Ku, or X-bands are possibilities.

The costs of the LL1 satellite are expected to be in the order of 166 M€ (best case) to 779 M€ (worst case) based on specific cost for dry mass [69][70][60](FY12). Operational costs depend on the number of operating satellites, infrastructure and staff. A small to medium satellite fleet costs about 1 M€/year per satellite [71](FY12). By 2040, changes in technology and production may allow smaller communication satellites and lower specific costs.

Required data rates depend on the type of data that needs to be transmitted. The mission does not involve direct science objectives. Thus, data that has to be exchanged with the base only contains housekeeping data, which usually only requires coarse resolution [60]. In case of continuous teleoperation of the mining (as opposed to occasional command inputs), commands and (visual/video) information have to be transmitted to and from the operators (located either on Earth or in the lunar base). Values for the required uplink and downlink capabilities carry a high uncertainty. First, the miner-LL1 link will be discussed, and consequently, the LL1-base link is considered.

The mining vehicles will be in contact with the base almost continuously. Automated operations that do not vary significantly over the course of the mission require only small data rates. The Curiosity Mission [72] and a rover design of Carnegie Mellon University [73] indicate an approximated range of link capacities of an automated system between 1 kbps and up to 10 Mbps. Typical housekeeping data rates are below 1 kbps [60]. Considering the lack of scientific data in the transmission, taking into account above figures from literature, and considering that a (low-quality) video link is desirable, the link from each miner to the base (via the relay satellite) is estimated at about 50 kbps. Communications from the base to the miners only involve commands, which are estimated to be about 1 kbps. Note that these values apply per miner. The relay satellite in LL1 needs to handle data from all miners and supporting rovers simultaneously. With a total of 2000 miners, the required capabilities of the relay satellite are to handle 100 Mbps of data from the miners, and to send 2 Mbps of commands (however, commands will not need to be sent continuously, so the actual value may be significantly lower). These values are within the capabilities of today's communication satellites. For supporting and transportation functions, a total of 400 Lunar Transport Rovers (LTR) are used. Their data requirement is similar to that of the miners, adding up to 20 Mbps telemetry and 400 kbps commands. Note also that data rates of maintenance operations or additional support rovers have not been included in this consideration.

The distance to LL1 (not accounting for the Halo-orbit and longitude of base) is assumed constant at about 61500 km the lunar surface (using $a \cdot \sqrt{3}\frac{m}{3M}$, where a is the distance between the bodies, m is the mass of the Moon and M is the mass of the Earth). With an arbitrary frequency of 25 GHz (0.012 m wavelength), this corresponds to a space loss of $3{\cdot}10^{-22}$ m^{-2} (-215 dB). Note that in a more detailed design of the lunar segment, each miner and LTR would be assigned a distinct frequency bandwidth. Gain factors, signal power, and SNR have to be chosen such that system noise and space loss do not increase the bit-error rate to a to-be-determined limit. Back-of-the-envelope calculations showed that required power for miners and LTRs is in the order of a few Watts. Power requiremnt for the relay satellite is in the order of a few kW.

4.3.3 Ground Segment Communications

This section presents the design of the ground segment communication. In particular, the link lunar base-Earth is considered. The lunar base has to deliver telemetry data of the mining operation as well as housekeeping data of the base itself. Additional data for the crew (such as video connections) adds to that. The uplink to the base delivers commands and crew-related data. Telemetry and command from and to the miners and LTRs was found to be 122.4 Mbps (Sec. 4.3.2). To determine the required additional data rate, two reference sources have been used. First, the ISS, and second, a proposal by Rheinmetall and EADS Astrium [74]. Both references concern manned outposts. The proposal by Rheinmetall and EADS Astrium uses a downlink of 50 Mbps and an uplink of 10 kbps for a lunar base [74]. Data rates of the ISS are around 50 Mbps continuously, but up to 100 Mbps can be used [75][76]. This mainly includes (scientific) data and video downlinks. Command uplink is no higher than 72 kbps, command downlink rates are up to 192 kbps [75]. According to Eckhart [77], a single studio-quality video link requires 22 Mbps. Based on the data rates of these reference values, a total additional rate (next to the 122.4 Mbps data rate for the miners) of the lunar base of 60 Mbps seems reasonable. About 0.5 Mbps of these are housekeeping data and downlink command data. About 0.2 Mbps are allocated to uplink commands. The remainder is reserved to other data and information and to video streaming and crew communication. The total data rate is a combination of the mining operation and the base. This all adds up to 182.4 Mbps.

An FSO link will be used. Space loss and system noise have to be taken into account. Note that any atmospheric effects on Earth are neglected. The main differences between RF and FSO are the gain and the frequency. Typical FSO systems operate in the near-infrared [78]. A beam of wavelength 1 μm has a frequency of 300 THz. The distance between Moon and Earth is taken to be constant at 385000 km. This yields a free-space loss of $4.272{\cdot}10^{-32}$ (-313.7 dB). The gain is given by Eq. 4.1 [79].

$$G = \left(\frac{\pi \cdot D}{\lambda}\right)^2 \cdot \eta \tag{4.1}$$

Here, D denotes the antenna diameter. Antennas with a diameter of one or two meters are easily achievable, both for Earth and for the Moon. Note that FSO requires much smaller antenna diameters to achieve high gain than RF does. In the above equation, η denotes the efficiency. A typical value of 0.55 is used [60]. A receiver diameter of 0.1 m gives a gain of $6.9087 \cdot 10^{10}$ (108 dB). The same gain is used for the lunar base and for the ground station. No values for noise at wavelength 1 μm have been found. In the following, a system noise of 700 K is assumed. The desired SNR is again set to be 5. Substituting the values in the link-budget equation and solving for transmitter power yields a required 43 mW. As FSO requires very precise pointing, the pointing accuracy and effects of inaccuracies should be considered. The half-power beam width is defined as the angle under which the beam power is above 50%. SMAD [60] offers an empirical relation, shown in Eq. 4.2.

$$\theta = \frac{21}{f_{GHz} \cdot D} \tag{4.2}$$

Substituting values (300 THz, 0.1 m) gives a half-power beam width of 2.52 arcsec. Although the applicability of this (RF-related) empirical equation to FSO is questionable, it is obvious that a high pointing accuracy is required for FSO systems [79]. Thus, whereas the transmitter power is almost negligible, power and mass of the pointing and supporting equipment may be considerable (but negligible with respect to the total mass and power of the base and the ground stations). A more detailed design of the lunar segment has to address this.

4.3.4 Space Dock Communications

The Space Dock mainly communicates with the ground (Earth). Communication with any part of the lunar segment or the relay satellite is not required. The data consists of housekeeping data and telecommands, plus a limited video link availability for the maintenance crew. Most of the time, few uplink commands will be required. An exceptional case is the approach, stay and departure of an CTTV at the Space Dock to ensure successful docking/undocking. The information could be exchanged via a ground station, but it is favourable to let the Space Dock and the CTTV to communicate directly. In the following, first the ground link of the Space Dock is considered. Subsequently, considerations for an dock-CTTV link are presented.

The Space Dock is manned with a maintenance crew. Thus, (crew) data requirements will be approximately similar to data rates of the lunar base (Sec. 4.3.3). Additionaly, the ISS is used as reference; its downlink data rate is 192 kbps, uplink command is 72 kbps. Together with communications for the crew, a total of 50 Mbps are used on the ISS [75]. Based on these values, and taking into account that the Space Dock will have less crew and no scientific experiments compared to the ISS, the data rates for the Space Dock are set at 40 Mbps downlink and 20 kbps uplink. As the signal has to cross through the atmosphere, the frequency should be below 20 GHz to avoid attenuation [60]. S-, C-, or X-band are reasonable options. Here, 7.5 GHz S-band is chosen for downlink, 2.65 GHz C-band is chosen for uplink [60]. The link budget equation is applied for the preliminary design. For downlink, a low-gain transmitter is used, with a gain of 10. This ensures a wide coverage. The receiver gain is set at 1000. The system noise is approximately 135 K [60] and the SNR is set at 5. Using the link-budget equation this gives a required transmitter power of 0.9 W. For uplink, the transmitter gain is set at a value of 100. The receiver gain is set at 1. The system noise is 135 K [60]. SNR is equal to 5. Thus the required transmitter power is 6 mW. For such a small data rate, this value is reasonable. For the Space Dock, mass of the communication system is in the order of kilograms, and thus negligible with respect to total mass [60]. The ground station may have considerable equipment and infrastructure. However, it is assumed that the mission uses existing ground station infrastructure.

For docking/undocking manoeuvres higher data rates, exchanged between CTTV and Space Dock are desirable. For this, both spacecraft will be equipped with Very High Frequency (VHF) or Ultra High Frequency (UHF) communication devices.

4.3.5 CTTV Communications

Communications for the CTTV have to be highly flexible. In general, five links have to be accounted for as can be seen in Fig. 4.4. First is the link to the ground station. Second and third are links with the relay satellites and fourth and fifth are communications with the LSAM and Space Dock. Continuous contact has to be maintained with the ground station. Since the Earth also has constant contact with the lunar base, it is assumed that CTTV-lunar base contact also happens via Earth. To mitigate failure risks, contact might also be established via one of the relay satellites. The links with the Space Dock and LSAM were assumed to be negligible, as explained in Sec. 4.3.4 and 4.3.6. The CTTV communications are designed in detail in Sec. 5.9.

4.3.6 LSAM Communications

The LSAM has a continuous link with the lunar base. From there, telemetry and telecommand will be exchanged with the ground station on Earth. If not directly in line-of-sight of the base, the communication will be relayed by the LL1 (or LL2) satellite. Note that the LSAM-data rates have not been taken into account in the communication budget of the lunar base, the LL1 satellite, and the ground segment, as they are negligible with respect to the data rates of the mining vehicles. Additionally, LSAM operations are restricted to those time periods when a CTTV arrives in LLO and loading operations take place. Communication from the far side of the Moon is provided by the LL2 relay satellite.

As the LSAM is unmanned and has no scientific objective, a housekeeping data rate of 20 kbps is reasonable. Maximum telecommand data rate is 10 kbps. When in view of the lunar base, omnidirectional UHF is used. For an orbit at 100 km the required power is negligible. The communication system is designed in more detail in Sec. 6.9.

4.3.7 Lagrange Point 2

As long as the mining operation is restricted to the lunar near side, a relay satellite around LL2 has lower requirements compared to its pendant around LL1. The only links that have to be supported are from the CTTV and the LSAM. A link is established with the ground stations on Earth. RF is used. If both CTTV and LSAM are in sight, the conceptual combined downlink to Earth is 420 kbps. Uplink sums up to 50 kbps. If mining operations are conducted on the far side, the same requirements apply as for the relay satellite in LL1.

4.4 Skylon: Low Earth Orbit Access Vehicle

To provide two-way transportation to LEO and back during mission operations it was decided to use a Space Plane. Because this segment was only to be designed at a conceptual level, an off-the-shelf concept was considered here. After a Space Plane market analysis, it was discovered that the plane furthest in development is the Skylon. It is assumed Skylon will provide access to LEO for this mission, although a back-up plan is presented as well. Purchasing Skylon gives limitations to the mission such as payload mass and payload volume. Skylon can provide LEO access on a frequent basis, which is desirable for the mission. Although Skylon does not have to be designed in full detail, it is important to understand its characteristics and limitations. It is required to know the in and outputs provided by Skylon. Firstly, the overall lay-out is shown in Fig. 4.6.

Figure 4.6: Skylon lay-out [80].

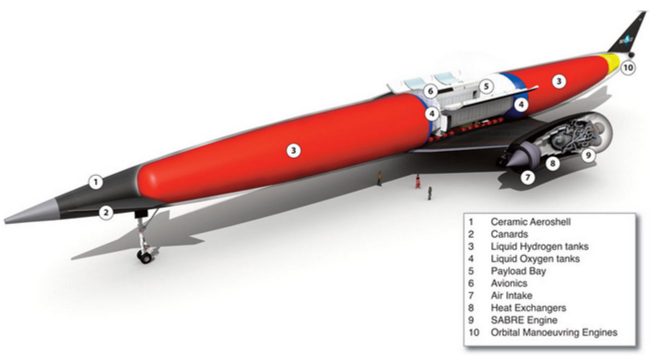


Table 4.6: Skylon C1 characteristics [81][82][83][84].

Parameter	Value	Unit
Nominal Take-off Mass	275	tons
Maximum Landing Mass	55	tons
Operational Life	200	flights
Investment Cost	668 - 856	M€/plane
Operational Cost	7.9 - 8.2	M€/flight
Maximum Mission Duration	up to 4	days
Turnaround time	1-2	days

Based on the C1 model, several characteristics can be identified in Tab. 4.6. Although Reaction Engines is currently working on newer models, the C1 model has the most data available. Some general characteristics are listed in Tab. 4.6.

4.4.1 Mission Profile

The mission profile of Skylon consists of several segments. Each segment has different characteristics and requirements which Skylon has to fulfil. It is assumed that the Skylon purchased is capable of performing the full mission. A typical mission profile consists of the following segments:

1. Ground operations (Attach Payload, Fuelling)
2. Take-off from designated runway
3. Ascent in flight mode
4. Ascent in rocket mode
5. Dock to Space Dock
6. Exchange payloads
7. Undock and manoeuvre to re-entry
8. Perform re-entry using gliding flight
9. Land on designated landing site

The docking and payload exchange will be discussed in Sec. 4.9. Point 7 & 8 will be elaborated upon in the rest of this section. The re-entry flight will be performed using gliding flight. The required ΔV to descent to the Earth surface, is dependent on the height and can be obtained using Eq. 4.3 provided by SMAD [60].

$$\Delta V_{de-orbit} = V_c \cdot \frac{H_i - He}{4 * (R_e - H_e)} \tag{4.3}$$

with the following parameters:

Table 4.7: Parameters de-orbit equation.

Parameter	**Description**	**Value**	**Unit**
H_e	End height of orbit	50	km
R_e	Earth Radius	6371	km
H_i	Initial height of orbit	Variable	km

The initial height will be left variable, to see the influence of the height of the Space Dock. The end height of the orbit is assumed to be a perigee of 50 km, ensuring re-entry [60] . The only unknown left is the circular velocity V_c. The equation for the circular velocity is given as:

$$V_c = \sqrt{\frac{GM}{r}} \tag{4.4}$$

It can be seen that the circular velocity depends on the radius of the orbit and thus on the initial height of the Space Dock. Radius r can be defined as the radius of the Earth plus the orbital height of the Space Dock. It is shown that the ΔV manoeuvre is dependent on the Space Dock height only. The equation was solved and plotted as can be seen in Fig. 4.7

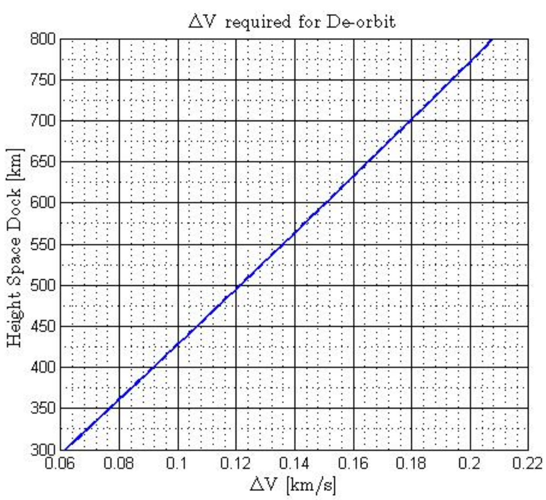


Figure 4.7: Required ΔV for de-orbit.

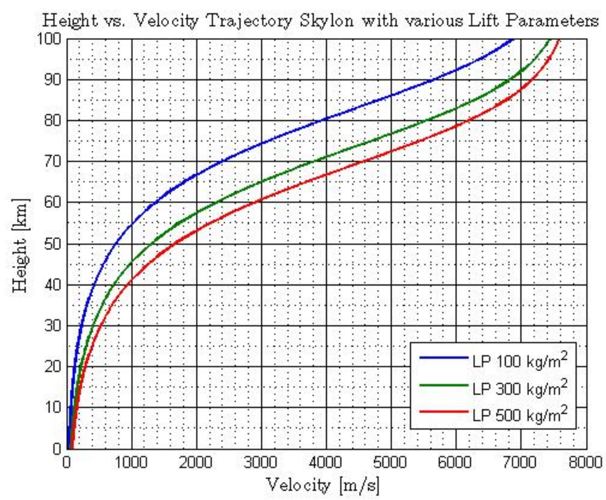


Figure 4.8: Orbital height vs. velocity re-entry Skylon.

The relationship between orbital height of the Space Dock and the ΔV required for de-orbit is almost linear between 300 - 800 km height. The de-orbit manoeuvre does not require a significant ΔV. This relationship is used in the trade-off for finding the orbital height of the Space Dock. After the manoeuvre the Skylon will perform a re-entry into Earth's atmosphere. For gliding re-entry the height vs. velocity relationship is given by Eq. 4.5 by Anderson [85]:

$$V_{re} = \sqrt{\frac{g}{(\rho/2) \cdot k + 1/R_e}} \tag{4.5}$$

The gravity and density in this equation depend on the height Skylon is flying at. Constant k represents the lift parameter LP$=m/(C_l \cdot S)$. Since Skylon is still in the designing phase, the exact value for the lift parameter is unknown. For this a range is assumed in between 100-500 kg/m^2. A typical height vs. velocity trajectory is graphically represented in Fig. 4.8.

4.4.2 Ground Segment

For the mission operations and potentially the precursor mission, launching and landing sites will be a necessity. The demand for new or improved sites will induce significant costs. These costs will consist of construction, maintenance and operational costs. Please note that this only includes the costs for the launch and landing site itself and not the costs for the Space Plane. Next to that, the delivery costs of He-3 to the fusion plants requires an infrastructure which will be described in this section.

4.4.2.1 Construction

The launching site will provide horizontal launch for Skylon. Horizontal launch will be comparable with an airport runway (which can be potentially used for He-3 delivery). Skylon can take off and land on conventional runways with the required length of 5.5 km [86][82]. The cost for a conventional runway is based on reference runways. A runway at Denver International Airport was constructed for approximately 128 M€ with a length of 4875 m [87]. Another example is the Atlanta International Airport runway constructed for 66 M€ with a length of approximately 3050 m. Both examples have a cost in between 0.02-0.03 M€/m runway. Extrapolating these figures to a runway suitable for Space Planes would give an estimated cost of 110 -165 M€. Potentially existing runways could be extended for Space Planes to save costs. In the estimations only construction costs of the runway are included, not the infrastructure around the runway. Multiple runways and platforms might be necessary depending on the payload mass requirements per year. The State of Florida [88] conducted a feasibility study to compare building a new spaceport (an airport dedicated to the use of Space Planes) and expanding an existing launch site. The study showed an estimated cost of 507 - 760 M€ for a new spaceport and 28 - 77 M€ for expansion of an existing site [88]. The first commercial spaceport, Spaceport America, had an overall project cost of approximately 185 M€ [89]. The cost depends on the capacity and number of runways/platforms. For the mission an initial spaceport at a cost of 300 - 400 M€ is assumed, having a runway suitable for Space Planes and other facilities required for operating and maintaining the Skylon.

4.4.2.2 Maintenance

Required runway and ground facilities maintenance will be comparable to maintenance performed on the larger airports. Take-off and landing of the Space Planes will damage the pavement of the runway, while as the other ground facilities require scheduled maintenance as well. The maintenance costs for the spaceport will be orders of magnitude lower than the construction costs. Since the construction costs of the spaceport is already a marginal part of the total mission costs, it is not further taken into account for the total mission costs.

4.4.2.3 Operations

The most important function of the launch site is providing good launch facilities. Operations consists of processing and integrating the vehicle with payload, launch management & control, post take-off responsibilities and logistics. The use of a Space Plane simplifies ground operations compared to vertical launch. The ground operations will be similar to small airport ground operations. The dedicated spaceport will facilitate all the operations. Since the spaceport will be rather small, the operational cost of the airport will have a negligible influence on the overall mission costs. The main operational cost will follow from the He-3 distribution in Sec. 4.4.2.4.

4.4.2.4 He-3 Distribution

The distribution of He-3 to the fusion plants is part of the mission. Considering the volume of the He-3 and safety issues it was decided to transport the He-3 around the world by aircraft. It would be a possibility to either charter an aircraft or to purchase a new aircraft. Several private companies offer an Airbus A330 for approximately 10000 - 19000 € [90][91]. In contrast, purchasing a new Airbus A330 would cost approximately 165 M€ [92]. The Airbus A330 its cargo volume is larger than required, but it would be a valid option based on security and storage pressure of the He-3. A larger volume would require a lower storage pressure for the He-3, decreasing cost and complexity. Next to that, security systems have to be installed to ensure safe flight. Considering the value of the He-3 transported, it might be interesting for terrorists and thieves. Assuming a life time of 30 years for the A330 and an average round trip flight time of 10 hours, purchasing the Airbus would be viable if 45-90 flights (round-trip) a year are performed. Since the Boeing can be used for other purposes (collecting payload, transporting equipment), investing in an aircraft will be profitable. The total estimated costs for distribution are assumed in between 200-300 M€. Operational cost will be calculated based on the fuel used for a full flight of the A330, carrying 97530 litres of fuel with an average fuel price of 0.59€ per litre. This results in an operational cost of 2.6 - 5.2 M€ per year, based on the 45-90 flights.

4.4.2.5 Ground Segment Summary

The costs found in the previous sections are summarised and given a reliability indication in Tab. 4.8.
The cost estimations for the construction of the spaceport and distribution of He-3 have a high reliability since the technology is already available and used. Compared to the overall mission these values will only have a small influence on the overall costs. The operational cost and maintenance cost of the spaceport will be negligible especially in the initial phase of the mission (lower scale). At a higher scale they might have a minor influence on the overall mission costs, although the other segments add costs to the mission in higher orders of magnitude.

Table 4.8: Cost summary per launching site.

Stage	Cost estimate [M€]	Reliability
Construction	200 - 300	High
Maintenance	Negligible	-
Operations	6.4 - 12.8	High
Distribution	200 - 300	High

4.4.3 Payload

The main function of Skylon is to transfer payload from/to the Space Dock. The payload that can be brought up to the Space Dock depends on three parameters:

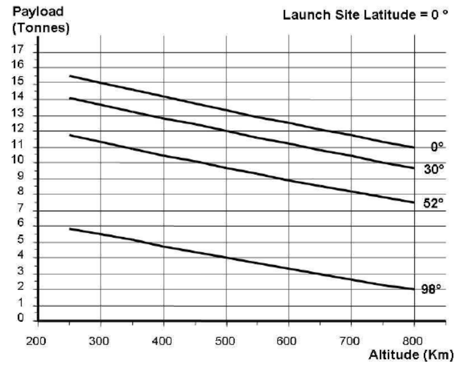
1. Launch site latitude
2. Space Dock height
3. Space Dock orbit inclination

The launch site is assumed to be close to the equator, with 0° latitude. Having the Space Dock at an orbit with 0° inclination, Reaction Engines Ltd. showed the effect on the payload capacity of Skylon with respect to height [80]. This is displayed in Fig. 4.9.

Table 4.9: Annual cost Skylon [81][83][84].

Cost	Per unit [M€]	Required per year	Annual cost [M€]
Investment cost	668 - 856 per plane	0.74	494 - 633
O&M cost	7.9 - 8.2 per flight	148	1169 - 1214
Total annual cost	-	-	1663 - 1847

Figure 4.9: Skylon C2 payload capacity vs. height [80].



In Sec. 4.5 it was determined that the height of the Space Dock will be approximately 500 km, to ensure a high flight frequency of Skylon. This results in a payload of at least 10 tons, which is sufficient to bringing up the empty canister 4.2 (Note: Fig. 4.9 is for the Skylon C2, for the C1 it is scaled down to find the final value of approximately 10 tons. New developments might lead to higher payload capabilities).

The He-3 transported back from the Space Dock depends on the maximum landing weight of Skylon, which is estimated to be 55 tons. This leaves a payload capacity of approximately 12 tons. In Sec. 4.2 it was shown that for every Skylon flight 2.28 tons of He-3 could be transported back to Earth. To supply the 10% demand an amount of 88 flights is required. Additional flights have to be performed to re-fuel CTTV at the Space Dock. In Sec. 4.6 it was determined that the propellant mass required for one CTTV flight is 27.4 tons. The total propellant mass (including 10% structures) requirement of 663 tons per year. To facilitate this 56 flights of Skylon will be necessary. An additional four flights per year are assumed for crew changes or unforeseen maintenance in the Space Dock. Total flights per year for Skylon equals 148 flights.

4.4.4 Annual Cost Skylon

The life cycle cost of Skylon depends on its lifetime, cost, operational cost and maintenance. The goal is to express these costs on an annual basis. The lifetime of Skylon is assumed to be 200 flights [80][81]. Looking at the number of flights per year, the average lifetime of a Skylon plane is approximately 1.35 years. Preferred would be a fleet of Skylons for redundancy (for example 23 Skylons for a mission duration of 30 years). Skylon will be purchased, eliminating R&D costs. The end-of-life costs of Skylon are assumed to be negligible relative to the other costs.

In Tab. 4.9 the units are brought back to what is required per year. For example, Skylon has a lifetime of 1.35 years if only 1 unit is used, this results in a requirement of 0.74 Skylon per year for costs. This requirement per year will not change if a fleet of Skylons is used, it will only be spread out over more years. It is assumed the Skylons are bought without loans. This results in an annual cost for the use of Skylon of approximately 1.75 B€.

4.4.5 Energy Input

The energy input that Skylon induces in its mission consists mainly of propellant. Other energy inputs are assumed to be negligible compared to propellant usage. Per ascent the Skylon uses 217 tons of propellant of which 67 tons hydrogen and 150 tons oxygen [81][82]. The calorific value of hydrogen is 141.8 MJ/kg. Oxygen is the oxidiser which does not add energy to the system. Using the 67 tons of hydrogen with the calorific value and converting it to MWh gives a energy input of approximately 2660 MWh/flight. Per year this results in an energy input of approximately 394 GWh. Propellant required for de-orbit and other energy inputs are assumed to be negligible compared to the ascent energy input.

4.4.6 Back-up

For the mission continuous operations is a key requirement. Ten percent of the worlds energy demand relies on the mission, causing reliability to be of critical importance. Therefore, in the case Skylon is not able to fly, a back-up system is needed. In the case of LEO access, the only currently available back-up is by using a conventional rocket. Rockets as the Ariane VI, Falcon Heavy and Soyuz can be used with a capsule (for example Dragon) to pick up the He-3 and return to Earth. This will increase the reliability of the mission, although the launch frequency will be lower then with the use of Skylon (i.e. more He-3 per flight has to be returned).

4.4.7 Conclusion & Recommendations

To finalise the conceptual study of Skylon the most important characteristics are summarised in Tab. 4.10. In the table a distinction is made for the scalability of the mission. For different percentages of the global energy demand different Skylon characteristics are required for the mission.

Table 4.10: Overall characteristics Skylon.

Characteristic	**0.1%**	**1%**	**10%**
Number of flights per year	6	18	148
Annual cost per year [M€]	67 - 75	202 - 225	1663 - 1847
Energy input per year [GWh]	16	48	394

It can be seen that the number of flights depends on the propellant requirement of the CTTV and the payload. The crew missions will be included for all the scales. All the values are rounded up to the closest integer. For the three scales the number of flights, cost and energy input are determined and can be used in the feasibility analysis in Chp. 8.

4.5 Low Earth Orbit Space Dock

This section will describe the conceptual design of the Space Dock. First, a general lay-out of the dock will be described. Then, a preliminary mass, power and cost budget will be presented. After that, the orbit of the Space Dock will be designed and the astrodynamic properties will be addressed. Finally, the scalability of the dock will be analysed.

4.5.1 General Layout

There are several requirements which the Space Dock should fulfil, as Tab. 4.1 shows. For the general layout, literature research has been performed on various sub-parts of especially the Mir and ISS Space Dock [93][94]. In order to meet the requirements as mentioned, the layout of the Space Dock is designed as follows. The core of the Space Dock will consist of two modules, around which the canisters of He-3 will be attached. These modules will compare to the Long Duration Orbital Station-7 (DOS-7) [93] module of the Mir space station and the Functional Cargo Block (FGB) [94] of ISS. Each module measures 4 m in diameter and 9 m in length and provides space for a crew of two. It has two docking and four berthing space ports. One of the modules would be used for living facilities, while the other module is the operating base for propellant resupply. Both modules are connected by a cross-shaped docking compartment comparable to Node 1, Node 2 and Pirs of ISS [94]. In order to be able to maintain the CTTVs, a module should be attached to allow Extra Vehicular Activity (EVA). As a reference, the Joint Airlock Quest (JAQ) of the ISS has been used which will be attached to the cross-shaped docking compartment [94]. This module is about 6.1 m long and 4 m in diameter. Furthermore, observation decks are to be attached, in order to observe all operations outside the ISS (i.e. robotic arms, spacewalks). It is assumed that two observation decks are sufficient, and the Cupola module of the ISS has been used as reference [94].

The two main modules will have a total of eight berthing and four docking systems respectively. This coincides with the worst-case scenario of eight berthed canisters of He-3, two docked CTTVs and one docked Skylon (all having the same docking system) leaving one docking system for redundancy. The canisters have to be transported from the CTTV to the berthing systems. In order to do that, retractable arms comparable to ESA's European Robotic Arm

(ERA) will be attached to the modules [95]. Using the ERA as reference yields a range of the arm of 9.7 m, meaning four retractable arms would be needed in order to cover the entire Space Dock. Using four retractable arms also results in redundancy if an arm is out of order and it makes the downscaling of the mission easier. It should be noted that the trade-off between robotic arms will be performed in Sec. 4.9.

The X-shaped docking compartment between the two main modules should also provide the possibility to attach a propellant storage compartment. To refuel the CTTV, 26.9 tons of propellant should be available. Assuming a 10% redundancy, a propellant storage of 29.6 tons would be required. Finally there should be solar arrays attached to the Space Dock, in order to provide power. The amount of solar arrays required is dependent on the size and subsystems of the Space Dock. A preliminary estimation of the total required solar arrays will be made in Sec. 4.5.3. The overall layout, excluding the robotic arms, is shown in Fig. 4.10.

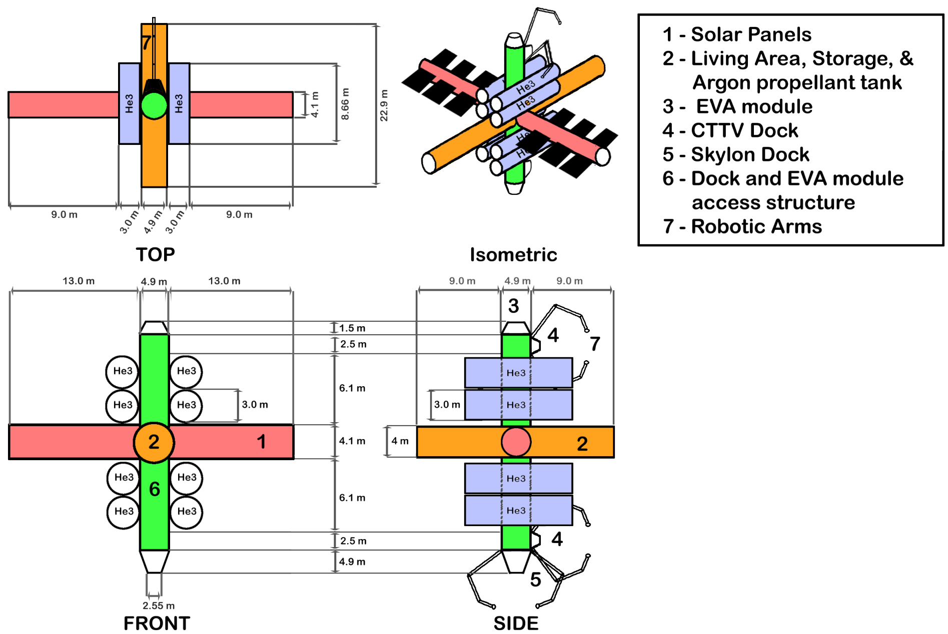


Figure 4.10: General layout of the Space Dock.

4.5.2 Mass Budget

A preliminary mass budget has to be estimated in the conceptual design. To determine the mass, compartments of the ISS and MIR space station have been used as references [93][94]. For the two main modules, the mass has been estimated based on the FGB (the base module of the ISS), the Salyut 7 (one of the modules of the MIR space station) and on the MIR base module [96]. All modules have approximately the same dimensions and are designed for a crew of two and maximum three. The masses of these modules are 24968, 19824 and 20400 kg respectively. This yields an average of approximately 22 tons. For the mass of the X-shaped docking compartment, the docking compartment Pirs, Node 1 and Node 2 of the ISS are used as reference . Node 1 and Node 2 (having six docking systems each) have a mass of 11895 and 14787 kg respectively. Assuming that a similar docking system for four space ports would be two-third of that mass (7930 and 9858 kg respectively) and converting the three-port mass of Pirs (7676 kg) to an equivalent of four ports (10234 kg), yields an average mass of approximately 9.5 tons for the X-shaped connection dock.

For the EVA allowance modules, Boeing's JAQ designed for the ISS is used as reference. The mass of this module is about 10 tons which includes the Common Berthing Mechanism (CBM). The Cupola observation dock of the ISS, having a mass of approximately 2 tons, is used as reference. As two of those are used, the total mass is 4 tons.

The mass of the retractable arms is derived from ESA's ERA and the Mobile Servicing System (MSS) of the ISS [95]. ERA was designed for a handling capacity of 8 tons, having a mass of 630 kg. The main purpose of the retractable arm is to replace the canisters with He-3. These canisters weigh 10.7 tons, but for contingency the arms will be designed for handling a mass of 15 tons. Assuming an equivalent mass of 79 kg per ton of handling capacity (from ERA), this would result in a total mass of 4720 kg for the arms. The MSS has a range of 17.6 m, meaning only two of these arms would be required. The total mass for the MSS would be 2994 kg. For the estimation of the mass of the retractable arms the average of these masses is taken, resulting in a placeholder mass of approximately 3900 kg for a set of two arms. This mass will change later in this chapter, after the trade-off between different robotic arms has been done in Sec. 4.9.

Next, the mass of the propellant storage structure needs to be estimated. As explained in Sec. 4.5.1, a propellant storage module should be present that stores 29.6 tons of propellant. The propellant used for the CTTV is Argon. Cryogenic Argon has a density of 1393 kg/m^3 resulting in a tank volume of 21.5 m^3 [97]. Using the cryogenic argon pressure vessels as manufactured by Airproducts [98] and the conceptual cryogenic vessel design as designed by D.B. Cline [99] as reference, an empty storage tank mass estimation could be made. The empty masses of a 160, 180 and 230 liter pressure tank are 114, 118 and 140 kg respectively. Extrapolating these masses to a 21500 l tank for the propellant storage, yields an empty mass of approximately 14 tons.

For the mass of the solar arrays, first the power budget is needed. Based on the power budget in Sec. 4.5.3, an estimation is made on the mass of the solar arrays. The ISS solar arrays are 3000 m^2, which provide 131.2 kW [100][94]. The total mass equals 8.7 tons, which is equivalent to 0.066 tons/kW. Therefore the mass for the 288.3 kW power supply for the Space Dock, is estimated to be 19 tons.

The estimated masses for each of the sub-parts of the Space Dock are summarised in Tab. 4.11. It also contains the power estimation, which is elaborated in the next section. The total mass of the Space Dock, for 10% of the global He-3 supply, would thus be 107.7 tons. The mass budgets are shown two cases; the Space Dock withoutthe attached He-3 canisters and stored propellant and the Space Dock with the canisters and propellant included. Also it has to be noted, that the masses per compartment are the total masses meaning they are already multiplied by the number of used elements. The reliability of the values of the mass of the various parts of the space can be considered as high, as all these parts are proven and applied concepts in similar missions. Only the reliability of the propellant storage module mass estimation is average, as there are no proved space designs for it. The second and third column show the mass estimation for the global energy demand of 1 and 0.1% respectively.

Table 4.11: Space Dock mass, power, and cost estimations for various energy demands.

Space Dock compartment	**10%**	**1%**	**0.1%**
Main module 1 [tons]	22	22	22
Main module 2 [tons]	22	-	-
X-Shaped connection dock [tons]	9.5	7	7
Retractable arms [tons]	6.8	3.9	3.9
Propellant storage module [tons]	14	14	14
Observation modules [tons]	4	2	2
EVA allowance module [tons]	10	10	10
Total mass without solar arrays [tons]	88.3	58.9	58.9
Solar array mass [tons]	19.4	18.9	18.9
Total mass without payload and propellant [tons]	107.7	77.8	77.8
Worst-case payload mass [tons]	86.4	43.2	10.8
Propellant mass [tons]	26.9	26.9	26.9
Total mass with payload and propellant [tons]	221	147.9	115.5
Total power [kW]	294.5	286.0	286.0
Investment cost [B€]	10.6	7.7	7.7
Operational cost [B€/year]	0.44	0.32	0.32

4.5.3 Power Budget

In this section a preliminary power budget will be presented. In order to estimate the power needed, several similar modules from the ISS and MIR Space Dock have been used as reference. Also the total power budget of the ISS and MIR have been considered. For every module or space station, the generated power has been divided by the total mass of the corresponding module or station. The average of these power/mass-ratios is used for a first preliminary power budget estimation of the Space Dock. An average of 0.284 kW/t was found, hence the Dock would require 30.5 + 264 = 294.5 kW (Dock + 8 He-3 canisters). The power required by the canisters will be delivered by the carrying device, in this case the Space Dock.

4.5.4 Cost Budget

For a cost budget, again estimation from the ISS have been used as reference [101]. The investment costs of the ISS were estimated to be 41.5 B€ and the operational costs were estimated to be 30.9 B€. The operational costs of the ISS were based on an operational lifetime of 18 years. This lifetime can be extended, but it is not known how operational costs will increase. For a preliminary cost estimation of the Space Dock, the cost-to-kg ratio of the ISS is assumed to be representative. For the 10% case, this means that Space Dock investments would be 10.6 B€ and the operational costs 7.9 B€ (0.44 B€ per year). For uncertainty, a 10% margin was added, which yields a cost of 9.5 B€ and 11.8 B€ for the best and worst case investment costs and 0.40 and 0.48 B€ per year for the best and worst case operational costs.

Orbit Design for Space Dock: One of the key design choices of the Space Dock is the orbit around Earth. The inclination of the orbit should be as low as possible, as the Skylon is then able to fully take advantage of its payload capacity as shown in Fig. 4.9. For very low inclinations, the equations for these two orbits are not valid anymore because J_2 has no significant precession effect [102]. Therefore an orbit was chosen in the equatorial plane, meaning the inclination will be zero degrees. Concerning the J_2 effect, a Space Dock in a zero inclination orbit can be placed at any LEO altitude. Based on that, the design choice has been made to orbit the Space Dock at a 500 km altitude. The argumentation for this design choice is the low atmospheric drag. That is favourable for the CTTV leaving the Space Dock and it yields easier station keeping due to a lower decay rate, as calculated in the next section. Next to that, Skylon is capable of reaching the 500 km altitude with almost its maximum payload conform Fig. 4.9. Finally, according to David Wright in 'Space Debris', the amount of space debris at 500 km altitude is relatively low which reduces the risk of collisions [103]. For a circular orbit at this altitude, its period would be $P = 2\pi\sqrt{\frac{a^3}{\mu}} = 5677$ s, or 15.22 revolutions per day.

4.5.5 Astrodynamic Characteristics

For the Space Dock astrodynamic characteristics play an important role in mission planning. Skylon and the CTTV should be able to navigate to the dock. The orbital height for the Space Dock was set at 500 km at zero degree inclination. To provide easier orbit maintenance, the Space Dock has to maintain a height in between 475 - 525 km. Due to drag perturbations the orbit of the Space Dock changes slightly per orbit. In this design phase only drag perturbations are considered, whereas other perturbations are considered negligible. The changes due to drag can be calculated using Equations 4.6, 4.7 and 4.8, where BC denotes the ballistic coefficient in kg/m^2, which is commonly between 125 and 200 kg/m^2 [60].

$$\Delta a = -2\pi \cdot (1/BC) \cdot \rho \cdot a^2 \tag{4.6}$$

$$\Delta P = -6\pi^2 \cdot (1/BC) \cdot \rho \cdot a^2/V \tag{4.7}$$

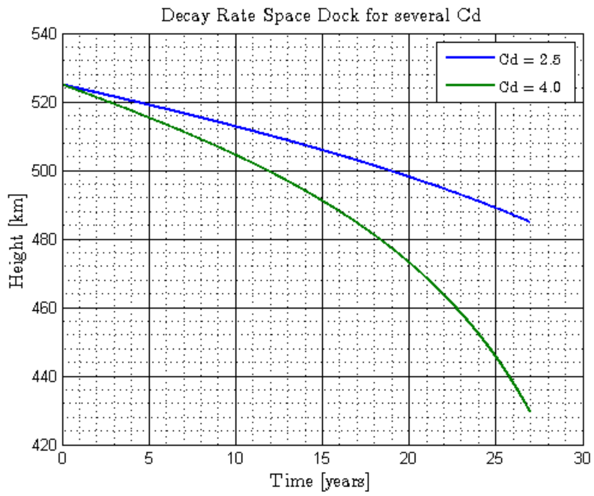
$$\Delta V = \pi \cdot (1/BC) \cdot \rho \cdot a \cdot V \tag{4.8}$$

Using these equations, the change in semi-major axis, period and velocity can be computed per revolution of the Space Dock. The orbit gains new characteristics for the new revolution and the changes can be computed again. This results in decay of the orbit and it finally reaches the minimum required altitude of 475 km. The ballistic coefficient is dependent on the mass, frontal area and drag coefficient of the Space Dock. The frontal area is estimated to be 212 m^2 based on the general lay-out. For the drag coefficient C_D a range of 2.5-4.0 is assumed. With these values the range of the ballistic coefficient is calculated. For the decay rate the worst-case condition scenario is assumed with a mass of 107.7 tons (as the decay rate will be worst for minimum mass). The decay rate is represented in Fig. 4.11. The estimated time for the Space Dock to decay from 500 to the minimum altitude of 475 km is approximately 2 to 7 years depending on C_D.

Table 4.12: ΔV computation for re-boost.

Parameter	**Value**	**Unit**
m_0	107000	kg
I_{sp}	450	s
Δ V	27.67	m/s
g_{avg}	9.81	m/s^2
m_p	1725	kg

Figure 4.11: Space Dock decay rate.



The decay rate from 525 to 475 km is around 18 to 30 years. It can be seen that the C_D coefficient of the Space Dock has a large influence on the total decay rate. When the Space Dock reaches its minimum altitude it needs to re-boost to the high boundary of the orbit range (525 km). For this manoeuvre a 2-boost Hohmann transfer is used. For small altitude changes it can be assumed that the required ΔV equals the difference in orbital velocity of the two circular orbits, hence the required ΔV for the manoeuvre equals $\Delta V = |V_{525km} - V_{475km}| = 27.67$ m/s. One should note though that the manoeuvre is still performed in the same direction as the velocity vector. The time for this manoeuvre is exactly half the period a 500 km circular orbit, 2838 s.
With the ΔV computed in Tab. 4.12 the propellant mass required per re-boost can be calculated using Eq. 4.9.

$$m_p = m_0 \cdot [1 - e^{-(\Delta V / I_{sp} \cdot g)}] \tag{4.9}$$

Assuming that hydrogen will be used as propellant with oxygen, since it is used throughout the mission which simplifies the supply, the specific impulse is estimated. The initial mass is estimated using the estimation in Sec. 4.5.2 and adding the maximum docked payload. This gives an initial mass of approximately 276 tons. Using the following parameters the propellant required for one re-boost is calculated.

The propellant can be provided by a Skylon mission, since few re-boosts will be required over the lifetime of the Space Dock. The propellant mass required for a re-boost represents the worst-case scenario, with all the payload attached.

4.5.6 Scalability

For the scalability, it has to be considered what the results are with respect to the requirements of the Space Dock. For 1% energy demand, only three CTTVs will shuttle between the LSAM and the Space Dock. Only nine canisters per year will be transported. For 0.1% of the energy demand, only one CTTV will be in space which carries only one canister of He-3.

The influence on the Space Dock of downscaling the mission is that only one of the main modules would be needed. One module still provides enough docking possibilities for the worst case, which is docking a Skylon, CTTV and four canisters for the 1% and a Skylon, CTTV and one canister for the 0.1%. Using only one main module, means that the X-shaped docking compartment becomes T-shaped and only half of the retractable arms would be needed to cover the whole Space Dock. Also only half of the operation decks would be needed, since one of the operating main modules would become superfluous. The propellant tank and EVA module will not change. The new masses and total mass without solar arrays for both 1 and 0.1% can be found in Tab. 4.11. As can be seen in the table, the mass of the Space Dock for 1 and 0.1% is equal. The power budget for this Space Dock would be 280.2 kW, calculated similar as in Sec. 4.5.3. The corresponding solar array mass yields 18.9 tons. The cost for the 1 and 0.1% case will reduce only from 10.6 to 7.7 B€ for the investment costs and also operational costs will be reduced by 120 M€ per year. For the total cost analysis in Sec. 7.4, a cost increase and decrease of 10% is assumed for the worst and best case scenario, due to the high uncertainty in costs.

4.6 Continuous-Thrust Transfer Vehicle

In this section, the conceptual design of the CTTV will be covered. The requirements of the CTTV have been discussed in Sec. 4.1. Once the requirements are established, a trade-off will be performed between different possible propulsion types in Sec. 4.6.2. The docking procedures will be explained in Sec. 4.6.3, its conceptual characteristics will be presented in Sec. 4.6.5, and a brief overview of the annual required energy, expectations on long-term operations, V&V and scalability will be presented in Sec. 4.6.5. In conclusion, the subsystems will be defined, which are designed in detail in Ch. 5. All design propositions that follow in this section have been designed keeping in mind the requirement of transporting 200 tons of He-3 (equivalent of 10% of the global energy demand by 2040).

4.6.1 Assumptions

- It is assumed that the LLO orbit is circular and at 100 km.
- No orbital perturbations act on the CTTV during transfer.
- The transfer orbit is always circular.
- Docking time in LLO and LEO takes a maximum of 14 days.
- The round trip assumes the same payload mass.
- The engines have a specific impulse of 5000 s.

4.6.2 Propulsion System

The first important choice that should be made when it comes to the CTTV is the trade-off of the propulsion system. It was previously proposed to use continuous-thrust, fuel-efficient transfer orbits as no human presence is required for the transfer of cargo. Below, a trade-off is shown of different propulsion systems, along with their specific impulse, average thrust levels and other considerations. Based on this trade-off, the final propulsion system for the CTTV will be proposed.

Table 4.13: Table showing the different proposed propulsion systems [104][105].

Propulsion type	Specific impulse [s]	Thrust [N]	Other considerations
Thermal			
Chemical	250 - 500	10^6	
Nuclear	500 - 60000	10^2 - 10^6	Nuclear fallout risk
Laser	unknown	unknown	Requires permanent visual contact with Earth
Solar	10000	1	Requires permanent visual contact with Sun
Electric			
Electrothermal	150 - 1200	10	
Electrostatic	1200 - 10000	0.3	
Electromagnetic	700 - 5000	100	
Nuclear			
Radioisotope	700	1.5	
Explosion	unknown	unknown	Testing forbidden by international treaties

From Tab. 4.13, it can be clearly seen that none of the proposed propulsion systems meets the requirements stated in Sec. 4.1, except for the electromagnetic propulsion system. The most developed project on the subject of electromagnetic propulsion is the VASIMR project by the Ad Astra Rocket company [106]. This VASIMR rocket will in part form the basis of the following preliminary estimated budgets. In the table below, the different specifications of the VASIMR engine are presented.

Table 4.14: VASIMR VF-200 engine characteristics [106].

Required power [kW]	Thrust [N]	Propellant	I_{sp} [s]
200	5.7	Argon	≈5000

To be able to provide enough thrust to transport the payload within a reasonable amount of time, ten VF-200 engines will be mounted together on the payload bulk. However, it is assumed that by 2040 the performance in terms of thrust is doubled to 11.4 N. Therefore, five VF-200 engines are required. These engines will be powered by an on-board nuclear fission powerplant, running on non-equilibrium He/Xe fuel [107]. Lastly, the working gas used by the VASIMR engine will be Argon, a noble gas that is readily available in the atmosphere. Argon is produced on Earth at a rate of 700 000 tons per year, which is about 1 100 times more than will be required by the CTTVs in total per year [108].

4.6.3 Payload Interactions

In order to transfer the payload between the different segments, protocols were set up for these events. Different possibilities were considered for both the attachment of payload to the CTTV in lunar orbit, and detachment of the payload at the Space Dock.

Lunar Orbit: In lunar orbit, no space station and no humans are present, just the payload canisters attached to the LSAM. This requires that the CTTV should have an integrated system to contact and dock with the payload canisters and LSAM. Below, the different docking possibilities that were considered are listed.

- The first option is to directly dock to the canisters, using a precise thrust-assisted approach. It must be noted that this is a precise and delicate manoeuvre, and it is sensitive to disturbances.
- A second option would be a combination of the first option, and a grapple-like mechanism, similar to a harpoon-gun. However, the harpoon would preferably be a non-destructive suction cup or a similar concept, instead of a puncturing arrow.
- Lastly, another option would be an extended robotic arm. This option would require a more important addition of mass that would have to be transferred back and forth between Earth and lunar orbit.

The above options were traded off with respect to required complexity and additional mass. The result follows from the considerations indicated in the above list: the harpoon-like mechanism is the lightest, simplest and most effective system to dock the payload to the CTTV. The robotic arm would be relatively heavy, while the direct thrust-docking would require extreme precision and nearly no disturbances, thus making them not suitable for this mission. Additionally, the presence of the LSAM ensures that a high-thrust system is present for manoeuvres when the two vehicles are still at a relatively large distance (i.e. orbital corrections). However, it would be possible for the LSAM to carry an arm with which to bring the payload canisters to the CTTV.

Earth Orbit: The considerations in Earth orbit differ significantly from the situation in lunar orbit. In Earth orbit, a manned space station can be used as a great asset during docking. Just like on the ISS, the arrived CTTV could be grabbed by a man-controlled robotic arm, docked to the space station, after which the arm can again be used to transfer the payload canisters to the location where the Skylon will dock. Taking into account above considerations, the design choice has been made to include payload-interchange mechanisms to be integrated on the Space Dock and LSAM and not on the CTTV.

4.6.4 Conceptual Characteristics

The estimates of power and mass for this vehicle are based off the VASIMR project as mentioned above, which provides a very possible and probable use as a so-called "space tug" for cargo transfer between Earth and lunar orbit. Next to the mass and power budget some other characteristics are determined. The characteristics in this section are found using an iterative optimisation process using a Matlab script.

4.6.4.1 General Overview

VASIMR propulsion was chosen, with the characteristics shown in Tab. 4.14. One of the requirements is to provide a thrust level between 10 and 100 N. As mentioned before, to realise a reasonable transfer time (less than a year), it is determined that five VF-200 engines will be required to provide a total thrust of 57 N. To be able to calculate the number of CTTVs that would be required to transport 200 tons of He-3 to Earth orbit, a Matlab script was set up. This script contained a model of basic orbital mechanics and a calculation on mass fractions resulting in a mass budget. Limiting the travel time to a year, the model was able to provide useful insight into the changes of mission parameters through the variation of time of flight, payload design (mass of canisters) and propellant mass. Using the mass budget as proposed in Sec. 4.6.4.4, different iterations were performed of the calculation. The results of the iterations are listed in the following sections.

4.6.4.2 Orbital Mechanics

The goal of this section is to determine the required ΔV for the continuous thrust spiral transfer. The lunar orbit is assumed to be at 100 km altitude and the Space Dock is at 500 km altitude, as has been determined in the conceptual design of the Space Dock. The transfer between the two circular orbits is performed by increasing the altitude, which is done by applying continuous thrust in the tangential direction, relative to the orbital velocity. For the conceptual model of this transfer, it is assumed that no perturbations act on the transfer vehicle. These will be included in the detailed design phase. For a simplified illustration of the spiral transfer trajectory, see Fig. 4.12.

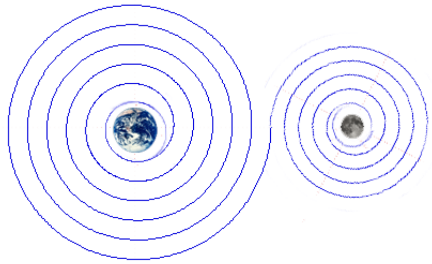


Figure 4.12: Simplified spiral transfer LEO to LLO.

The orbital velocity can be calculated using Eq. 4.10, assuming that the transfer orbit is a series of circular orbits. In this equation, μ is the gravitational parameter of the central body and r is the distance from this body in km. An example calculation is shown for a LEO orbit at 500 km altitude. The values for Earth's gravitational parameter can be used up until the Moon Sphere of Influence (SOI). After this, the gravitational parameter of the Moon is used. The SOI is determined using Eq. 4.11. Values are determined from NASAs Moon fact sheet [109], a is the semi-major axis of the Moon orbit, m is mass of the Moon and M is the mass of the Earth.

$$V_c = \sqrt{\frac{\mu}{r}} = \sqrt{\frac{G \cdot M}{R_E + h}} = \sqrt{\frac{(6.67 \cdot 10^{-20}) \cdot (5.97 \cdot 10^{24})}{(6378) + (500)}} = 7.61\ km/s \tag{4.10}$$

$$r_{SOI} = a \cdot \left(\frac{m}{M}\right)^{2/5} = 384400 \cdot \left(\frac{(7.35 \cdot 10^{22})}{(5.97 \cdot 10^{24})}\right)^{2/5} = 66138\ km \tag{4.11}$$

The total ΔV that is required is determined by the difference in orbital velocity between LEO (500 km) and the Moon SOI, which has to be added to the difference between the orbital velocity in LLO (100 km) and the Moon SOI. The total ΔV is determined in Eq. 4.12, resulting in a preliminary value of 7.85 km/s for the required ΔV. This is close to the value determined by Ad Astra [**?**]. The total round trip will require twice this value.

$$\Delta V = V_c(LEO) - V_c(SOI)\rceil_E + V_c(LLO) - V_c(SOI)\rceil_M = (7.61 - 1.12) + (1.63 - 0.27) = 7.85\ km/s \tag{4.12}$$

4.6.4.3 Power Budget

The main power-consuming elements on board of the CTTV will be the VF-200 engines. These engines consume 200 kW each [106]. Other on board utilities are expected to use less than 10 kW in total, including household computing and communication. The amount of 10 kW is actually expected to be grossly overestimated, as other satellites total power budget usually is in the order of 100's of Watts [110][111]. However, this is a different mission from other scientific or communication satellites. Additionally, the total mass is higher than that of standard missions, therefore it is expected that at least the ADCS will be requiring more power. Next to that, the payload required 33 kW per canister. It is expected, as mentioned in Sec. 4.6.2, that the on-board power plant will provide approximately 1.2 MW of power [107]. This includes some contingency.

Table 4.15: Power budget of the CTTV.

System	**Power usage [kW]**	**Quantity**	**Total [kW]**
Engines	200	5	1000
Other subsystems	10	1	10
Canisters	33	4	132
Total			1142

4.6.4.4 Mass Budget

The required ΔV is known to be 15.7 km/s for the total round trip. A mass budget is derived in this section. First, the dry, payload and structure mass are estimated based on the design of a nuclear fission plant-powered VASIMR system [107]. Using the results of this design, it was estimated that the mass of the total system is a function of the power, or 10 kg/kWh. For 1 MW, this will indicate that the total system will be about 10 tons. The subsystem mass required for this system is estimated at 1 ton for power levels between 100 kW and 2000 kW [107]. A contingency factor of 1.1 is used for those values, resulting in the structural mass (M_S) of 12.1 tons. Another contribution to the dry mass is the mass of the propellant tank (M_{PT}), which is estimated at 10% of the propellant mass [112]. The payload mass is determined using the mass of 10.77 tons for one canister, see Sec. 4.2.5. After some iterations, it was determined four canisters of He-3 will be carried per CTTV, resulting in a payload mass of 43.1 tons, of which 9.12 tons is He-3. With a total dry mass 55.2 tons and an I_{sp} of 5000 s and ΔV of 15.7 km/s, one can find, using Tsiolkovsky's equation, the total mass with propellant to be $M_{wet} = 1.1 \cdot M_{dry} \cdot e^{\frac{\Delta V}{I_{sp} \cdot g_0}} = 1.1 \cdot 55.2 \cdot e^{\frac{15700}{5000 \cdot 9.81}} = 83.6$ tons, where the factor 1.1 denotes the tankage correction. Hence 25.8 tons is propellant and 2.6 tons is tankage. To conclude this section, the conceptual mass budget is given in Tab. 4.16.

Table 4.16: Conceptual mass budget of the CTTV.

Mass	**Mass [tons]**	**Quantity**	**Total mass [tons]**
- Propulsion subsystem	11	1	11
- Other subsystems	1.1	1	1.1
- Propellant tank	2.6	1	2.6
Total Structural mass	**14.7**	**1**	**14.7**
Propellant mass	**25.8**	**1**	**25.8**
Payload mass	**10.8**	**4**	**43.1**
Total	-	-	**83.6**

4.6.4.5 Other conceptual characteristics

The mass budget and the required ΔV are known, from which other characteristics can be determined. The transfer time is calculated by first determining the average acceleration in Eq. 4.13. Secondly, the Transfer Time (TT) from LEO to LLO and back (round trip) can be determined by assuming a docking time of 14 days, see Eq. 4.14. From this value, the number of CTTVs in orbit is determined using Eq. 4.15. It should be noted that these are simple analytical approximations.

$$a_{AVG} = \frac{T}{\frac{M_{dry}+M_{wet}}{2}} = \frac{57}{\frac{57800+83600}{2}} = 8.06 \cdot 10^{-4}\ m/s^2 \tag{4.13}$$

$$TT = \frac{\Delta V}{a_{AVG}}/86400 + T_{dock} = \frac{15700}{(8.06 \cdot 10^{-4})}/86400 + 14 = 239\ days \tag{4.14}$$

To determine the number of CTTVs, the annual amount of canisters needed has to be determined. From the payload design covered in Sec.4.2.5 the amount of He-3 per canister is 2.28 tons. To provide the annual demand of 200 tons He-3, the required amount of canisters per year n_y is 88. Using Eq. 4.15 the number of CTTVs needed in orbit is 15.

$$\frac{(TOF)}{(T_{ES})} \cdot \frac{(n_y)}{(n)} = \frac{(240)}{(365)} \cdot \frac{(88)}{(4)} = 14.4 \rightarrow 15 \tag{4.15}$$

It can be derived that on an annual basis 22 deliveries of four canisters get to the Space Dock. Using this information, the total annual amount of propellant needed can be determined by $22 \cdot M_P = (22) \cdot (25.8) = 568$ tons. All the results mentioned in this section are listed in Tab. 4.17. This includes the results from Sec. 4.6.4.2 which covers the astrodynamic characteristics. The total delivery per year will be 200.6 tons of He-3, which is 0.3% more than the requirement.

Table 4.17: Other conceptual characteristics of the CTTVs.

Characteristic	**Value**	**Units**
Required ΔV (round trip)	15.7	km/s
Average acceleration	$8.06 \cdot 10^{-4}$	m/s^2
Transfer time (round trip)	239	days
Number of CTTVs in orbit	15	-
Annual required propellant (22 trips)	572	tons
Annual He-3 yield	200.6 (+0.3%)	tons

4.6.5 Additional Considerations

In this section, the annual required energy will be covered, along with prospects on the subject of long-term operations of the CTTV. After this a short V&V of the simulation model will be performed. The section will be concluded with an analysis on the scalability of the CTTV.

Annual Required Energy: From Sec. 4.6.4.3, it was obtained that the expected power usage of the CTTV is approximately 1200 kW. This is equal to $1.9 \cdot 10^{-6}$% of the world energy demand by 2040 [3].

Long-Term Operations: An important aspect to keep in mind if this operation is to continue for years on end, is the maintainability of the CTTV system. Because it spends a long time in interplanetary space, the CTTV will be irradiated and probably suffer from impacts from foreign bodies. Therefore, the CTTV must be properly maintained. As the space station in Earth orbit is expected to be manned, maintenance is planned to occur there while the payload is being transferred. Docking times of 7 days at a time have been planned for, allowing for sufficient time to repair minor damage. More severe damage might require one of the backup CTTVs to be sent on its way, while the damaged one remains docked for maintenance.

Verification and Validation: In this section the Matlab script used for the calculations is verified and validated. For the verification the equations used where thoroughly checked and recalculated by hand for one of the data points. At this point the equations used are simple compared to equations that will be used in the detailed design. Therefore, this procedure is considered sufficient for the conceptual design. One of the values can be validated, this is the required ΔV. The required ΔV is validated using a calculation by the University of Colorado [113]. From this article, it was determined that the required ΔV to get from 300 km LEO to GEO is 4.651 km/s. The Matlab script, involving Eq. 4.10, that is used in the previous section gives a ΔV of 4.860 km/s. This is a difference of 4.3%, which is acceptable.

Scalability: In terms of scalability to 1 and 0.1% of the global energy demand by 2040 the same procedure as mentioned in is used. The results for the three amounts can be seen in Tab. 4.18. It should be noted that the design of the lower two amounts is done with the same starting system as for 10%, no specific design for the lower two amounts is proposed.

Table 4.18: Scalability of CTTV characteristics in percent of global energy demand by 2040.

Characteristic	**10%**	**1%**	**0.1%**
Total power per CTTV [kW]	1200	1200	1200
Structural mass [tons]	14.7	14.2	13.2
Propellant mass [tons]	22.8	26.0	10.8
Payload mass [tons]	43.1	32.3	10.8
Total mass one CTTV [tons]	83.6	73.1	34.8
Number of canisters per CTTV	4	3	1
Number of canisters per year	88	9	1
Number of CTTVs in orbit	15	3	1
Required ΔV (round trip) [km/s]	15.7	15.7	15.7
Average acceleration [m/s^2]	$8.06 \cdot 10^{-4}$	$1.0 \cdot 10^{-3}$	$1.9 \cdot 10^{-3}$
Transfer time (round trip) [days]	239	196	108
Annual required propellant [tons]	568	62.7	10.8
Annual He-3 yield [tons]	200.6 (+0.3%)	20.5 (+2.5%)	2.28 (+14%)

4.7 Lunar Surface Access Module

The LSAM has to be designed in full engineering detail. Different surface access vehicles will be used for segments in the mission. This section focuses on the module to bring canisters with He-3 from the lunar surface into LLO. This will be done using a reusable module, which brings up 21.54 tons of payload into LLO, attaches it to the CTTV and then descends again, with an empty He-3 canister in order to be reused. For some characteristics, NASA's Altair [114] will be used as the baseline, until a specific element is replaced or rescaled to the specific needs of this mission.

4.7.1 Functions

The LSAM has to fulfil the functions addressed in this section. From these functions it follows which subsystems are needed during each phase of the LSAM operations. The phases within the LSAM segment are listed below and are shown in Fig. 4.13.

1. He-3 loading.
2. LSAM ascent
3. LSAM reaches LLO
4. LSAM manoeuvres to CTTV.
5. Rendezvous with CTTV.
6. Descent to Lunar surface.
7. Land at take-off site.

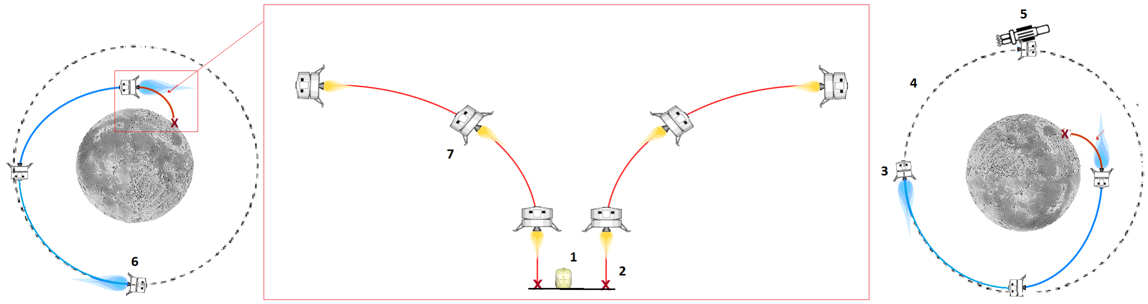


Figure 4.13: Sketch of the LSAM operation phases.

The first phases are aimed at transportation of the He-3 canisters back to LLO. The general functions for the LSAM are to withstand the outer space, lunar environment and provide safe storage and transportation of the He-3 canisters between LLO and the lunar surface. Next to the functions which hold for the entire LSAM operations, the phase specific functions are the following:

- **1.1** The He-3 needs to be loaded into the LSAM by a mechanism.
- **1.2** The LSAM provides a large enough volume to store the He-3 vessels in.
- **2.1** The LSAM delivers a certain amount of directioned thrust.
- **2.2** The trajectory of the ascent is tracked and controlled.
- **3.1** The system times when it is positioned in orbit.
- **4.1** The LSAM determines whether it is in stable orbit.
- **4.2** The LSAM performs correction manoeuvres (prepare docking).
- **5.1** The LSAM docks to the CTTV and transfers the payload.
- **5.2** Manoeuvres need to be timed.
- **6.1** Timed thrusting manoeuvres are provided.
- **6.2** The LSAM trajectory needs to be traced and controlled in terms of braking.
- **7.1** The LSAM determines its position with respect to the landing site.
- **7.2** The LSAM structure should withstand the impact at the lunar grounds.

4.7.2 LSAM Mission Architecture and Trajectory

The driving parameter besides the amount of payload to be transferred is the ΔV required for ascent and descent. It is assumed for now that the thrust settings are constant, which means that the ascent and descent manoeuvre are assumed identical but opposite in direction. A Matlab script has been written in which the user can change the altitude at which the manoeuvre starts and the maximum acceleration the propulsion system can deliver. This will allow quick analysis to get the optimal acceleration and starting altitude for the manoeuvre. The following has been assumed in this script:

- Constant mass and thrust, and therefore acceleration.
- The determined thrust angle with respect to the vehicle does not change.
- The manoeuvre is performed until an altitude of 150 meters, at which the vehicle will be assumed to ascend vertically.

The script does the following:

1. Define constants.
2. Import user-defined values (acceleration and starting altitude).
3. Calculate the ΔV to bring the vehicle from a 100x100 km to a 100x[user-defined] km orbit and its respective velocity. This velocity will be the starting point of the following iteration.
4. Start iteration to determine the ideal thrust angle θ.
5. Start iteration to determine the trajectory for a given θ, acceleration and starting altitude.
6. Depending on the settings, plot the trajectory, $\Delta V - h_{init}$ or $\Delta V - a$.

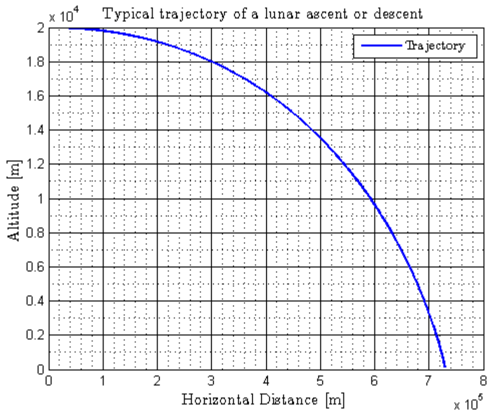


Figure 4.14: Typical trajectory of a lunar ascent or descent.

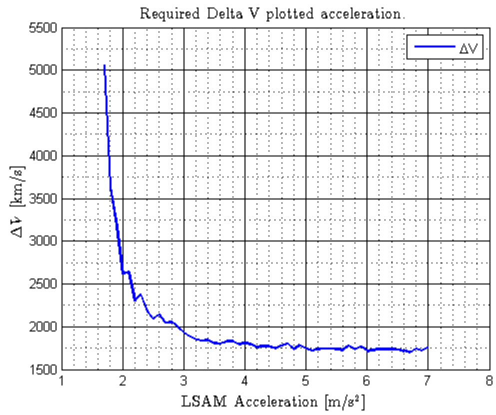


Figure 4.15: ΔV plotted against acceleration at 20 km starting altitude.

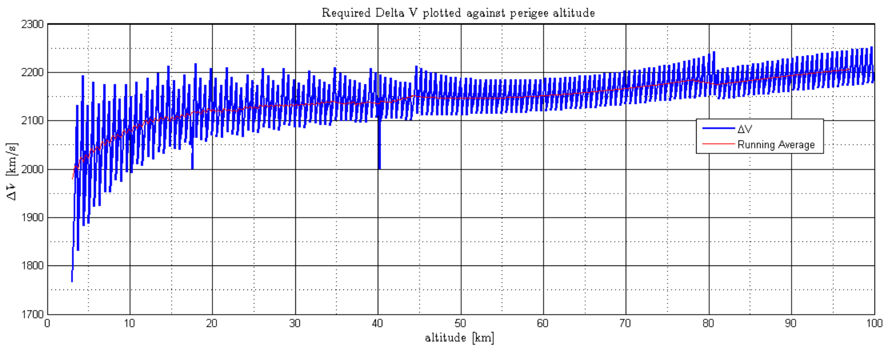


Figure 4.16: ΔV plotted against initial altitude.

The results are given in Figs. 4.14-4.16, which show a typical trajectory, ΔV against a and ΔV against h_{init} respectively. One can conclude from Fig. 4.16 that the lower the starting altitude is chosen, the less ΔV is required. If the altitude is chosen too low additional risks are introduced. Therefore a starting altitude of 20 km was chosen. With this value the desired acceleration was evaluated. Fig. 4.15 shows that the amount of ΔV required decreases when the acceleration is increased. From a value of 3 m/s^2, the improvement per acceleration increment starts decreasing rapidly. A value of 2.5 m/s^2 was chosen as required acceleration. It can be derived from Fig. 4.15 that the required ΔV is 2121 m/s.

4.7.2.1 Staging

A fundamental question to address is whether a single-stage-to orbit system or a staged LSAM should be used. Staging can yield mass savings. To determine whether the savings justify an increase in complexity, calculations have been performed using Tsiolkovsky's Equation (4.16). The conclusions are presented in the following. Note that the calculations do not represent actual design values, but are only intended to give a qualitative comparison of single-stage and multi-stage.

$$\Delta V = I_{sp} \cdot g_0 \cdot \ln \frac{M_{begin}}{M_{end}} \tag{4.16}$$

Here, I_{sp} denotes the specific impulse (450 s), M_{launch} and M_{end} denote the spacecraft mass at begin of manoeuvre and at end of manoeuvre, respectively. Mass consists of the three main components payload, propellant, and structural mass. Structural and total mass are related by a ratio. Typical values are higher than 7% (here, 8% are assumed). For multi-stage rockets, the total mass of the second stage is the payload mass of the first stage.

For a payload of 50 tons and an example ΔV of 3.268 km/s (twice the required orbital velocity), a single-stage LSAM has a total mass of 126 tons. Structural mass is 10 tons, propellant mass is 66 tons.

The results for a two-stage LSAM depend on the distribution of ΔV. If the first stage delivers 1000 m/s, the total launch mass would be 134 tons. The first stage would have 11 tons structural mass and 27 tons propellant mass. The second stage would have a total mass of 96 tons, with 38 tons of propellant, 8 tons of structural mass, and 50 tons payload. Values have been computed for a range of ΔV-distributions. The total savings in mass and propellant are insignificant. Disadvantages of multi-staging are the increased complexity, recovery/return of the first stage to base, and re-assembly of first and second stage for the next launch. Based on this, the LSAM is chosen to be single-stage.

4.7.3 Budgets

To evaluate the LSAM preliminary budgets on mass and power have to be estimated.

4.7.3.1 Mass Budget

The LSAM's primary task is to move large amounts of payload from lunar surface to LLO. The primary driver of the mass budget is the payload mass and the amount of propellant required. The main components contributing to the mass budget will be the payload, docking equipment, the propulsion system, and other structural components. These will be treated specifically in this section.

Payload: The total mass of the canisters filled with He-3 is set at 10.77 tons (see Sec. 4.2). The CTTV will take four canisters to Earth every two weeks. This puts the requirement on the LSAM to deliver four canisters to LLO every two weeks. To minimise the time the CTTV stays in LLO it is favourable to bring up the canisters in short succession. The latter option has two setbacks. First, launching two weeks' worth of He-3 in one single launch on one vehicle increases the risk. Second, an LSAM for a payload of four canisters would have a large total mass, possibly making handling and ground operations of the vehicle difficult. There might also be volumetric issues, as each canister has a volume of 63.8 m^3. An advantage of launching all canisters on one LSAM is that only one launch per two weeks is required, and the number of rendezvous and payload exchange operations is lower. Taking into consideration advantages and disadvantages, the number of canisters launched per LSAM is set at two. Each LSAM will carry 21.54 tons of payload to LLO. The full canisters will be exchanged with empty ones. Together with additional supplies for the base the empty canisters constitute the payload to be brought back down to the surface.

Docking Equipment: It has been shown that the LSAM should also be accommodating docking manoeuvres with the CTTV, which is orbiting in LLO, waiting for payload to carry back to Earth. Docking equipment includes an attachment point to the CTTV and an arm. Estimations are based on the European Robotic Arm (ERA) and Canadarm2 [95][115]. ERA weighs 630 kg and can move a maximum mass of 8 tons. Canadarm2 can move 116 tons and has a mass of 1.8 tons. Note that Canadarm2 is more efficient than ERA with respect to mass handling. The arm of the LSAM is estimated to weigh about 0.8 tons with a handling capability of 13 tons (approximately one payload canister).

Structure: An approximation of the structural mass is derived from the Altair cargo model [114]. It is given that a 43.2 ton model can bring 14.5 tons from LLO to the lunar surface. Subtracting this mass from the total mass along with the propellant mass required for this manoeuvre, one can derive a dry mass of 13 tons. Scaling this up linearly for a payload mass of 21.54 tons would yield a dry mass of over 19.3 tons. It can safely be assumed that this value is significantly lower, because the size of the vehicle does not have to be scaled up linearly and additionally, the payloads can be carried externally, as they will be delivered to the CTTV in LLO already. A preliminary value of 15 tons of structural mass has been chosen.

Propulsion System and Propellant Tanks: To get a concept for propulsion system, the RL10B-2 system was chosen as a reference engine. This engine delivers 110 kN of thrust and weighs 277 kg and uses liquid Oxygen and Hydrogen in a 5.5:1 mixture [116], which can be taken (see Sec. 4.8.2 from the in-situ mined volatiles as required. The propellant tanks are assumed to take up 10% of the propellant mass, for simplicity just multiplying the to-be-calculated mass fractions by 1.1. A desired acceleration of 2.5 m/s^2 has been used when determining how much thrust is required. The amount of thrusters therefore depends solely on the total take-off mass, depending on the amount of propellant required. To determine the mass of the propulsion system, propellant and tankage, one has to consider dry mass and the fuel fractions found through Tsiolkovsky's equation. For an assumed ΔV of 2.121 km/s for a surface to LLO manoeuvre and vice versa and an I_{sp} of 450 s, this value is found to be 1.6168. Multiplying this figure by 1.1 to account for tankage approaches is within the desired accuracy range of this conceptual consideration. This yields a mass fraction between lunar surface and LLO of 1.7785. The LSAM ideally has to come back with exactly no propellant, for optimal efficiency. The dry mass is found by adding up all previously considered components, yielding a mass of 34.48 tons, which excludes the delivered 4.56 tons of He-3. Multiplying by 0.7785 yields a propellant plus tankage mass for the first manoeuvre of 26.84 tons. Then the second manoeuvre is to bring up 34.48 + 4.56 + 26.84 = 65.68 tons. This will require an additional 51.13 tons of propellant plus tankage. Adding up all the figures will give a total lift-off mass of 116.8 tons, of which 78.14 tons is propellant and 7.10 tons is dry tank mass and engines. To accelerate this mass with a maximum value of 2.5 m/s^2, one requires a thrust of approximately $116800 \cdot 2.5 = 292$ kN, three RL10B-2 engines, accounting for a total of 0.84 tons. These additional values require an additional iteration of which only the results are given. All the dry mass has been added, and the mass fraction of 1.6168 is used in this instance. This gives a total lift-off mass of 116.01 tons, of which 69.93 tons is propellant.

Table 4.19: Conceptual mass budget for a single LSAM.

Component	**Individual Mass [tons]**	**Amount Required [tons]**	**Cumulative Mass [tons]**
Structure	15	1	15
Robotic Arm	0.8	2	1.6
Payload Containers	8.49	4	16.98
He-3	2.28	4	4.56
Propulsion Engines	0.28	3	0.84
Propellant Tanks	7.10	1	7.1
Propellant	69.93	1	69.93
Total			116

4.7.3.2 Power Budget

A preliminary breakdown in power budget can be made as the available subsystems and partly the elements of the subsystems are defined. The power-consuming subsystems are mainly the payload subsystem and the ADCS. Also slightly smaller fractions of the total power are assigned to the communications, command and data handling, thermal and propulsion subsystem. For the percentages of operating power of the different subsystems, the typical power consumption by a module is taken from SMAD (table 10-9) [60]. For previous and planned autonomous landers fuel cells are often chosen as a power supply.

Payload system: The payload segment demands the largest amount of power from the power supply. The compressors for the payload storage tank are said to have a power requirement of 22 kW for each tank. An additional 11 kW is required for thermal control. It is assumed that the power fraction of this system ranges from 40 to 80% of the total operating power. As the power requirement of these tanks is rather large compared to reference spacecraft payloads, the power fraction of 80% is used.

ADCS system: The power need for the attitude control and determination is significant. Attitude should be correctly determined during the ascent and descent trajectories, at the moment the LSAM reaches LLO and during the docking with the CTTV. The gyros will have a 90 to 150 W consumption and the reaction wheels will consume 10 to 110 W depending on their size. The power for the thrusters is not available yet. A first estimate on the power consumption for this subsystem is in the order of 200 to 500 W.

Communications and C&DH system: It is stated in Sec. **??** that for a 100 km orbit the power consumption is negligible. The communication with the guidance, especially for the docking with the CTTV is of great importance. Typical percentages for this subsystem are power consumptions of 5 to 10% of the total operating power.

Thermal system: For the thermal subsystem surface coatings, heaters, radiators and sublimators are incorporated. Typically the thermal subsystem demands 5% from the total operating power.

Propulsion system: For the propulsion system and the propellant tanks the RL10B-2 system was used. Power consumption of this subsystem normally is likely to be in the range of 0 to 5%. Due to the use of in-situ resources there might be extra equipment needed on this, this will probably require new storage and propulsion systems. To include this the power demand is estimated at 2% of the total power. A rough number on the power of a thruster is 5 W per thruster for the conventional designs.

Power system: When the system uses rechargeable power sources and the system has a power management system, electrical power is needed for this subsystem as well. As the payload already demands a large fraction of the total power, the power system power budget is put at 4%.

Table 4.20: Conceptual power budget for the LSAM.

Subsystem	**Power required [W]**	**Fraction of total operating power [%]**
Payload	66000	80
ADCS	500	8
Communications and C&DH	250	4
Thermal	125	2
Propulsion	125	2
Power	250	4
Total	67500	100

Preliminary estimate power budget: The outcome for average operating power of the LSAM to be 67.5 kW might be slightly overestimated compared to the Altair characteristics. This can be explained by the fact that the LSAM will have at least double the mass of the Altair and the He-3 canisters have a high power demand. The average operating power of 67.5 kW has a baseline function for the further design of the subsystems.

4.8 Lunar Base

The following sections will give the overview on the lunar operations. Lunar operations include not only extraction and processing of the volatiles, but also transportation, storage, etc. Furthermore, all considerations are based on the annual 200 tons He-3 requirement. Additional figures and considerations for scalability will be addressed in Sec. 4.8.6.

Firstly, in Sec. 4.1, the requirements and assumptions for the lunar operations have been listed and further assumptions on the lunar base are listed in the next section. Sec. 4.8.2 then, gives an analysis of the lunar regolith. This will be followed by the mining operations in Sec. 4.8.3. The interacting elements, such as the miners, Lunar Transport Rover (LTR) and lunar base will be treated, as well as the entire layout of the operation. Then, in Sec. 4.8.4 a suitable power supply will be chosen. Analysing the required amount of energy input is done thereafter. Finally, scalability will be treated in Sec. 4.8.6.

4.8.1 Assumptions

- Regolith density is uniform along a depth of 3 m: 1800 kg/m^3.
- Regolith contains He-3 at a concentration of 20 wppb in the Mare Tranquilitatis.
- The Mare Tranquilitatis has a uniform He-3 concentration along its area.
- The Mark Two miner concept is able to attain the mentioned performance.
- The LTR drives at 2 m/s.
- The LTR is capable of performing robotic movement manipulations.
- The Miners and LTR have have a failure rate of 10% and 3% per year, respectively, resulting in 2-3 crew per day for maintenance (considering either nuclear fission or solar power).
- The maintenance on a Miner or LTR can be finished in less than 24 hours.
- The base would need one permanent inhabitant, excluding critical situations
- The power supply maintenance would need no crew in case nuclear fission reactors would be used[1].
- Some modified LTRs are used solely for assisting crew in their operations (maintenance, building, replacing, etc.).
- The power supply maintenance in case solar arrays are used can be done by three crew (in other words, roughly 150000 m^2 per crew per day, or roughly one 387 by 387 m square)
- Smaller task or task with a lower frequency are performed by the available crew in between their duties.
- No extra crew is allocated for major simultaneous failure rates of the systems involved.
- Crew will be busy operating during continuous operations only (precursory missions are beyond this scope).
- Lunar processing plant requires 2.4 MW.
- Wireless energy transfer (by means of electromagnetic waves) takes place at 65% energy efficiency.

4.8.2 Regolith Analysis

From a NASA study, which deals with the simulation of the composition of lunar regolith [117], lunar regolith is defined as: "Lunar regolith is a mixture of rock, mineral and glass fragments transformed into a distinctive material by a unique combination of space weathering processes."
From Slyuta et al, an overview was obtained of the possible He-3 concentrations per category [27], which has already been presented in Tab. 3.1. Cat. I is the richest in He-3. The Apollo 11 samples confirm this value of 15.1 wppb. It must be noted that due to agitation during the return of the samples, much of the He-3 was lost, up to 42 % [118] as stated by Schmitt. Therefore the concentrations listed in the aforementioned table might be well below the actual values. Additional sources also state that some other lunar samples contained 20-30 wppb [119]. To stay conservative, a value of 20 wppb of He-3 is assumed from now on in Cat. I regolith. Slyuta et al. suggest the mare regions, covering an area of almost 500 000 km^2, containing 61 000 tons of He-3.

Additionally, it is known that 80% of the He-3 is found in regolith particles in the size <50 microns [117]. The density of lunar regolith does also vary along the depth of the soil. It can vary as much as from 1.4 to 2.2 kg/m^3 [117] yielding an average for calculations of 1.8 kg/m^3.

[1]Sec. 4.8.4, as a SSTAR (small, sealed and transportable autonomous reactor) unit would not require any maintenance.

Figure 4.17: The Moon's near side, highlighting the Mare Tranquilitatis by a red circle.

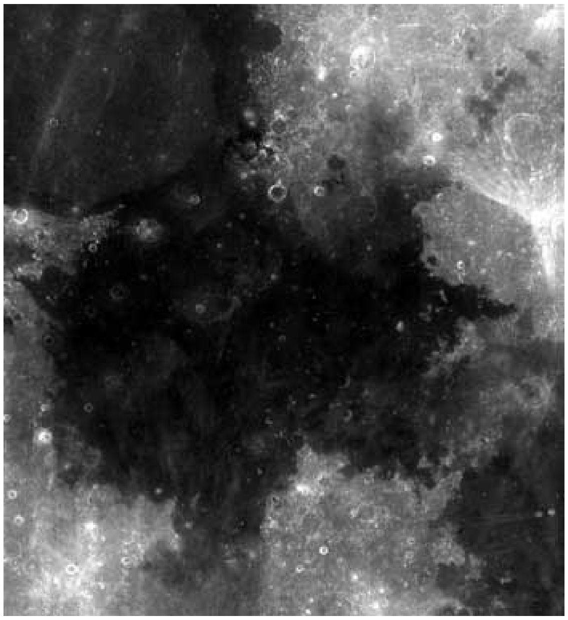


Figure 4.18: Detail of the Mare Tranquilitatis.

Fig. 4.17 shows the different mare regions as the darker spots on the lunar near side. Apollo 11 (which brought back the richest soil samples in terms of He-3 (from Tab. 3.1) landed in the south-west part of the Mare Tranquilitatis (MT, Fig. 4.18). The MT itself (highlighted by a circle) has a diameter of about 870 km [120]. It would therefore be advisable, with the current knowledge, to fix the operations within this He-3 rich Mare region.

As previously mentioned, the lunar regolith contains some other volatiles next to He-3. Tab. 4.21, summarises the most interesting volatile elements for resource usage. In addition, Taylor suggest building structures out of the lunar regolith and Hintze proposes extracting volatiles by heating it with microwave radiation [121][122].

Table 4.21: Summary of all extracted volatiles by Mark III miners.

Gas	**Mass fraction of the total volatiles mass [%]**	**Total hourly mining rate [ton/h]**
He-3	0.0055	0.0114
H_2	33.51	69.6
H_2O	18.13	37.6
N_2	2.75	5.7
CO_2	9.34	19.4
CH_4	8.79	18.3
CO	10.44	21.7
He-4	17.03	35.4

4.8.3 Mining Operations

4.8.3.1 Machinery and Storage

In the following subsections, the three interacting elements will be introduced together with their specifications and characteristics. These are the machinery equipment (miner and rover) and the base (including the process plant along with all other complementary functions).

Miner

As the He-3 can not be readily picked up from the surface, miners are required to extract the volatiles from the regolith. To perform this task, an existing concept was used as to represent the capabilities of a lunar miner. This miner concept is the "Mark III" miner (henceforth named Mark III). It was derived from a previous, rather similar miner concept (Mark II) [123]. It should be noted though that this miner was designed to perform small-scale operations, relative to the 200 tons requirement, which means that drastic improvements on the miner capabilities could be possible. Its characteristics are presented in Tab. 4.22 and Fig. 4.19 shows the lay-out of the proposed miner concept, while Fig. 4.20 shows the original Mark II concept it was improved on.

Table 4.22: Mark III characteristics.

Specification	Value	Units
Mass	10	tons
Dimensions (l x w x h)	13.4 x 5.4 x 4.9	m
Volume	357	m^3
Mining velocity	27.3	m/h
Non-mining velocity	0.56	m/s
Excavation rate	1258	t/h
Processing rate	390	t/s
Depth of mining	3	m
Width of mining	11	m
Power usage excluding heating	350	kW
Power usage for heating	12300	kW
Volatiles mined per hour	127	kg
Volatile mining loss (included in processing rate)	9.9	%

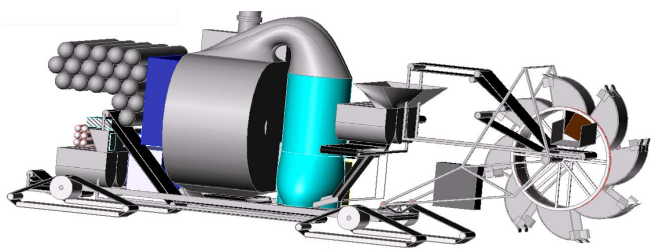


Figure 4.19: Mark III miner subsystems lay-out, without its protective shielding [123].

Figure 4.20: Mark II conceptual drawing [124].

Note that this miner concept is more than just an excavator. Important to know is that this concept comes with a solar-collector system, as to provide in its power need. This radically halves operation time from a theoretical upper limit of 100 to 50%, as these miners can only operate in sunlight. Compared to a permanent power source, this would require twice the amount of miners. Sec. 4.8.4 includes an analysis of alternatives for power supply. In all cases the energy would be supplied to the miners by the use of electromagnetic beaming.

The Mark III is able to excavate the lunar soil at a rate of 1258 tons regolith per hour using a bucket wheel excavator that sweeps through a 150° arc, cutting a trench 3 meters deep and 11 meters wide. Then, the regolith is sieved, to only let through the particles smaller than 100 μm. As the regolith is processed through the miner, the volatiles are passed through a series of processing methods (involving screw conveyors, sieves, fluidized chambers, heaters, electrostatic separators) to end up in the volatile storage system. This storage system has two phases: first the gases are cooled followed by a compression stage, hereafter they are stored in pressurised tanks (carbon-carbon tanks are proposed) [123]. The total amount of volatiles mined and thus stored per hour is equal to 127 kg/h.

Lunar Transport Rover

Considering the amount of volatiles collected by a miner per hour (around 127 kg) and the fact that all miners are used for mining purposes only (as to minimise the amount of miners needed in total) a dedicated LTR will be used throughout the lunar operations. These LTRs might also be used for other purposes (i.e. maintenance assistance, operations manipulation, etc.). An existing concept is used as to quantify the capabilities of the LTR. This concept is the ATHLETE rover, which can be seen in Fig. 4.21. Its power usage was estimated to be in the order of 25-35 kW, as a Mark III miner uses 16 kW in order to move around at a speed of 0.53 m/s (when not mining) [123]. The LTR should be capable of driving at a speed of 2 m/s and be able to carry out robotic movements and manipulations. Furthermore, the autonomy (in hours) and derived range (in km) were more of a design constraint, and represent worse-case values (the longest distance covered for one possible LTR sortie). They were found by considering the operation plan from Sec. 4.8.3.2. These values will determine the amount of fuel cells, as it was chosen to use those (contrary to the miners) as the LTRs require much less power. The distance to a specific miner can vary significantly, therefore these cells should be readily interchangeable and able to be stacked up in modules, depending on the range/autonomy requirement at a specific moment.

Table 4.23: LTR characteristics.

Specification	Value	Unit
Mass structure	2.34	t
Mass payload	14	t
Dimensions		
....(l x w x h)	9.4 x 9.4 x 1.1	m
Volume	97.2	m^3
Velocity on Moon	2	m/s
Power usage	30	kW
Volatiles carried (full)	10000	kg
Autonomy (worst case)	16	h
Range (worst case)	114	km

Figure 4.21: ATHLETE rover [125].

Base

The lunar base is the sole contact point with all possible non-lunar elements within the complete mission lay-out. It is therefore important to consolidate all requirements, functions and considerations with respect to this lunar element. Several requirements were identified and listed in Sec. 4.1. These requirements go closely along the functions it must perform. The lunar operations will therefore comply with the following functions.

Stationing and operation of machinery and spare units/parts: For the mining operations space will have to be provided for all machinery and their spare parts or units. A miner takes up a surface area of 72 m^2, while a lunar transfer rover about 75 m^2 [2]. Having several hundreds of these across the lunar surface -in and around the base- will induce a certain need for spatial organisation (especially for those used as a reserve). Using the results found for continuous "around-the-clock" operations in Sec. 4.8.3.2 for the amount of miners and LTRs, about 26000 m^2 needs to be found for redundant miners, while an extra 5000 m^2 is needed for the redundant LTRs. Furthermore, appropriate protective shielding from lunar dust might be needed in order to minimise the losses throughout operational phase.

Storage and refinement of He-3 and supplies (both in as ex-situ)): Within the mining logistics, the lunar base is primarily involved in processing and/or storing the total amount of volatiles. All the volatiles will be kept separate in pressurised tanks in a designated zone. A separate storage facility needs to be incorporated, considering the large amounts of useful volatiles. These are collected and supplied by the LTRs, and were first processed at a processing facility at the base. This facility needs to be capable of handling a considerable volatile throughput capacity (as much as 207 tons/hour). Recently, promising research has been done on semi-permeable membranes/layers of graphene [126]. This technique might reduce the need of such a refinement plant considerably. In addition, all external supplies also need their storage space. Though this can be considered to be significantly smaller than the volatile storage requirement.

Living area for crew: Since there will be human presence, a protective habitat needs to be provided. All life-supporting supplies and protective equipment should be included in here. Also, radiation shielding for crew and assets is important, even more considering that the lunar albedo exposes the astronauts more to radiation as compared to free space conditions [44]. Different solutions exist, and Pham et al argue that using regolith as a shielding substance is somewhat more effective than aluminium alloys [127].

Launch & landing: Since an LSAM will be used for both ascent and descent operations between the lunar surface and LO, dedicated launch and landing sites will have to be provided, preferably near the base. In contrast, for safety and landing accuracy reasons, this site cannot be placed anywhere near vital parts of the base (as there are: power supply, human habitat). Furthermore, these sites can be made out of in-situ resources, as stated before.

Research and maintenance: A separate centre will have to be included for carrying out research used for sustaining lunar operations (i.e. getting more precise He-3 concentrations measurements, or terrain mapping missions). In accordance with other lunar missions this might be a shared centre. Furthermore, a maintenance and overhaul centre has to be included as well, where a protective (both for machinery and crew) workplace is provided.

Power supply and distribution: All operations must be continuously powered. This will be discussed in more detail in Sec. 4.8.4. Power shall be provided by means of electromagnetic energy beaming and collection by the miners. For the base, a power grid will be needed as to power all functioning parts at any given time.

[2] at maximum wheelbase distance, which is variable

Transportation: Providing for easy and risk-free transportation will only enhance logistics. LTRs are travelling along the lunar surface to miners and base. Maintenance stipulates easy access to miners and base. Landing and launch sites need dedicated transportation: the He-3 tanks will have to be prepared for launch onto the LSAM. This means that they must be transferred into the payload canisters of the LSAM, on the launch site. A road infrastructure might become indispensable as operations get more complex.

Operations supervision, control and communication with mission control centre on Earth: A vital part which has to be included is the supervision and control of the lunar operations. This will mean that a "decision centre" (a crew member for example, or central computer, or both, depending on the level of automation) will have to be included in the operations, where all data and decisions will be processed. These originate from the terrestrial mission control centre. Additionally, these decisions and data do not flow purely in one way. Two-way communication is needed as to account for optimal operations supervision and control.

Further on, a short analysis of the required crew capacity is needed. It was decided upon that as a minimum 3-7 crew would be the needed continuously to conduct maintenance and control of the lunar operations. This remains true for the period when automation is not yet completely implemented. This analysis sets the lower bound for the amount of crew needed considering the assumptions stated in Sec. 5.2.1

A power usage was established considering the amount of crew. Project Boreas [128] states that for a basic scientific outpost for six crew the power requirement should be around 60 kW. Considering the mission layout, we might assume this value only as the lower bound for the power requirement for the crew habitat.

The power demand of the refinement facility heavily depends on its throughput capacity. Considering the large amounts of volatiles and the lack of studies on lunar refinement processing, very little can be said about the power demand for such installations. Koelle states that the power required for a lunar base would be driven by the chemical processing plant. He concludes with a power requirement of 3.2 MW for the base [129]. Considering that this is almost negligible compared to that of the miners, not much elaboration is given on this power issue.

4.8.3.2 Mining Operation Plan

Extraction and Processing Requirement

A requirement for the total amount of He-3 needed has been determined in Sec. 3.1: 200 tons of He-3 being shipped back to the terrestrial clients. In order to have this yearly output a series of lunar operations has to be conducted successfully. This process includes:

- Excavating regolith at a required rate, which will drive the total amount of miners (excavators), as their excavation rate is a fixed specification.
- Extracting the volatiles by the individual miner.
- Storing all the volatiles together in pressurised tanks on board the miner.
- Transferring the volatiles to lunar transfer rovers, which will carry the volatiles in their own pressurised tanks to the processing facility near or in the base.

This section has the goal to come up with the fixed values for the operations used in the plans proposed in the sections hereafter. Tab. 4.24 summarises all the relevant aforementioned characteristic, together with some new characteristics. Note that these values account for the 10 % energy demand. The most interesting value is the amount of Mark III miners required, 1634, without any contingency.

Table 4.24: Mining characteristics.

Operation/requirement	**Value**	**Unit**
Annual He-3 requirement	200	tons
Miner Excavation rate	1258	tons/h
Mining efficiency (grains <100 microns)	50	%
He-3 concentration	20	wppb
Regolith density	1800	kg/m^3
Mining depth	3	m
Miner processing rate	0.39	tons/s
Annual Regolith requirement (to be processed)	10 000 000	tons
Annual Regolith requirement (to be excavated)	20 000 000	tons
Required mining rate	634	tons/s
Equivalent surface useful mining area	3700	km^2
Equivalent surface total mining area	6500	km^2
Available total surface area MT	84 000	km^2
Time till depletion MT	7.36 years	years
Amount of Mark III miners required	**1634**	**units**

Miner Logistics

With the values of area and required miners determined, there is still a logistical challenge to be solved. The logistics of the miner vehicles will be discussed first in this section, after which the required storage capacity will be shown. Lastly the transportation of the raw mining materials to the base will be explained.

To fulfil 10% of the global energy demand, an area of 3700 km^2 needs to be mined per year, as shown in the previous section. For the operation plan, a first proposition would be that one square region will be mined with the lunar base at the centre. This results in a square region with the length of the sides of about 61 km to be mined. The miners would be subdivided among four square regions around the base, which is shown at Fig. 4.22. The miners would then drive to the outer boundary of the region and back to a point where they can unload their storage tanks (this might be at the base or on the transport vehicle's path). To mine this area it was calculated that 1634 miners are needed, based on the Mark III excavation rate of 0.39 tons/s.

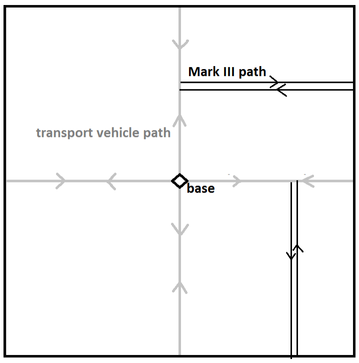


Figure 4.22: First proposition for mining operations.

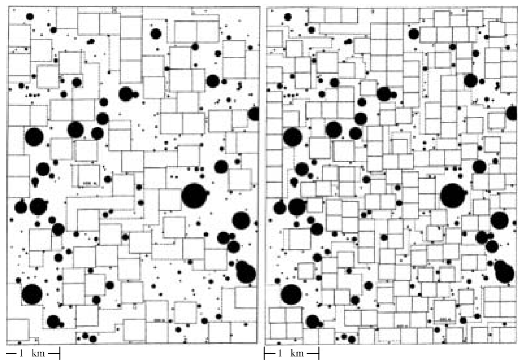


Figure 4.23: Example of squares mapping process.

This proposed approach has two restrictions/bottlenecks:

- **Mark III velocity:** The velocity of the Mark III is too low to get to the outer region of the mining area and back to the base before the storage capacity of the mined volatiles is reached. In practice this means that the mineable area is smaller than required and the amount of He-3 extracted will not be sufficient.
- **non-mineable areas:** The Moon is full of craters, ridges, rocks and other non-mineable features. Although the surface of the Mare Tranquilitatis is smooth relative to the other lunar regions, it is still unavoidable that many non-mineable spots are located in the determined mining area. This again leads to a lower quantity of volatiles and thus He-3 which is mined in this area.

A solution to these issues is to use multiple smaller mining regions. The base will be established at a central location with respect to these mining regions. In a lunar map the non-mineable areas can be marked out, such that the areas which are large enough to be mined can be discovered. For the Mare Tranquilitatis this process was performed on a small scale by Cameron and Kulcinski [123]. They excluded regions where craters of 24 m and larger diameter are present. The result of the process is shown in Fig. 4.23, where the left side of the picture shows the possible mining for 400 m x 400 m squares and the right side shows the same for 300 m x 300 m squares. Choosing 300 m x 300 m yields a usable area of 57% of the total area (see Tab. 4.24).

Storage capacity

After the regolith is excavated and the volatiles are extracted, the gathered volatiles need to be stored as well. The Mark III miners have 20 storage tanks each, which have a total mass of ten tons altogether. For gaseous substances pressurised at 20 MPa a structural mass fraction is derived from existing tanks with volumes around 600 L [130], which results in a storage capacity of 5340 kg of volatiles for one miner. A miner obtains about 127 kg/h of volatiles based on the mass fractions of volatiles in regolith which are shown in Tab. 4.21.

The time after which the tanks of the miners are full, and the tanks need to be transported back to the base, is calculated to be 42 hours. As the mining needs to continue, LTRs will be used to transport the gathered resources from the mining areas to the base. In this case the transport vehicle is similar to the ATHLETE rover covered in Sec. 4.8.3.1. The amount of LTRs will be driven by the time it takes to fill up the tanks (also including the "docking" and "un-docking" time).

Transport Vehicles

As it is not possible to have a base in each of the mining regions, transport vehicles for the extracted lunar resources are needed. The maximum range of these vehicles is determined by the assumption that for one operational year of mining, only one base is available. For a year of mining operations over 12000 square mining areas need to be mined. These sections are not directly connected as explained earlier, as 43% of the area is non-mineable. So in the worst case the ATHLETE transport vehicle first has to cross these non-mining regions before it reaches the Mark miner vehicle to transport the volatiles back to the base. This means that the total area of the lunar operations is around 6 500 km^2 instead of the 3 700 km^2 (which is the effective mining area).

To approach the average travel time of a LTR, it is assumed that the total amount of 300x300 meter squares are equally distanced from each other with a packing factor of 57%. These will be modelled as points for the sake of simplicity. If the base is assumed to be in the middle of the complete mining area, one can divide the area in four quadrants and then determine the average distance from the base of all points in the quadrant. This is done by evaluating the following equation:

$$d_{avg} = \frac{\Delta d}{N} \sum_{i=1}^{\sqrt{N}} \sum_{j=1}^{\sqrt{N}} \sqrt{i^2 + j^2} \tag{4.17}$$

In this equation, N denotes the amount of 300 by 300 m^2 in one of the four quadrants. For the 200 ton requirement, this would be 10288 squares on an area of 40.3x40.3 km. This yields a distance between the points of $\Delta d = \frac{40.3}{\sqrt{N}} = 403$ m. Evaluating this equation in Matlab yields an average distance of approximately 77% of the length of one quadrant. This is the same for any arbitrary square with an arbitrary amount of points equally distributed. This means the solution is applicable to any size of mining area, yielding an average distance between miner and base of 30.8 km for an annual demand of 200 tons (3 703 km^2 useful area to be mined).

The velocity of the ATHLETE transport vehicle is planned to be 2 m/s at the lunar surface [131], which allows the ATHLETE to travel average distance from miner to base approximately five times until the tanks of the Mark III miners are full. Through this it can be stated that one LTR can be assigned to five miners. The actual results for the total number of vehicles is shown in Tab. 4.25. When taking into account the redundancy, the total amount of TRLs will be about 400. The redundancy is included because the lunar vehicles demand significant maintenance and the average distance travelled might become longer as the theoretical path is likely to be obstructed by rocks and craters. The time to transfer the lunar raw material from the miner to the base will also be slightly higher, as the storage loading and unloading is not taken into account in the analysis.

Table 4.25: Values on the amount of vehicles needed.

Area to be mined	3700 km^2
Amount of Mark III miners (continuous operations	1 634
Amount of Mark III miners (including redundancy)	2 000
300 m x 300 m areas to be covered per miner	7.5
Average distance from base to mining area	30.8 km
Required ratio LTR:miner	1 : 4.88
Amount of LTRs	335

Long-Term Operation

The previous calculations were based on mining operations lasting for one year. To secure that the project returns on its investment, the mission has to be designed for longer duration. When the lunar operations are designed for five years of operations the mining area would be simply too large to be covered by the vehicles. Therefore lunar bases have to be set up elsewhere. This does not necessarily mean that the same production costs have to be made again. If multiple bases are set up at the same time, and one fifth of the originally allocated miners is placed near a base, the same amount of regolith will be processed in five years, instead of in one year. Because of this issue on how many miners should be assigned to one base, the mission duration needs to be defined before the operations take place. If it is designed for the 0.1% or 1% world energy demand requirement, the concept might stick to only one base. It is furthermore expected that the depletion time for the mineable regolith situated in the Mare Tranquilitatis amounts to exceed seven years.

4.8.4 Power Supply

This section gives an overview of all the different power sources that were considered for the lunar base. Firstly, the power requirement will be outlined. After that, an extensive summary is presented of the different options that were considered. After having considered all options, the final proposal will be presented for delivering the power for all lunar operations.

The power requirement on the Moon is mainly dictated by the mining activities themselves. It has been found that an expected 1634 miners will be constantly operating at 12.7 MW peak power. This yields a 20.7 GW power requirement considering all the miners. The base power requirement, which is in the order of 100-500 kW (or even 1 MW when a large amount of computer power is required, i.e. for the automation of operations) is therefore negligible and will not be further elaborated upon.

This comes down in the end to a total power requirement of 25.3 or 50.6 GW (including roughly 20% redundancy), depending on whether the miners will be constantly operating or only during the lunar days. In addition to this, there are also losses due to Radio Frequency beaming (RF beaming). It is assumed that these losses account for an efficiency factor of 0.65 (roughly 0.8 for both sending and receiving losses) [132]. This yields a power requirement of roughly 38.9 GW continuous or 77.9 GW during daytime.

It should be noted that the driving criterion in this trade-off is the specific power of the respective source.

4.8.4.1 Solar Cells

The first suggestion is the most conventional option, solar power. Firstly it is worth noting that solar cells are expected to significantly increase in performance over the coming decades. ONeill expects solar cell efficiency to approach 50% efficiency, corresponding to an areal power of <600 W/m^2 and a specific power of up to 1 766 W/kg for unsupported structures and 1 060 W/kg for arrays requiring structural strength, when attached to spacecraft for example [133].

If solar power is used, there will be a central location where all the power is collected by numerous arrays of solar cells. From there, the energy will be beamed to the miners. With the aforementioned values, one would need 3.6 km^2 of solar array weighing in at 1220 tons. What one should also consider is additional mass to prevent the accumulation of too much lunar dust and degradation of solar cells, requiring extra arrays as well. Assuming a 10% increase for maintenance and a 20% increase to cover degradation, 172 km^2 would be required or 58600 tons of solar arrays (1337 W/m^2). According to research done by Jordan and Kurtz, mean degradation of solar cells equals 0.8% per year, corresponding to 25 years until a decrease to 80% efficiency [134]. From 25 years onwards, an additional 0.8% has to be supplied of the total mass on an annual basis, corresponding to 470 tons of arrays.

4.8.4.2 Chemical Energy

One of the options to consider to supply energy to the miners would be to use propellant extracted from the volatiles found in the lunar regolith. Assuming that the power requirement of the base itself is negligible with respect to that of the mining activities, one can consider this power source purely from a per-miner perspective.

As has already been stated, the Mark III miner requires 12.3 MW at full power. Considering the expected hourly yield of volatiles, assuming continuous operation by 1634 miners, derived from the table presented by M. Gadja [123] which are presented in Tab. 4.21, one can reason H_2 to be a reasonable fuel, partly because of its high energy density, and partly because of its abundance. It is known that the energetic value of H_2 is 141.86 MJ/kg [135]. Considering a constant power requirement of 12.3 MW, one would need almost 312 kg of Hydrogen per hour to power one miner. This is much more than the actual 35 kg/h that is mined, omitting the fact that Oxygen is the actual limiting factor, of which almost 2500 kg/h would be required per miner and only 37 kg/h is mined (found using the stoichiometric equation $2\,H_2 + O_2 \rightarrow 2\,H_2O +$ [Energy], where H_2 weighs 2 gr/mol and O_2 weighs 32 gr/mol).

Considering only roughly 1% of the energy could be delivered assuming perfect energy conversion, this option is discarded.

4.8.4.3 Nuclear Fission

Fission is a controversial source of energy on Earth. Although it is a relatively clean source of energy in terms of direct emission into the atmosphere, it is still far from widely accepted. This is mainly due to the waste material coming from the process and the risk of catastrophic failures. Despite bringing these issues, and the issues of launching the materials from Earth to the Moon, it should still be seriously considered for this mission, as solar power for example might be proven impossible on such a huge scale.

In recent days, many research has been done on small nuclear reactors which are easily shippable. These are mainly being developed for remote areas and developing countries. Examples of reactors like these are the SSTAR (10-100 MW, 200-500 tons) [136], NuScale (45 MW Modules, $\sim$ 650 tons) [137] and Toshiba 4S (10 - 50 MW, unknown mass) [138]. Taking the best case scenario, a specific power could be achieved of 200 W/kg with the 100 MW SSTAR. One should consider though that it is very well possible, that if this study creates a need for developing a reactor with a higher specific power, this number increases significantly. Still, as further research on this topic is beyond the scope of this feasibility study, the specific power will remain at 200 W/kg for now. The SSTAR has the advantage that the reaction can be initiated and then goes on for 30 years, requiring no refuelling and little to no maintenance. The propellant requirement for this plant is 20 kg thorium per MW per year. This comes down to 778 tons of propellant per year at the proposed energy demand.

Using the aforementioned specific power, one can derive that one requires a power plant with a mass of about nearly 200 000 tons on the Moon to deliver 38.9 GW of power. Assuming the reactors can be refuelled on site and only new propellant and some repairs are needed, this 5400 tons is a one-time mass requirement, with an additional 50 tons annually in maintenance and propellant.

4.8.4.4 Nuclear Fusion

Although this study assumes the commercial availability of nuclear fusion, it can not be readily assumed that it is possible to have a fusion reactor available on the Moon. Currently, there are two possible developments in this industry. Firstly, there is Lockheed Martin's Skunkworks with their proposal for a 100 MW reactor that could fit on the back of a large truck [139]. This is assumed to correspond to a mass of 50 tons. This in turn yields a specific power of 2000 W/kg. Although this is an impressive figure, it is very speculative that this will actually happen.

A safer prediction would be that a reactor like ITER's will provide fusion power in the future. ITER weighs in at approximately 23 000 tons and is expected to deliver 500 MW while requiring 50 MW input. This means there is an additional power supply requirement to start the reaction. Omitting this 50 MW requirement for simplicity, the specific power for this option still only comes out to be 21.7 W/kg. One should also consider that the materials and systems used in a fusion plant are delicate and might not survive launch from Earth.

Although fusion is an attractive source of energy on Earth, it will be an entirely different story on the Moon. If no breakthrough discoveries are made, it is fair to assume that fusion will not be an option on the Moon. Therefore, this suggestion is ruled out in the search for the best source of power.

4.8.4.5 Conclusion

Four options to deliver the 40-80 GW power have been suggested above, of which only two remained as viable options. All considered options are summarised in Tab. 4.30.

Table 4.26: Summary of different power options.

Power source	Specific power	Total mass [tons]	Suggested?
Solar Power	1337 [W/m^2]	58186	Yes
Chemical Power	-	-	No
Fission Power	200 [W/kg]	194615	Yes
Fusion Power	21.7 [W/kg]	1807869	No

It is clear that solar power is the best option in terms of mass. Although the main issues will be the 172 km^2 of solar panels to be installed as well as to maintain them. Additionally, when choosing solar power, the available time to excavate is decreased by 50%. This would not only increase the amount of miners and rovers required, but also the complexity of the mission. Finally, a 172 km^2 area would be vulnerable to meteorites and lunar dust. Although the mass of the fission option is much higher (194615 tons against about 58000 tons), it is still favoured against solar power, due to above considerations.

4.8.5 Annual Required Energy

A total value for the required input energy into the whole Lunar Operations has to be provided, as to be able to contribute in quantifying the overall energy efficiency of the whole mission. For the Lunar Operations, this will be done by converting the power usage provided throughout this chapter into annual energy usage. Effectively, three elements will be counted: The miners, LTRs and the lunar base.

Table 4.27: Summary of energy input for continuous operations.

Element	Power need [MW]	Energy usage [TWh]
Miners (+RF loss)	38 920.0	341
LTRs	8.4	0.073
Lunar Base	3.4	0.028
Total	38 931.8	341

One can conclude that the total energy input into the lunar operations would amount to roughly 341 TWh. To put this in perspective: Considering the energy equivalence of 10% of the global energy demand to be at least 205000 TWh and at most 226000 TWh, the operations would require at least 0.15% and at most 0.17% energy input in order to sustain the power need for the lunar operations.

4.8.6 Scalability

Summarizing all previously obtained results for this mission, the scalability can be addressed. For each of the elements mentioned in Tab. 4.30, the results have been adapted for smaller scaled mission requirements: 1% and 0.1% of the global energy demand in 2040 (corresponding to 20 and 2 tons He-3 respectively per year).

4.8.6.1 Comparative facts

The scale of certain figures supplied in this chapter remain something yet to be grasped by the mind. Even more so, considering the lack of lunar infrastructure. Some comparative figures to put things in perspective:

- Focusing on the power generation on Earth by means of nuclear fission plants, for a moment: The most powerful Nuclear Power Plant (NPP) is situated in Chooz, France, and has an output of roughly 1.5 GW [140]. On Earth there are 437 NPPs in operation (and 68 under construction) [141]. These supply around 15% of the total energy in the world [140], reaching 372.5 GWe [141]. In order to only sustain the lunar operations, a total of 10.74% of the global nuclear energy generation would be needed on the Moon. Assuming an average output of 800 MW per NPP, 50 would be needed (or 35 of the most performing types, like Chooz).

- Similarly, considering the miner excavation rate. The biggest terrestrial bucket wheel excavator "Bagger 293" can move 240 000 m^3 per day [142]. 127 of these would be needed in order to sustain the yearly lunar regolith excavation requirement for this mission. As an additional detail to mention: one such machine has a mass of around 14000 tons.

4.8.6.2 Operation Scalability

Because the miners are scaled linearly and are the driving factor in the energy demand, the power demand can also be considered to scale linearly with the required He-3.

Table 4.28: Summary of values for mining operations of scaled mission.

Item	1%	0.1%	unit
Required regolith excavation rate	63	6	tons/s
Amount of miners	164	17	-
Amount of LTR	13	2	-
LTR autonomy	5	2	h
LTR range	36	11.4	km
Average distance base-mining area	9.8	3.08	km
Total mining area	650	65	km^2
Depletion time Mare Tranq.	74	737	y
Total volatiles mined per hour	21	2	tons/h

4.8.6.3 Power Scalability

Table 4.29: Summary of values for power supply of scaled mission.

Item	1%	0.1%	Unit
Power requirement miner	2.51	0.25	GW
Mass fission plants	$19.5 \cdot 10^3$	1950	tons
Fusion Power mass	$194.6 \cdot 10^3$	19460	tons
Solar Power mass	$5.82 \cdot 10^3$	582	tons

Table 4.30: Summary of energy input for continuous operations.

Item	1%	0.1%	Unit
Miners	$34.08 \cdot 10^3$	$2.19 \cdot 10^3$	GWh
LTRs	2.34	$76 \cdot 10^{-3}$	GWh
Lunar Base	28.03	28.03	GWh
Total	34.11	2.22	GWh
Share of global demand	15-17	10-11	[%]

4.8.7 Lunar Mining Segment Costs

In this section a cost estimation is made on the lunar mining operations. To put the size of the segment into perspective, the forementioned scaling of the annual He-3 requirement is applied and estimates are given on 0.1, 1 and 10% supply of the energy market in 2040. Due to the exotic nature of the mining segment, finding accurate estimations on the costs is difficult. The resulting possible inaccuracy of the values should be kept in mind when reviewing the results. This however does not reduce the importance of this cost estimation, since it gives an indication on the general cost-size of the segment and shows the areas that require additional research and optimization. Compared to other mission segments only little research has been done on the mining previous to this study. There is still much room for improvement in future studies, which should be done to make the mining of He-3 economically feasible.

4.8.7.1 General Assumptions

The cost estimations presented in this chapter are derived from the conceptual design of the lunar operations which have been explained in Sec. 4.8.3. Different systems for the different tasks have been used as a placeholder, such as the Mark III miner, that, for example, is used for mining the regolith. For the transportation rovers the ATHLETE-rovers were used as a placeholder. All input values on power usage, mass and required number of vehicles have been obtained from the Lunar Operations chapter. The consulted sources can be found there.

Since the cost of the mining is not only based on the costs of purchasing the different systems, this chapter also looks at the power required for the operations with the accompanying costs, as well as the total mass of the miners, rovers and power systems. As mentioned in Sec. 4.8.4 there are multiple options for the power generation, with solar power and nuclear fission as the most feasible options. Therefore both are presented in this section as well, with the cost impacts of both options compared. All costs have been calculated for fiscal year 2012 (FY12).

4.8.7.2 Power Supply

The two different power generation options relate to two different economic values. The results for both options in terms of power are shown below, while their implications in terms of purchase-, delivery- and operational costs are presented in the subsequent sections.

Fission Energy: In case the the choice is made to use nuclear fission as a power source, the mining operations can be continued regardless of solar eclipse. All operations will be continuous. The mining operation will require a certain amount of miners and rovers, which all need to be powered by the nuclear reactors. The miners require 350 kW each for the mining of the regolith, with additional 12.3 MW per miner needed for the heating and extraction of the volatiles. This brings the power requirement per miner to 12.65 MW, provided by energy beaming. With 2000 miners operating year-round to extract the required 200 tons of He-3 and an assumed beaming efficiency of 65%, this yields a total required power of 38.9 GW. With 17 kW required per LTR the 400 rovers add 6.8 MW to the power requirements. Thus, their power requirement is negligible. Since micro-nuclear reactors (with the SSTAR as a placeholder) are considered, which currently are predicted to deliver 100 MW per plant, a total of 390 fission plants is required. The micro-reactors are currently still in development, so cost estimations on both purchase and operations are difficult to find or not available at all.

Solar Energy: The second option for the power supply of the mining operations is the use of solar panels. For this, multi-junction panels are proposed, as they are expected to improve their efficiency towards 48% in the far-term future, while achieving lower mass and costs. Specific weight and area per unit of power are expected to be 0.7445 tons/MW and 2195 m^2/MW, respectively. The downside is that energy production is limited to 90% of the time during the lunar day, which in turn lasts for half a month. To mine the same amount of He-3 an increased number of miners and rovers has to be operational. Expressed in numbers this means that a total of 4000 miners are needed, assisted by 900 rovers that transport the extracted volatiles. Note that the required amount of miners uses a different redundancy than the amount of rovers. From the given energy consumption of said miners and rovers, a total of 77.85 GW of power is needed during the operational time of the lunar day. From the expected specific weight and area per unit of power, this corresponds to 171 000 000 m^2, or 58 200 tons of solar panels.

4.8.7.3 Purchase and Production

Both energy scenarios come with a different number of vehicles, as stated in the previous section, which all have to be purchased or produced. Since the TFU (Theoretical First Unit cost) will be much higher than the N^{th} unit cost, a learning curve has been applied as given in Eq. 3.4 in the cost conversion section (see Sec. 3.4). The results are given below.

Purchase and Production for fission-powered Scenario: The required number of vehicles and power plants for the fission energy scenario are the following: 2000 miners, 400 rovers and 390 fission plants. The proposed miner concept is the Mark III miner of the University of Wisconsin. Unfortunately, no cost estimations can be found on the concept, so a comparable estimation is used. The Mark II miner is the precursor of the Mark III and has the same mining capacity, with an estimated cost of 90 M€ [143]. For the production of 2000 miners a learning curve slope of 85% is applicable, bringing the total production costs to 30.3 B€.

For the transportation of the volatiles a system similar to the ATHLETE rover was proposed. It is designed for large payload transportation on the Moon, which makes it relevant for the mission. However, it also is a rather exotic concept at this moment. No cost estimations could be found. From a comparable ATHLETE-based Martian rover the cost estimation has been applied. The costs were estimated at 75 M€ per rover [144], which together with a learning curve slope of 85% accumulates to 7.4 B€.
A total of 390 nuclear plants (100 MW each) are required. Micro nuclear reactors like the SSTAR are expected to provide a very cost- and mass-efficient source of fission energy. The cost per power plant is currently estimated at 20 M€ [145], which, using the 85% learning curve slope, accumulates to 1.9 B€.

The summarized purchase and production values for the use of fission energy have been given in Tab. 4.31, including the down-scaled results for the supply of 0.1% and 1% of the energy demand. Since the estimated costs for the different systems have been obtained from different reference concepts, which are not necessarily completely applicable to this mission, a column with 'reliability' has been added to denote how accurate the estimations are considered to be. Note that because of the exotic nature of lunar mining operations, at this point a low reliability of cost estimations for most elements is inevitable. The reliability is ranked with Low, Medium and High.

Table 4.31: Purchase and production costs for fission-powered mining operations.

	Annual energy demand scenarios [B€]			**Reliability**
	0.1%	1%	10%	
Miners	1.14	5.20	30.3	Low
Rovers	0.14	0.66	7.4	Low
Power plants	0.07	0.45	1.9	Low/medium
Total	**1.35**	**6.31**	**39.6**	

Purchase and Production for Solar-powered Scenario: As stated before, in case solar panels will be used to power the mining operations, an increased number of miners and rovers is required for the same output in He-3 per year. With 4000 miners active, the purchase and production costs accumulate to 52 B€, with an 85% learning curve slope. To transport the mined volatiles, 900 rovers are needed, which have a total cost of 14 B€ with the same learning curve as before.

Finding any reliable numbers on the expected costs of far-term solar panels proved to be a great challenge. From personal correspondence with a space solar cell manufacturer an estimation could be obtained to include in the cost estimations [146]. For one m^2 a price of 58.5 k€ is expected. Given the very large area that should be covered with solar panels, applying the previously mentioned learning curve would reduce the average cost in a too high manner. This would yield average prices that would be highly unlikely, since there is a certain lower limit to the production-cost of the cells. This limit is expected to be 36 k€/m^2, which yields a price of 36 B€ /km^2. With an area of approximately 171 km^2 this would result in a total cost of 6 156 B€.

The total purchase and production costs have been summarized in Tab. 4.32. The reliability of the results has been evaluated.

Table 4.32: Purchase and production costs for the solar-powered mining operations.

	Annual energy demand scenarios [B€]			**Reliability**
	0.1%	1%	10%	
Miners	2.05	8.8	52	Low
Rovers	0.21	1.3	14	Low
Solar panels	61.56	615.6	6,156	Medium
Total	**63.82**	**625.7**	**6,222**	

4.8.7.4 Delivery

A major part of the set-up cost of the mining segment is the delivery of the systems to the lunar surface. Because of the large scale of the operations and the corresponding large masses involved, the need for a low-cost precursor transportation system becomes evident. Estimations on the cost of getting payload to the Moon (in €/kg) are differing significantly and seem to be contradicting in parts. As Schmitt states in his book 'Return to the Moon' [143], the cost/kg may range from a maximum of 53,000 €/kg (based on previous missions) to a possibly achievable target cost of 2,700 €/kg. The maximum cost can be considered as a very pessimistic upper limit of the costs, given that technological improvements over the past decades and the decades to come should be able to lower the costs. The target value of 2,700 €/kg that Schmitt considers *achievable* may arguably be a rather optimistic view, especially considering expected costs of future HLLV. The Falcon Heavy for example is expected to achieve a range of 4,500 - 7,100 €/kg to reach V_{esc} [147], so the cost of reaching the lunar surface will be slightly lower. These values show the need for a very low-cost precursor option to make the mission economically feasibly, as the following subsections will show.

Delivery of Mining Systems for the fission-powered Scenario: The estimations on the delivery costs depend on the number of vehicles and systems active for the two scenarios. When fission power is used, the total delivery masses are as follows, based on the mass estimations for the different systems.

In total 2000 miners are required, with a mass of 10 tons for every Mark III miner estimated. This will accumulate to a total of 20,000 tons for the miners. The ATHLETE rover is expected to have a mass of 2.3 tons, so with 400 required rovers, 920 tons have to be delivered. The SSTAR power plants are currently estimated to have a mass of 500 tons per reactor. Since 390 reactors are required in order to provide power for the mining operations, a total of 195,000 tons have to be delivered to the lunar surface.

The results are summarized in Tab. 4.33. The cost range of 2700 - 53000 €/kg has been applied to express the delivery costs of the required masses. As stated before, it is a very broad range based on expected developments. This range is mainly indicatory of the necessity of obtaining a low-cost precursor option. The reliability of the numbers is higher than for the purchase and production costs, since the mass estimations are more accurate and reliable and the actual cost/kg which will be applicable will most certainly be within the presented range.

Table 4.33: Delivery costs for the fission-powered mining operations.

	Annual Energy Demand Scenarios [B€]			**Reliability**
	0.1%	1%	10%	
Miners	0.54 - 10.6	5.4 - 106	54 - 1,060	Medium
Rovers	0.01 - 0.24	0.1 - 1.6	2.5 - 49	Medium
Power Plants	5.27 - 103.4	52.7 - 1,034	526.5 - 10,335	Low/Medium
Total	**5.82 - 114.24**	**58.2 - 1,141.6**	**583 - 11,444**	

Delivery of Mining Systems for the solar-powered Scenario: For the solar power system the same approach as for the fission power system is applied. However, the mass estimations are based on specific power (W/kg), the values for which are applicable to future solar panels that will be used, as explained in Sec. 4.8.3. With 77.85 GW required, the total mass of the solar panels is thus estimated at 58,200 tons.

The 4000 miners have an accumulated mass of 40,000 tons, while the 900 rovers add up to 2,070 tons in total. The summarized results are shown in Tab. 4.34.

Table 4.34: Delivery costs for the solar-powered mining operations.

	Annual Energy Demand Scenarios [B€]			**Reliability**
	0.1%	1%	10%	
Miners	1.08 - 21.2	10.8 - 212	108 - 2,120	Medium
Rovers	0.02 - 0.37	0.18 - 3.5	5.6 - 110	Medium
Solar Panels	1.57 - 30.9	15.7 - 309	157.1 - 3,085	Low
Total	**2.67 - 52.5**	**26.7 - 525**	**270.7 - 5,315**	

4.8.7.5 Maintenance and Operational Costs

Obtaining numbers on the production costs of the different mining systems has proven to be very difficult. Determining maintenance and operational costs of the systems decreased the reliability of the numbers to such an extent that it was decided to estimate these values in a different manner. The maintenance and operational costs of the different vehicles and the solar panels are addressed in a redundancy factor that has been included in the required amount of systems needed. This redundancy and with this the maintenance and operational costs have therefore already been determined in the previous sections. This is a valid approximation, as maintenance and operational costs mainly concern spare parts, for which the redundancy in the number of required miners, rovers, and power plants accounts.

The operational costs of the nuclear plants, however, were retrievable from the fact that micro nuclear reactors are predicted to have lower operational costs than current large-sized reactors [136]. Those have operational costs of approximately 5 - 10% of their production costs [148][149]. When applying the same percentage on the SSTAR power plants, an annual operational cost of about 1 - 2 M€ per reactor is a conservative estimate. The maintenance and operational costs of the fission reactors per energy demand scenario are shown in Tab. 4.35.

Table 4.35: Maintenance and operational costs for the nuclear reactors.

	Annual Energy Demand Scenarios [M€]			**Reliability**
	0.1%	1%	10%	
Power Plants	0.4 - 0.8	3.9 - 7.8	39 - 78	Medium

4.8.7.6 Conclusions and Recommendations

The different cost estimates for the two power source scenarios have been summarized and are shown in Tabs. 4.36 and 4.37 for the nuclear and solar powered scenario, respectively. From the results different conclusions can be drawn.

Table 4.36: Total estimated costs for the fission-powered mining operations.

	Annual energy demand scenarios			Reliability
	0.1%	1%	10%	
Purchase and production [B€]				
Miners	1.14	5.20	30.3	Low
Rovers	0.14	0.66	7.4	Low
Power plants	0.07	0.45	1.9	Low/Medium
Delivery [B€]				
Miners	0.54 - 10.6	5.4 - 106	54 - 1,060	Medium
Rovers	0.01 - 0.24	0.1 - 1.6	2.5 - 49	Medium
Power plants	5.27 - 103.4	52.7 - 1,034	526.5 - 10,335	Low/medium
Total fixed costs [B€]	**7.8 - 115.6**	**64.5 - 1,148**	**622.6 - 11,484**	
Maintenance and Operations [M€/year]				
Power plants	0.4 - 0.8	3.9 - 7.8	39 - 78	Medium

Table 4.37: Total estimated costs for the solar-powered mining operations.

	Annual energy demand scenarios			Reliability
	0.1%	1%	10%	
Purchase and production [B€]				
Miners	2.05	8.8	52	Low
Rovers	0.21	1.3	14	Low
Solar panels	61.56	615.6	6,156	Medium
Delivery [B€]				
Miners	1.08 - 21.2	10.8 - 212	108 - 2,120	Medium
Rovers	0.02 - 0.37	0.18 - 3.5	5.6 - 110	Medium
Solar panels	1.57 - 30.9	15.7 - 309	157.1 - 3,085	Low
Total [B€]	**66.53 - 116.29**	**652.38 - 1,150.2**	**6,492.7 - 11,537**	

Although cost estimations for exotic missions preferably are to be given in ranges, this has shown to be difficult due to the lack of research that has been done on lunar mining by the scientific community. Most information that is found is still very preliminary. It thus is in general difficult to derive financial numbers from the scarce information available. To give an indication on how accurate the estimations are, the reliability of the different numbers has been assessed in all the different subsections. This reliability is based on the credibility of the sources the numbers have been obtained from, the amount of research that has been done on the subjects by the scientific community and the assumptions that have been made while using the numbers. That the overall reliability of the numbers presented in this chapter is rather low does not imply that the numbers are incorrect. It does show, however, that the numbers should not be applied blindly. The overal low score on reliability is due to the fact that lunar mining is a field of research that has not been adressed sufficiently to obtain accurate estimations.

The main drivers of the high costs come from the need to have a very large amount of vehicles on the lunar surface in order to obtain the required amount of He-3 per year. Since the individual vehicles require a high amount of energy, the costs are increasing even more due to the need of more power sources and the corresponding extra mass involved. It is evident that in order to achieve a reasonable financial cost envelope for the lunar mining segment, significant research should be done to provide accurate concepts for the different systems. The number of required vehicles should be decreased. The miner design should be made more power-efficient to decrease the required power sources and achieve a better mass budget. In addition, a low-cost mode of transportation is required to have all systems of the mining segment delivered to the lunar surface at the lowest possible price. Even if the mining vehicles and power sources can be decreased significantly in terms of mass, the precursor transportation will still be the main driving cost factor (See Sec. 4.10). There are great challenges to be overcome in the mining segment, but given the low amount of research that has been dedicated to it so far, there is still large room for improvement.

4.8.8 Lunar Base Life Cycle Cost

As it is decided to have a manned mission, operating from a lunar base, is has to be determined what the costs will be to build such a base. High costs might be involved, as an equivalent mission has never been performed before. It is therefore difficult to predict what the costs of a lunar base will be. The costs will consist of Research & Development (R&D), construction, operations, maintenance, crew & supplies and disposal. This section shows how the cost for each of these aspects has been obtained. First general assumptions are addressed. More detailed assumptions and guesstimates are elaborated per section, to approach the costs of a lunar base as close as possible to reality. Also a summary, discussion and recommendations are included.

4.8.8.1 General Assumptions

In order to estimate the costs, a few general assumptions have been made. First of all there are three bases considered; a lunar outpost of 50 tons, a minimum base of 75 tons and a maximum base of 1000 tons. In a lunar base cost analysis led by Peter Eckart from the University of Berlin, it was assumed that the base would be at least 75 tons for a crew of eight people [150]. Assuming that the required base mass equivalent to one crew member is thus 9.4 tons, results in a capacity of five, seven and 105 crew members for the lunar outpost, minimum base and maximum base respectively. Furthermore, the majority of the costs is derived from the reference lunar base model as simulated by LUBSIM (this will be explicitly mentioned per section). LUBSIM is a simulation program with more than 350 parameter inputs, in order to simulate the masses and corresponding costs of all the lunar base facilities [151]. The most important assumptions of this LUBSIM reference model will be addressed. A life cycle of 40 years is assumed (10 years of installation and 30 years of operations). The initial base of LUBSIM has a capacity for 25 crew members and grows to a population of 100 crew members after 30 years operation. It is believed that LUBSIM simulates a base which is a good compromise between a rather small base and a lunar settlement [152]. Finally it is assumed that the construction of the base is built out of lunar regolith (using an inflatable or aluminium shell) [153]. All costs are derived to the fiscal year 2012. For every size of lunar base, a range in costs is provided as obtained from literature research and documented assumptions.

4.8.8.2 Research & Development

The R&D costs for the lunar base will be negligible compared to other costs of the base. It is assumed that R&D costs only consist of a feasibility study. This assumption had to be made, as there are no further numbers available on these costs. Development costs of the physical base and other lunar base facilities are included in construction cost estimates, as will be explained in the next section. Concerning the R&D costs, NASA has been used as reference [154]. NASA plans to perform a feasibility study on having a lunar outpost on the Moon with the same size as mentioned in the general assumption section. NASA plans to perform the feasibility study in six months at a cost of 256000 €. Based on this value, it is calculated what the feasibility costs are with respect to a single crew member. Assuming this is the average cost for a feasibility study, the feasibility costs of a minimum base would be 411000 €. For the maximum base it is assumed that the feasibility costs will not exceed further compared to a base of 50 crew members, yielding a cost of 2.5 M€. Although there are not many numbers available on feasibility study costs of similar missions, it can be stated that the reliability of assuming these costs negligible compared to other costs is high.

4.8.8.3 Construction

The construction cost is subdivided in various parts as done in the lunar base cost analysis by Eckart [150]; a precursor development and acquisition cost, the lunar base elements development and acquisition costs and transportation (installation) costs. The costs of development and acquisition are captured in one number. As acquisition costs are considered to be part of the construction costs, the development costs of precursor missions and lunar base elements are added to this section.

For the analysis as performed by Eckart, it was for the precursor missions assumed that Heavy Lift Launch Vehicles (HLLV), Lunar Excursion Vehicles (LEV) and LTR are used [150]. Furthermore he assumes in his cost analysis that in the best-case hardware development and acquisition costs are 110000 €/kg and transportation to LEO costs are 1100 €/kg. For the worst case, development and acquisition costs are 1.1 M€/kg and transportation to LEO costs are 11000 €/kg [150].

Precursor Development and Acquisition: With the assumptions as mentioned above, the development and acquisition of the precursor missions for a lunar outpost are estimated to be 10.9 B€ in the best case and 42.4 B€ in the worst case [155][150]. As the lunar outpost is 50 tons, the average cost per ton is 0.218 B€ and 0.848 B€ in the best and worst case respectively. Assuming this would be the average cost per ton for every base, yields a precursor mission cost of 16.3 and 217.9 B€ in the best case of the minimum base and maximum base respectively. In the worst case scenario this yields a cost of 63.6 and 848.1 B€ respectively.

Lunar Base Elements Development and Acquisition: The development and acquisition cost for the lunar base elements vary of course for every base, as the lunar outpost for example is smaller than the maximum base for 105 people. The estimates for these costs are based on the LUBSIM simulation and estimates made by Eckart [156][150]. In the best case, the costs for the lunar base elements of the lunar outpost, minimum base and maximum base are estimated to be 5.7, 8.5 and 114 B€ respectively. In the worst case they are estimated ten times as high, yielding 56,8, 85,5 and 1140 B€.

Transportation (Installation): Other than the development and acquisition costs as mentioned in the previous two sections, transportation costs to install the lunar base have to be taken into account. Based on TRASIM, a LUBSIM like simulation tool for transportation costs from Earth to the Moon (again assumed that the used vehicles are HLLV, LEV and LTR), and on the cost analysis of Eckart the total transportation costs could be estimated [157][158]. Taking the average of both cost estimations, the transportation costs for the lunar outpost, minimum base and maximum base yield 2.8, 4.3, and 11.4 B€ for the best case respectively and 28.5, 42.7 and 114 B€ for the worst case.

Having these values, the range of total installation costs for a lunar base can be expressed. The lunar outpost installation will cost at least 19.4 B€ and probably at most 127.7 B€. A minimum lunar base will cost at least 20.4 B€ and at most 133 B€. A maximum base would finally cost 343.4 B€ in the best case and 2102 B€ in the worst case. The reliability of these numbers is average. It is heavily dependent on for example the future price per kg to transport payload to the Moon and dependent on hardware costs. However, multiple sources are used to estimate the construction costs and together with the broad range of the estimations make this makes the number average reliable.

4.8.8.4 Operations

For the operational costs of a 40 year life cycle of the lunar base, simulation results of LUBSIMs lunar reference model are used to estimate the costs. This includes for example electricity and Lunar Liquid Oxygen (LULOX) processing [152]. LUBSIM calculated that operational costs would be 34.7 B€ in order to operate an initial base of 25 crew and to expand it to a 100 crew at the end of the life cycle [152]. Taking these costs as reference and the operational cost made by Weppler (on a lunar outpost equivalent base) [159], results in an estimation of costs of 8.7 B€ for the best case. The worst case is assumed to be 50% higher, yielding 13 B€. The best case is 25% of the cost estimation of LUBSIM. For the minimum base the costs are assumed twice as high, yielding 17.4 and 26.1 B€. To estimate the costs of the maximum base the costs of the lunar outpost could be multiplied by 21. However, it is assumed that a larger base will be relatively cheaper. It is assumed that the maximum base is 134% of the costs as calculated by LUBSIM, resulting in 46.7 B€ and 70 B€ for the best and worst case. The reliability of these numbers are low to average. LUBSIM is a far developed simulation tool, but only shows results of the so-called lunar base reference model (as explained in Sec. 4.8.8.1). Therefore the estimates for the various bases are based on rather thin air. The range of estimated costs makes the reliability low to average.

4.8.8.5 Maintenance

For maintenance again the simulation of LUBSIM has been used. The maintenance cost consists of imported equipment, spares and consumables [152][160]. LUBSIM estimates these costs at 21.3 B€. As there are no further sources available, the same percentages with respect to the LUBSIM simulation for the lunar outpost, minimum base and maximum base are assumed as for the operational costs. Maintenance costs will then be 5.3, 10.6 and 28 B€ respectively in the best case and 7.9, 15.9 and 42 B€ in the worst case. The reliability of these costs is also low to average, due to the same argumentation as the operational costs.

4.8.8.6 Crew & Supplies

For the lunar base a crew needs to be trained and salaries need to be paid. Again LUBSIM estimated these costs for the lunar base reference model, so for a crew increase of 25 to 100. This value, 4,13 B€, is assumed to be equivalent to a base of continuously 60 crew [152][160]. The equivalent cost for one crew member is then 69 M€. In the best case that yields a total crew & supply cost of 0.21, 0.37 and 5.4 B€ for the lunar outpost, minimum base and maximum base respectively. For the worst case again an increase in cost of 50% is assumed, yielding 0.32, 0.56 and 8 B€ respectively. The reliability score of the crew & supplies cost can be considered high. Although it is derived from a LUBSIM simulation, it is compared to the crew & supply cost of astronauts at the ISS.

4.8.8.7 Disposal

Space related disposals are mostly desired in order to avoid for example space debris. However, a lunar base would not cause debris. It is therefore assumed that the base will not be removed at the end-of-life of the mission. This means that there is no need to take disposal costs into account concerning the lunar base.

4.8.8.8 Summary

The results of all the costs elaborated in this section are summarised in Tab. 4.38. Also the total Life Cycle Cost (LCC) for the various bases in the best and worse case are shown in the table, as well as the average cost per year.

Table 4.38: Lunar base LCC.

	Lunar outpost (B€)	Min. base (B€)	Max. base (B€)	Reliability
R&D	*Negligible*	*Negligible*	*Negligible*	High
Construction	19.4 - 127.7	29.2 - 191.8	343.4 - 2102.4	Average
Operations	8.7 - 13	17.4 - 26.1	46.7 - 70	Low/Average
Maintenance	5.3 - 7.9	10.6 - 15.9	28 - 42	Low/Average
Crew & supplies	0.21 - 0.32	0.37 - 0.56	5.4 - 8	High
Disposal	-	-	-	-
Total LCC Cost	34 - 149.4	57.2 - 233.8	423.9 - 2223.2	
Avg cost per year	0.9 - 3.7	1.4 - 5.8	10.6 - 55.6	

4.8.8.9 Discussion & Recommendations

Although the idea of a lunar base is not new, research has been barely performed. Therefore the development of a lunar base is very preliminary and unknown. As a consequence, it is very hard to predict the costs involved to build a lunar base and most of the cost estimates are uncertain due to a lack of information. The assumptions made in this section are believed to approach reality as close as possible. However, further research would be required for a more detailed cost estimation. Research should be performed on the construction of a lunar base, as these costs have by far the largest contribution to the costs of a lunar base. Although the development costs of this analysis are partially included in the construction segment, it seems like a majority of the costs for a lunar base is due to the construction costs (more than 50% of total costs). If these costs can be reduced, a lunar base would be even more feasible. Next to that, also extra research should be performed on operations and maintenance, as the costs assigned to these aspects have the lowest reliability.

4.9 Docking Systems

The concept for docking procedures is described in this section, with a focus on the Space Dock, including Space Dock-CTTV docking. Generally, docking describes an active vehicle, while berthing denotes attachment of a passive vehicle. The docking procedure for LSAM-CTTV docking is explained in detail in Sec. 6.11. Using an active docking procedure uses more propellant and has more risk as compared to berthing. High risks for operation involving a crewed vehicle (Space Dock) should be avoided. Therefore, to transport payload between the CTTV to Space Dock and Space Dock to Skylon, a robotic arm in combination with a direct docking system is used. The LSAM will actively dock with the CTTV, using ATV-derived technology. This section describes the robotic arm and the direct docking system, followed by a summary of the final docking/berthing concept.

4.9.1 Robotic Arm

A variety of robotic arms is available in the space market. The five most promising robotic arms are the European Robotic Arm (ERA), Canadarm, Candarm2, Japanese Experiment Module Remote Manipulator System (JEMRMS) and the Special Purpose Dexterous Manipulator (SPDM) [161]. Tab. 4.39 shows the three most promising choices and their characteristics. Noticeable is that the maximum payload of Canadarm2 is considerably larger than that of the others. The primary reason for this that it was designed for docking (berthing) the Space Shuttle with the ISS [162]. Besides docking, it also provides a method for moving modules and performing repairs. For this mission, the minimum payload size (one canister) will be 10.77 tons. Thus ERA's, JEMRMS' and SPDM's payload handling capabilities will eliminate them as a choice. Canadarm2 is designed with an Automatic Collision Avoidance System (ACAS) [162] and supports payloads of 116 tons. This means it can be used for both payload transfer and berthing of the CTTV and Sklyon to the Space Dock. Taking this into consideration, Canadarm2 has been chosen for the Space Dock. No Canadarm2s are installed on the CTTV or the LSAM (which uses different robotic arms).

Table 4.39: Overview of commonly used robotic arms.

	ERA [163]	**Canadarm [162]**	**Canadarm2 [162]**	**JEMRMS [164]**	**SPDM [165]**
Length [m]	11.3	15	17.6	9.9	3.5
Diameter [m]	-	0.33	0.35	-	-
DOF	6	6	7	6	15
Mass [kg]	630	410	1800	780	1660
Payload [tons]	8	29.5	116	7	1
Speed					
... unloaded [cm/s]	10	60	37	0.6	7.5
... docking [cm/s]	-	6	1.2	2	3
... payload [cm/s]	-	6	15	2	3
Power					
... average [W]	475	-	435	-	600
... peak [W]	800	-	2000	-	2000

Canadarm2 (see Fig. 4.25) will perform two functions, namely berthing and payload transfer. During the berthing procedure, in worst case, the arm has to expand its full length 17.6 m unloaded with a speed of 37 cm/s (see Tab. 4.39) and dock it by retracting 17.6 m (max) at a speed of 1.2 cm/s. This entire process would take 25 min, of which 48 sec are to reach the target spacecraft and 1470 sec to berth it. The robotic arm must then either grab the Space Dock's payload or that from the attached spacecraft and bring it to its new location. Assuming again worst case, the robotic arm will travel unloaded to the payload by a distance of 17.6 m and move it loaded to the new location 35.2 m away (worst case 2×17.6 m), with a speed of 15 cm/s. The entire payload transfer would take roughly 5 minutes, of which 48 sec are to reach the payload and 235 sec to transfer to the new location. The Space Dock will have two robotic arms, such that a payload canister can be loaded and unloaded simultaneously. This is necessary as the payload bay of a to-be-loaded Skylon already is occupied by an empty canister, which in turn has to be moved to the payload bay that is occupied by the to-be-loaded full canister. Using a safety margin of 25%, a total of 55 min is required to berth and load/unload four payload canisters.

The grapple node is included in Candarm's total mass. Its counterpart (on the payload canisters, Skylon, and CTTV), used to grab the to-be-berthed spacecraft can be assumed to have a negligible mass compared with its structural mass. Canadarm2 development and construction cost was 480 M€ (600 M$ in 2011) [166]. As the arm no longer needs to be developed a total cost of 400 M€ is assumed.

4.9.2 Direct docking system

The direct docking system will be used to connect Skylon, CTTV, Space Dock, and LSAM, respectively. Note that loading/unloading is not done through the docking system but via robotic arms on the Space Dock on LSAM. There is a variety of docking systems available. Being the most up-to-date, the NASA Docking System (NDS) [167] was chosen, which will support both docking and berthing (see Fig. 4.24). Furthermore, this system is lighter than other systems. NDS can be used for both the active and passive vehicle and its mass is no larger than 340 kg [167]. The NDS system's height is 56 cm (retracted) and 102 cm (extended). The diameter is 172.7 cm. NDS' major components are: Soft Capture System (SCS), Hard Capture System (HCS), pressure tunnel and mounting flange structures, avionics controller and power supply box, external umbilical connectors, passive and active thermal control system, and the micrometeorite/orbital debris shield, as can be seen in Fig. 4.24 [167]. Only the Skylon and one docking port on the Space Dock will be equipped with a pressure-tunnel NDS. All other links do not require a pressure tunnel. The CTTV will be equipped with an additional system for berthing.

No cost estimations for NDS were found. It is assumed that the cost for the NDS will be roughly the same as that of the Russian Docking System (RDS) [168]. RDS costs about 14.5 M€ (M$14 in 2000) [168]. The Space Dock will have a total of three NDSs. CTTV and LSAM will have one NDS each. NDS is intended to have a lifetime of 15 years.

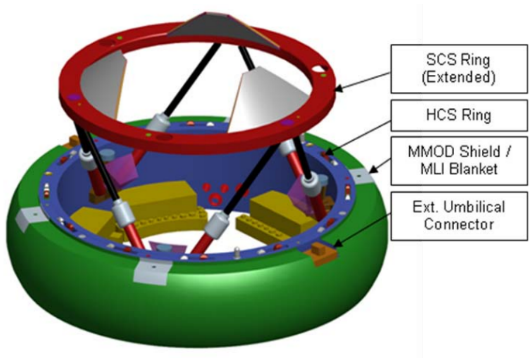


Figure 4.24: NASA Docking System [167].

Figure 4.25: Candarm2 robotic arm docking with the ATV [169].

4.9.2.1 Summary Docking/Berthing Procedure

Tab. 4.40 shows a summary of cost, mass, power, and time of the docking components. Skylon, the Space Dock, the CTTV, and the LSAM will be equipped with the NDS docking system. Furthermore, all three docking modules on the Space Dock will be equipped with two Canadarm2 robotic arms. Skylon and the CTTVs will be equipped with a grapple node, such that the robotic arm can securely attach to the vehicles for berthing. The estimated time for docking, loading, unloading, and undocking is 4120 and 3271 s respectively. Since the procedures are not sensitive to time, a generous safety factor of two is introduced, pushing the docking, loading/unloading and un-docking time to roughly 2 hours for each operation.

Table 4.40: Summary of berthing procedure components.

Space Dock		
No. Arms	6	-
No. NDS	3	-
Cost		
... per arm	400	M€
... per NDS	14.5	M€
... total	2.44	B€
Total mass	10.8	tons
Robotic Arm - Canadarm2		
Mass	1.8	tons
Length	17.6	m
Diameter	0.35	m
Payload	116	tons
Speed		
... unloaded	37	cm/s
... docking	1.2	cm/s
... payload	15	cm/s
Power		
... average	435	W
... peak	2000	W

Direct Docking System - NDS		
Mass	340	kg
height		
... collapsed	56	cm
... expanded	102	cm
Diameter	172.7	cm

Berthing Time		
... Attach to guest	48	s
... Dock/Un-dock guest	1470	s
... Load/Unload Payload	283	s
... Procedure 1*	4120	s
... Procedure 2**	3271	s

[*] Procedure 1 - Berth CTTV to Space Dock, load/unload payload (4 canisters) and un-berth CTTV.
[**] Procedure 2 - Berth Skylon to Space Dock, load/unload payload (1 canister) and un-berth CTTV.

4.10 Precursor Missions

Precursor missions are required in order to get all the lunar infrastructure up and running by the year 2040. Sec. 7.9 will show the Operations and Logistic Diagram (OLC) for both the recurring and the setup phase. The one portraying the setup phase is of interest here. The precursor missions should complete the setup diagram by doing:

- Setup of the Mission Control Centre and its components.
- Prepare and put into use dedicated launch and landing sites.
- Set up the space segment (Space Dock, lunar communication network, CTTV networking).
- Put masses in orbit/transfer orbit.
- Bring all required masses stipulated by the lunar operations with no damage to the lunar surface.
- Set up/deploy the lunar base, including all its complementary systems.

It is already clear that a complex system has to be dealt with. The chronology of operations is therefore not yet defined as it depends on a vast amount of conditions. Concurrent deployments and phasing of operations (varying percentages of completeness) will therefore be indispensable in order to conduct such a large-scale operation. In the following sections, the mass inventory will be set up and a cost will be estimated for the total operation.

4.10.1 Mass Inventory

Table 4.41: Summary of all masses to be launched in space and their corresponding costs.

Component	Mass [tons]	Launch cost [M€]
2000 lunar miners	20000	133000
400 lunar transfer rovers	936	6224
Lunar base	75	499
lunar power plants	196154	1304424
4 Lunar communication satellites	12	80
22 CTTVs	846	1269
1 Space dock	87	131
9 LSAMs	124	825
264 canisters	2843	4265
Total	221077	1450717

The launch costs have been found by assuming two values: 1500 €/kg for LEO destinations [147], and 6650 €/kg for LLO destinations [170][171]. The accumulated launch costs were found to be 1450.7 B€. These numbers do not account for mishaps, or failures, for which new launches will have to be undertaken. The acquisition costs are covered in Sec. 7.4 where the total LCC are spread over the operational years. Thus the reasoning as to why only the launch costs are shown here. Dedicated rocket carriers will have to be further developed in the future (to comply with unit mass/volume capacities) or additional research will have to be conducted. In Sec. 4.8.8 it was stated that the transportation development and acquisition would cost 64 B€ in the worse case.

4.11 Manned Missions

Considering that no infrastructure is yet present on the Moon, manned missions will have to be included in both the precursory and recurring phases (an overview of these phases can be seen in Sec. 7.9). Space robotics have yet to come up with major breakthroughs for supplying solutions for the scale and operational density involved in this mission. All operations will have to be undertaken autonomously in the long run, while humans should at most retain a control and mitigation function either from the Moon, Earth or both.

Table 4.42: Crew missions overview

	Value	Unit
No. crew at lunar base	3	-
No. crew per flight	3	-
LSAM est. total wet mass	50	tons

	Value	Unit
Rocket launches/year	3	-
Est. launch cost/year	998	M€
Est. one way transfer time	60-80	hours

In Sec. 3.3 it was established that humans should not be present on the lunar surface for more than 180 days. This value assumed no solar flare activity or travel time radiation exposure. Assuming that in the initial years of operation humans will be present continuously, a strict minimum of three crew replacement missions should be carried out successfully each year. This will allow for contingency in terms of effective radiation exposure.

In Sec. 4.8.3.1, the amount of crew needed for continuous operation and maintenance was fixed at three (given that fission reactors would power the operations). These three annual transfers will be carried out through relative quick transfer trajectories (Hohmann or free-return), compared to cargo spiral routes. These can be done by using either free return trajectories or not. All free-return trajectories have trans-lunar transit times between 60 and 80 hours. Similar trajectories will then be used on the way back, using the same LSAM vehicle between Earth orbit and Moon. It will not be reusable and the crew can access Earth by the use of capsules. This would mean that propellant will be needed for the way back, but the payload can be much smaller compared to the LSAM that carries He-3 canisters. Tab. 4.42 summarises the lunar crew missions.

The crew LSAM will bring both crew and some additional equipment/supplies with them to the lunar surface. For the crew LSAM the useful mass will be set at 10 tons. Today resupply missions to the ISS are capable of 7 tons (using the ATV as reference [172]). Besides, the Apollo lunar lander ascent module is roughly 5 tons. It is therefore a good first estimate to fix the payload around 10 tons, comparable to the "Altair" lander from the Constellation programme. Its total mass would be just under 50 tons, enough to include it as a payload for a Falcon 9 Heavy. Three of those launches would be needed per year. These would amount to 998 M €/year for launch costs, assuming 6650 €/kg (as explained in Sec. 4.11). The ΔV needed for trans-lunar trajectories using high trust is roughly 4000 m/s, where an additional 2000 m/s will be needed to perform landings on the lunar surface [60].

As the mission would be scaled down fewer crew is needed, which will drive the design for the LSAM again. On the contrary, it might not be a bad idea to keep the 3 crew even in lower scaled missions. This decision could for example enhance social contact of the crew, increase morale and add reliability in case of mishaps or fatalities, as to keep the productivity per flight mission high enough to complete the tasks set.

5. Detailed Design: CTTV

To continue on Sec. 4.6, the CTTV will be designed in full engineering detail in this chapter. First, the subsytem requirements will be stated for all the subsystems. Secondly, the transfer trajectory for the CTTV will be designed. This is followed by the design of all the subsystems. At the end a summary will be given including a discussion on the scalability. The design process will use the conceptual values defined in Sec. 4.6 as starting values for the design. Once the subsystems are designed an iteration will be performed. The values in this chapter are the final iteration values. An illustration of the CTTV can be seen in Fig. 5.1.

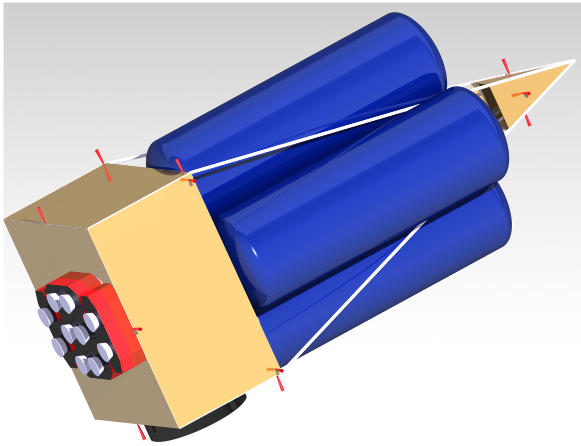


Figure 5.1: Illustration of the CTTV external layout.

5.1 Subsystem Requirements

The requirements for the subsystems are given in Tab. 5.1. The subsystem design will be performed to apply to these requirements and this will be evaluated with a compliance matrix in Sec. 7.1.

Table 5.1: Overview of all subsystem requirements for the CTTV.

Requirement-ID	**Requirement description**
CTTV-Prop-1	The propulsion system shall use readily available propellant from Earth.
CTTV-Prop-2	The propulsion system shall provide between 10 and 100 N of thrust.
CTTV-Prop-3	The propulsion unit shall be restartable.
CTTV-Prop-4	The propulsion system shall use a propulsion concept that will be at least ready by 2040.
CTTV-Struc-1	The structure shall provide structural support for the payload, propellant, and other subsystems.
CTTV-Struc-2	The structure shall sustain all loads acting on the structure over the course of the CTTV mission.
CTTV-Struc-3	The structure shall be mass-efficient.
CTTV-Struc-4	The structure shall have a safety margin of 1.5 included.
CTTV-Struc-5	The structure shall have natural longitudinal and lateral frequencies higher than frequencies of relevant reference vehicles.
CTTV-Struc-6	The structure shall provide load paths for the loads acting on the structure.
CTTV-Struc-7	The structure shall withstand longitudinal loads of -9g.
CTTV-Struc-8	The structure shall withstand lateral loads of -1.5g.
CTTV-Struc-9	The structure shall withstand longitudinal vibrations of 27 Hz.
CTTV-Struc-10	The structure shall withstand lateral vibrations of 7.5 Hz.
CTTV-Therm-1	Maintaining the temperature inside the spacecraft in the operational range of 263 - 298 K at all times.
CTTV-ADCS-1	The ADCS shall contain enough propellant for the mission, including a margin factor of two.
CTTV-ADCS-2	The ADCS shall include separate sensors for rough and precise attitude determination.
CTTV-ADCS-3	The ADCS shall include different types of sensors.
CTTV-ADCS-4	The ADCS's thruster plumes shall not damage CTTV or LSAM.
CTTV-ADCS-5	The ADCS actuators shall be placed such to minimise unwanted rotations.
CTTV-ADCS-6	The ADCS shall have a determination accuracy of 0.1°.
CTTV-ADCS-7	The ADCS shall have a full determination range (360°).
CTTV-ADCS-8	The ADCS shall have a control accuracy of 0.1°.

CTTV-ADCS-9	The ADCS shall have a full control range (360°).
CTTV-ADCS-10	The ADCS shall limit jitter to 0.1°.
CTTV-ADCS-11	The ADCS shall limit drift to 4°.
CTTV-ADCS-12	The ADCS shall have a setting rate of 10°/min.
CTTV-GNS-1	The location of the CTTV shall be known with an accuracy of 500 m.
CTTV-GNS-2	The CTTV shall have a position accuracy in LEO of 7 m during docking in LEO.
CTTV-GNS-3	The CTTV shall have a position accuracy of 15 mm during docking in LLO.
CTTV-Power-1	The power supply system should deliver a constant power of 1.2 MW.
CTTV-Power-2	In case of a power failure, the TT&C should obtain sufficient power to last 2 more months.
CTTV-TT&C-1	The TTC subsystem shall be able to communicate with the ground stations.
CTTV-TT&C-2	The TTC subsystem shall be able to communicate with the relay satellites.
CTTV-TT&C-3	The TTC subsystem shall be able to communicate with the Space Dock or CTTV during docking procedures.
CTTV-TT&C-4	The subsystem shall be able to transmit data at a rate of 400 kbps.
CTTV-TT&C-5	The subsystem shall be able to receive data at a rate of 40 kbps.
CTTV-C&DH-1	The system must be able to function redundantly for 1 year operation.
CTTV-C&DH-2	The system shall be able to withstand high irradiation levels (>1 Mrad) for its lifetime.
CTTV-C&DH-2	The system shall be able to communicate payload data from CTTV to the ground station.
CTTV-C&DH-4	The system shall be able to collect payload telemetry and household data from the CTTV..
CTTV-C&DH-5	The system shall be able to receive and parse commands from ground control.
CTTV-Dock-1	The docking system shall establish a structural connection with the CTTV.
CTTV-Dock-3	The docking system shall allow for transfer of onboard data.
CTTV-Dock-4	The docking system shall be compatible with LSAM and Space Dock docking system.
CTTV-Dock-5	The docking system shall be autonomously operable.
CTTV-Dock-6	The docking system shall allow transfer of additional payload (supplies).

5.2 Transfer Trajectory Design

This section shows the design of the transfer trajectory of the CTTV. First, it is determined where the CTTV enters the Sphere of Influence (SOI) of the Moon, defining the required Moon orbit. Then the transfer from LEO to the patch point and from patch point to LLO will be discussed. Also orbit phasing will be addressed. Finally, there applied V&V techniques and possible model improvements will be mentioned. A summary of the main results is given in Tab. 5.8.

5.2.1 Assumptions

- No perturbations act on the CTTV.
- The Moon orbit around Earth is assumed circular.
- The transfer from Moon to Earth has the same characteristics as the transfer from Earth to Moon.
- Δi in patch point to Moon orbit transfer is neglected.
- The required Δi is assumed constant over time.
- The SOI of Moon and Earth are perfectly circular.
- Earth and Moon are considered point masses.

5.2.2 Required Moon Orbit

First it needs to be defined to what Moon orbit the CTTV has to transfer to. It has been determined that the Moon orbit is a circular orbit at 100 km altitude with a 9° inclination, see Sec. 6.2. This will be the final orbit the CTTV has to attain. The orbit characteristics around the Moon have to be determined using the Moon SOI. First, the optimal orbit is chosen. Depending on the patch point location on the SOI, the resulting Moon orbit has unique characteristics. Eq. 5.1 estimates the required ΔV for a direct transfer between an elliptical- and circular orbit. Please note that extra ΔV is required if r_f is larger than 1838 km (radius Moon + 100 km), in order to decrease the circular orbit altitude to 100 km. Here, μ is Moon's gravitational parameter, r_f is the final orbit altitude and a is the semi-major axis. The required ΔV was plotted versus a and r_f and it was concluded that both variables should be as low as possible the lowest ΔV requirement. The perigee (r_p) will be at 1838 km altitude and the semi-major axis is dependent on the location of the patch point. The equation used is for a direct impulsive transfer, instead of the continuous-thrust spiral transfer. However, the direct transfer is considered to be representative to determine a preliminary ΔV.

$$\Delta V = \sqrt{\frac{\mu}{r_f}} - \sqrt{\frac{2\mu}{r_f} - \frac{\mu}{a}} \tag{5.1}$$

The Moon SOI is determined to be 66183 km using Eq. 4.11. Now, the required orbit depends on the orbital characteristics of the patch point. An illustration of the results can be seen in Fig. 5.2. The characteristics of this orbit are listed in Tab. 5.2. The orbit will change depending on the location of the patch point on the Moon SOI. It would be optimal to obtain an orbit with an pericenter at LLO altitude, the process to obtain this optimal orbit is described below.

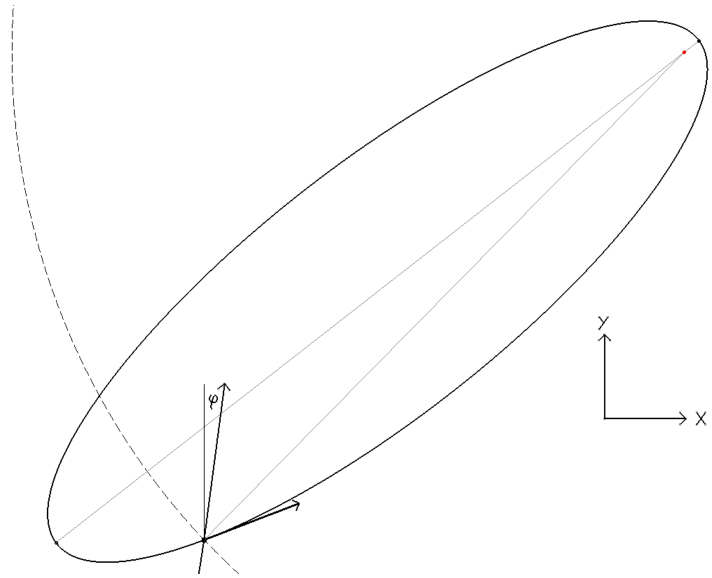


Figure 5.2: Elliptic Moon orbit once entering the Moon SOI (dashed line) with velocity vectors as arrows.

Table 5.2: Moon elliptic orbit characteristics.

Characteristic	Value	Units
Patch point distance	340083	km
Semi-major axis	39652	km
Pericenter	1838	km
Apocenter	77466	km
Eccentricity	0.9536	-

Table 5.3: Conversion of patch point velocity.

Characteristic	Value w.r.t. Earth	Value w.r.t. Moon	Units
Orbital velocity	1.0826	0.1566	km/s
Velocity angle ϕ	7.8	69.7	deg
Velocity x-direction	0.1468	0.1468	km/s
Velocity y-direction	1.0726	0.0543	km/s

The first parameter to be determined is the velocity angle as a function of the patch point, with respect to the x-axis (ϕ). It can be determined using trigonometry, resulting in Eq. 5.2. Here, a_{Moon} is the semi-major axis of the Moon orbit of 384400 km and r_{patch} is the location of the patch point with respect to Earth.

$$\phi = cos^{-1}\left(\frac{SOI^2 - a_{Moon}^2 - r_{patch}^2}{-2 \cdot a_{Moon} \cdot r_{patch}}\right) \tag{5.2}$$

The orbital velocity at the patch point can be calculated using Eq. 4.10, where μ is the gravitational parameter of Earth and r equals r_{patch}. The velocity in x- and y-direction is determined using the velocity angle ϕ. These two values are required to determine the new orbital velocity. Once the vehicle enters the SOI of the Moon, the Earth influence is neglected and the orbital velocity has to be converted to a system only including the Moon. This is done by subtracting the orbital velocity in y-direction from the orbital velocity of the Moon (1.0183 km/s). The velocity in x-direction does not change. This results in a new orbital velocity and a new velocity angle. The conversion for the orbit illustrated in Fig. 5.2 is given in Tab. 5.3.

The resulting semi-major axis of the orbit is calculated using Eq. 5.3. The equations are implemented in a Matlab program which determines these parameters as a function of the patch point (r_{patch}). The values that still have to be determined are the pericenter (r_p) and apocenter (r_a) distance, and the eccentricity (e). The velocity vector is always tangent to the orbit path with respect to Earth. Using this, the orbit can be sketched using CATIA. This sketch results in semi-major axis, and hence the pericenter and apocenter distance, fixing the orbit. Using an iterative process converging to a pericenter distance of 1838 km, the optimal orbit has been determined.

$$V^2 = \mu\left(\frac{2}{r} - \frac{1}{a}\right) \tag{5.3}$$

5.2.3 Earth to Patch Point Transfer

This section will model the orbit from LEO until the patch point. Using an out-of-plane thrust angle (β), a simultaneous inclination and semi-major axis change is achieved. The inclination change is dependent on the inclination of the Moon and the Earth axis tilt (i_E), which varies over time. The Earth axis tilt is about 23.5° and varies over a 41000 years cycle. Using an equation to determine the axis tilt in 2040, an angle of 23.43° with respect to the Earth was obtained (i.e. orbit of Earth around Sun) [173].

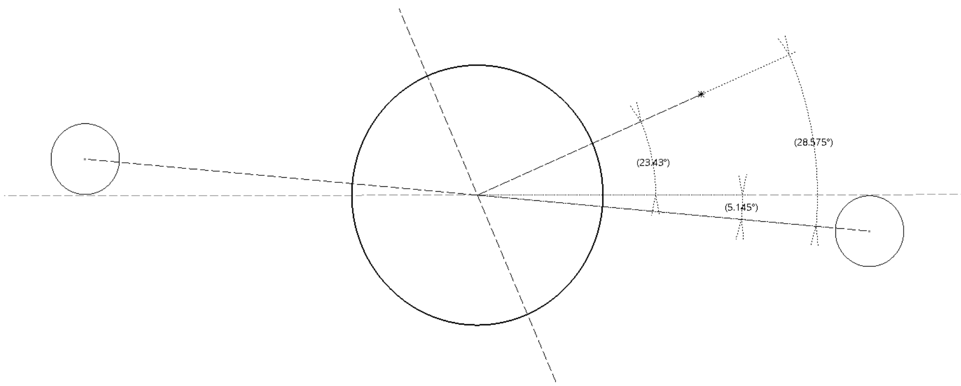


Figure 5.3: Required Δi (not to scale).

Once the patch point has been attained, the orbit should be in the same plane as the Moon orbit plane. Therefore, the difference in inclination has to be determined. To illustrate, see Fig. 5.3. The small circle is the Moon, whereas the larger circle represents the Earth. The inclination of the Moon orbit is about 5.145° with respect to the ecliptic [109]. However, the Δi of the Moon changes over time, which is the difference in inclination of the Earth equatorial plane and the plane of the Moon orbit. This difference is in the range of 23.43±5.145° [109]. For now the 'worst case' value of 28.58° is assumed as Δi for the transfer.

Once the initial orbit velocity (V_0) at 500 km altitude (7.613 km/s) and the final orbit velocity (V_f) at the patch point (1.083 km/s), have been calculated, the acceleration can be calculated using the mass flow, which is 0.0012 kg/s (Sec. 5.3). With this value and the initial mass, the momentary mass is calculated. This results in an average acceleration (a_{avg}) of $7.83 \cdot 10^{-4}$ m/s^2. Note that the initial (wet) mass (M_{wet}) is obtained using the dry mass (M_{dry}) of the spacecraft in an iterative process. The wet mass is dependent on the total ΔV which is a result of this model.

The next equations were acquired from a study on a low-thrust transfer to GEO [174]. In order to acquire a simultaneous inclination and semi-major axis change, an out-of-plane thrust angle (β) has to be applied. First, the initial out of plane thrust angle (β_0) is calculated using Eq. 5.4. This is the thrust angle that is required at the start of the transfer (i.e. when leaving the Earth orbit). Fig. 5.4 shows this angle, where the two dotted lines show two orbits and the circle is Earth. This angle is shifted (multiplied by -1) every time the CTTV is crossing the Earth orbit plane to keep the same inclination change.

$$\beta_0 = tan^{-1}\left(\frac{sin(\Delta i \cdot \frac{\pi}{2})}{\frac{V_0}{V_f} - cos(\Delta i \cdot \frac{\pi}{2})}\right) \tag{5.4}$$

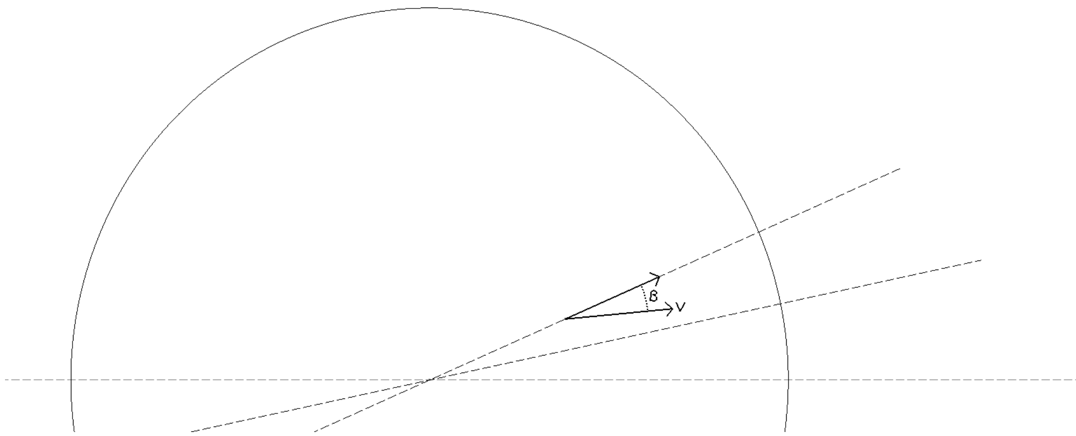


Figure 5.4: Out-of-plane thrust angle.

With the initial out-of-plane thrust angle and the initial- and final orbit velocity, the required ΔV to transfer from the Space Dock to the patch point (ΔV_1) can be calculated using Eq. 5.5.

$$\Delta V_1 = V_0 \cdot cos(\beta_0) - \frac{V_0 \cdot sin(\beta_0)}{tan(\frac{\pi}{2} \cdot \Delta i + \beta_0)} \tag{5.5}$$

There are values that can be determined as a function of time (t) in seconds. The velocity (V), out-of plane thrust angle (β), and inclination (i) can be calculated using Eqs. 5.6, 5.7 and 5.8 respectively.

$$V(t) = \sqrt{V_0^2 - 2 \cdot V_0 \cdot a_{avg} \cdot t \cdot cos(\beta_0) + a_{avg}^2 \cdot t^2} \tag{5.6}$$

$$\beta(t) = tan^{-1}\left(\frac{V_0 \cdot sin(\beta_0)}{V_0 \cdot cos(\beta_0) - a_{avg} \cdot t}\right) \tag{5.7}$$

$$i(t) = i_E + \frac{2}{\pi} \cdot \{tan^{-1}\left(\frac{a_{avg} \cdot t - V_0 \cdot cos(\beta_0)}{V_0 \cdot sin(\beta_0)}\right) + \frac{\pi}{2} - \beta_0\} \tag{5.8}$$

5.2.3.1 Results

With Eqs. 5.4 - 5.8, the transfer model can be programmed using both Matlab and CATIA. Additional equations calculating the radial distance (r) (see Eq. 4.10), and the true anomaly (θ) are used to determine the position in Cartesian coordinates in time. To obtain the position in Cartesian coordinates, a coordinate transformation is performed using $x = r \cdot cos(\theta)$, $y = r \cdot sin(\theta)$, $z = 0$, and applying a rotation of i about the y-axis. The results are shown in Fig. 5.5. It must be noted that at the start of the transfer, the inclination and altitude are increasing relatively slowly. This increases exponentially over time. In order to illustrate this, the inclination and out-of-plane thrust angle are plotted versus time in Fig. 5.6. The total Δi in this case is 28.58°, and the patch point is located at 340083 km. The altitude over time-plot is shown in Fig. 5.7.

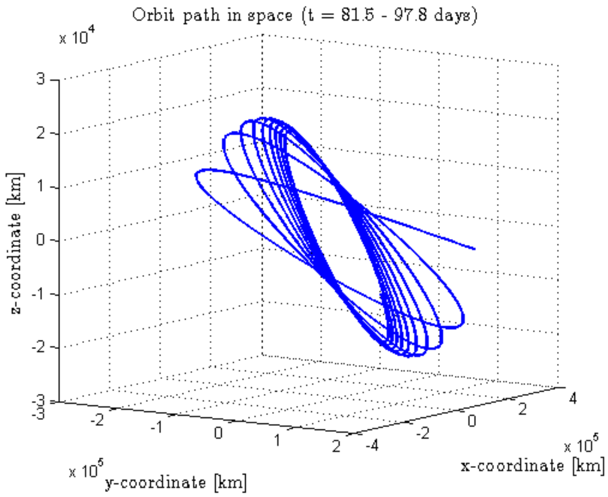


Figure 5.5: Orbit path in space for t = 81.5 - 97.8 days.

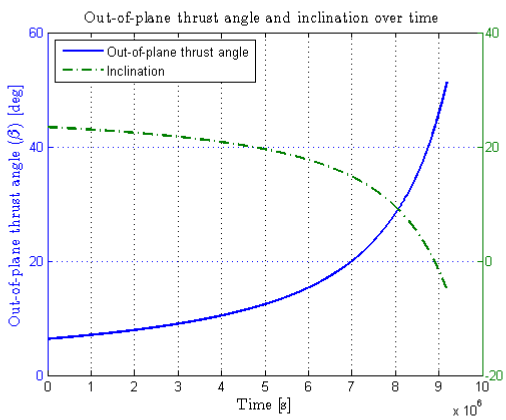


Figure 5.6: Inclination and β over time.

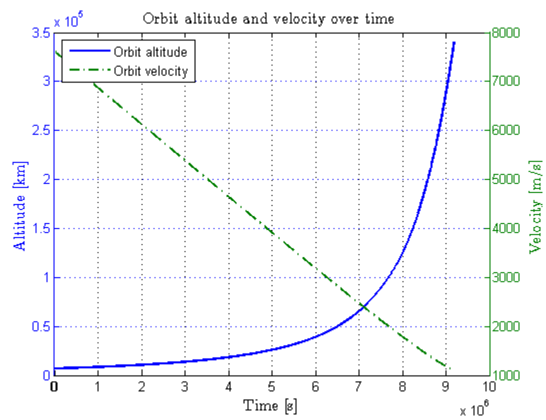


Figure 5.7: Altitude and velocity over time.

The results for the transfer from LEO to the patch point are listed in Tab. 5.4. The required ΔV is about 67% higher compared to a typical Hohmann transfer (ΔV=3.914 km/s) with no inclination change, though it requires half the propellant mass. This will result in lower recurring cost, as less propellant has to be transported to the Space Dock. The required ΔV for the Δi only is about 5.2% (358.14 m/s) of the total required ΔV.

Table 5.4: Transfer characteristics LEO orbit to patch point.

Characteristic	**Value**	**Units**
Required ΔV_1	6.88	km/s
Transfer time (TT_1)	106.41	days
Total Δi	28.58	deg

5.2.4 Patch Point to Moon Transfer

As the conditions at the patch point are now known, the orbit around the Moon can be considered. It is also given that the initial orbit is defined by a pericenter altitude of 100 km and an apocenter altitude of 77466 km. It should be noted that this apocenter altitude should be lowered to within the Moon SOI in one orbit, to prevent the CTTV from travelling back to Earth. In addition to that, applying continuous thrust would be impossible, as the CTTV would then crash into the lunar surface before reaching pericenter. Therefore, a different model has been developed, designed to determine the true anomaly θ range in which thrust is applied to sufficiently lower the first orbit. Consequently, this range of θ is then initially used to determine the travel time and required ΔV. When this is done, some variations are manually attempted in order to find the approach to optimise both travel time and ΔV. The optimisation process will not be performed to full extent.

Finding the optimal thrust area to lower apocenter from 77466 to 66138 km is done iteratively. It is known that the most efficient way to lower apogee is to perform manoeuvres only at pericenter. For continuous thrust this is not possible. Moreover, the farther the velocity change is performed from the pericenter, the less efficient the manoeuvre is. The model approaches the problem as follows. It evaluates one single orbit, divided into $2\pi \cdot 1000$ segments with constant $d\theta$. It then performs a step-by-step evaluation along the orbit. Firstly, the distance from the origin is determined, using the actual θ. This is done using Eq. 5.9.

$$r(\theta) = \frac{a \cdot (1 - e^2)}{1 - e \cdot cos(\theta)} \tag{5.9}$$

When this distance is known, the orbital velocity can be determined using Eq. 5.3. Also the travelled distance can be determined, hence the time per segment can be found using Kepler's Second Law of orbital motion. Changing the velocity by applying thrust will then cause the semi-major axis to change when using Eq. 5.3 again. The angles θ between which thrust is applied can then be found through trial and error. When multiple orbital periods are considered, the aforementioned sequence is redone for a new value of r_a. The pericenter distance (r_p) is always corrected at apocenter with an instantaneous thrust manoeuvre, hence it will be reset to 1838 km every orbit. Tab. 5.5 shows what values were considered for the first orbital period. More values than the final value are shown to illustrate the influence of different ranges. An important characteristic in the table is ΔV_{sp}, which denotes the efficiency of the used ΔV. It is defined as the altitude change in meters per m/s of ΔV. It should be noted that θ is measured from the pericenter, which means that the motion is performed between $-\pi < \theta < \pi$.

Table 5.5: Overview of iterative process to determine the optimal thrust range for the first phase.

θ **Begin** ($\times\frac{\pi/2}{1000}$)	θ **End** ($\times\frac{\pi/2}{1000}$)	**ΔV [m/s]**	**Δr$_a$ [km]**	**ΔV$_{sp}$ [1000 s]**	**Δr$_p$ [km]**
-1	1	0.0015	4.3	2879	0.1
-1000	1000	2.468	5746	2327	73.3
0	1000	1.252	3006	2400	0.62
-1000	0	1.28	3069	2386	72.8
0	2000	109	28401	260	0.62
-750	1250	2.49	6395	2140	47.15
-975	**1550**	**7.583**	**11408**	**1504**	**70.1**

The following criteria hold for the process. Firstly, the required Δr$_a$ is 11328 km, simply because the spacecraft will fly out of the SOI of the Moon again at apocenter if this is not achieved. This distance is determined using the apocenter value from Tab. 5.2 and subtracting the SOI radius of 66183 km. Secondly, the change in pericenter altitude is not allowed to exceed 70 km considering 30 km redundancy. Correcting this again in apocenter would require a 0.84 m/s velocity change. It must be noted that this correction is not incorporated in the table. This ΔV can be delivered by the 57 N thrust close enough to apocenter to assume an instantaneous manoeuvre. From the table, it can be concluded that a continuous thrust manoeuvre between $-87.75\,\text{deg} < \theta < 139.5\,\text{deg}$ is required to reach $r_a < 66138$ km. If this range is kept throughout the entire manoeuvre, it will yield a $\Delta V = 1.384$ km/s and $TT_2 = 110$ days for the entire transfer between patch point and LLO. Previously, a requirement was set for a transfer time between LLO and the patch point (and vice versa) of 50 days. This can be achieved without requiring too much extra ΔV. In a primary instance, the first 20 orbits will have an extended range of $-87.5\,\text{deg} < \theta < 171.0\,\text{deg}$, because the semi-major axis has to be decreased as fast as possible in the earlier phase as outer orbits take up the majority of travel time. With this setting, the apocenter is lowered to 11274 km in 32.67 days, using a ΔV of 495 m/s. After this, the rest of the journey will be performed with constant thrust settings, which can be varied to the demands of the mission, as can be seen in Tab. 5.6. A few typical results are shown to again illustrate the idea.

Table 5.6: Overview of iterative process to determine the optimal thrust range for the second phase.

θ **Begin** ($\times\frac{\pi/2}{1000}$)	θ **End** ($\times\frac{\pi/2}{1000}$)	**TT$_2$ [days]**	ΔV **[m/s]**	ΔV_{sp} **[1000 s]**
-975	1850	36.01	1185	7.58
-1000	1500	45.21	1100	8.17
-1250	1850	32.43	1219	7.37
-1100	**1800**	**35.37**	**1180**	**7.61**
-1800	1800	26.02	1343	6.7

In contrast with the LEO-patch point transfer, the lunar phase has time restrictions due to disturbances. In addition, the first angle where thrust is applied should not be too low, because the propulsion system is not able to compensate for large pericenter altitude changes. In this study, a thrust range of $-99\,\text{deg} < \theta < 162.0\,\text{deg}$ has been assumed, yielding a ΔV of 1.181 m/s and a travel time of 35.37 days. Adding up both phases yields $\Delta V = 1.676$ m/s and a TT of 68.04 days. It is stressed that there is room for improvement and further study on this problem. The results are summarised in Tab. 5.7. The final LLO orbit is at a 9° inclination, it is assumed that the Δi of 9° is neglected, as was explained in Sec. 5.2.3.1.

Table 5.7: Transfer characteristics patch point to LLO orbit.

Characteristic	**Value**	**Units**
Required ΔV_2	1.675	km/s
Transfer time (TT_2)	68.04	days

5.2.5 Moon to Earth Transfer

The Moon to Earth transfer is important to determine the total round trip Transfer Time (TT) and total ΔV required (for a propellant budget). For now, it is assumed that the way back to Earth is identical to the transfer from Earth to Moon. Using this assumption, the total ΔV is determined to be 17.13 km/s. Adding a contingency (+5%) for additional manoeuvres, this results in 18 km/s. The total TT should include the docking and rendezvous time needed in LEO and LLO. As has been determined in Sec. 5.11, the docking and rendezvous in LEO will take about two hours. The docking and rendezvous in LLO will take about six hours in total, as explained in Sec. 6.11. Adding some contingency and including orbit phasing (Sec. 5.2.6), the total docking and rendezvous time is determined to be five days, resulting in a total TT of 354 days.

5.2.6 Orbit Phasing

There is a possibility that the CTTV will enter the final orbit in LEO with a 180° difference in angle with respect to the Space Dock. This will result in an additional manoeuvre to reposition the CTTV near the Space Dock. In the Moon orbit, this orbit phasing considered negligible because the LSAM can easily reposition itself close to the CTTV. Orbit phasing can be performed by changing the circular orbit to an elliptical orbit which has a smaller semi-major axis and therefore a lower orbital period.

In order to determine the additional TT and ΔV, Eqs. 5.10 - 5.12 are used [175]. The first equation determines the semi-major axis that is required to provide a certain orbit phase angle (θ_z) and the number of orbital revolutions required (n) in an orbit with a radial distance of r_z (6878 km). With the second equation, the total phasing time (T_z) is calculated. The total required ΔV for the manoeuvre is calculated using the third equation. The absolute value for the ΔV is calculated for the first manoeuvre (to elliptical orbit) and the second manoeuvre (to circular orbit), therefore it is multiplied by two.

$$a = r_z \cdot \left(1 - \frac{1}{n} \cdot \frac{d\theta_z}{2 \cdot \pi}\right)^{\frac{2}{3}} \tag{5.10}$$

$$T_z = 2 \cdot \pi \cdot \sqrt{\frac{a^3}{\mu}} \cdot n \tag{5.11}$$

$$\Delta V = |\sqrt{\frac{2 \cdot \mu}{r_z} - \frac{\mu}{a}} - \sqrt{\frac{\mu}{r_z}}| \cdot 2 \tag{5.12}$$

With these equations, a 'worst case' scenario of 180° in LEO (500 km) is plotted versus the number of orbital revolutions in Fig. 5.8. The transfer time changes linearly whereas the required ΔV decreases exponentially. This will be a small percentage of the total transfer time, so there is some flexibility. For now, it is assumed that the limit on rendezvous and docking time will be five days. This indicates that the orbit phasing can take five days, minus eight hours of docking time. In this case, a ΔV of 35.5 m/s is required. The propulsion system of the CTTV does not have enough thrust to provide this ΔV as an impulse, so an additional system will be required. Another option is to increase the impulse time. However, this is outside the scope of this feasibility study.

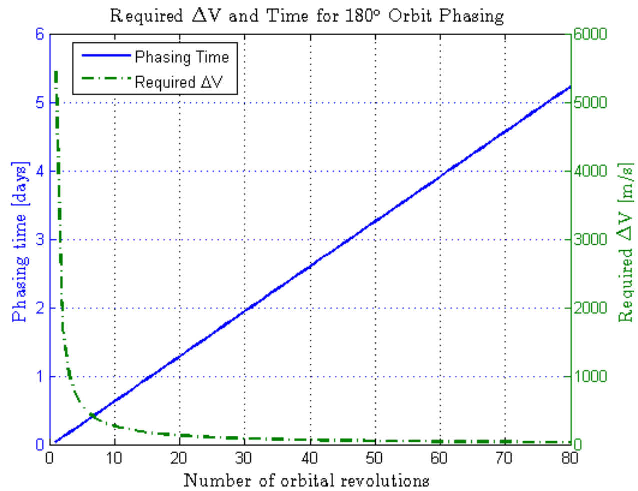


Figure 5.8: Orbit phasing manoeuvre, ΔV and transfer time.

5.2.7 Summary

Table 5.8: Summary transfer trajectory CTTV.

Characteristic	**Value**	**Units**
Required ΔV	18	km/s
Transfer time (TT)	354	days
Number of CTTVs	22	-

5.2.8 Verification and Validation of the Model

The Matlab model has been verified by checking the equations with arbitrary values where the solutions are known. This is done for Eq. 5.2 by putting the patch point at exactly the x-axis. The result of ϕ should be 0°. Eqs. 5.4 - 5.8 are verified by setting the Δi to 0°. The out-of plane thrust angle should be 0°, whereas the inclination stays at 23.43°. The required ΔV should be lower. The model was validated using the same inputs from the low-thrust LEO to GEO model [174]. The results were found to be exactly the same and the curves showed the same trends. The TT from 300 km to GEO (Δi of 28.5°) is 68.875 days and the total required ΔV is 5.951 km/s. This was using an acceleration of $1{\cdot}10^{-3}$ m/s^2. The starting out-of-plane thrust angle starts at 21.33° and ends at 66.11°.

5.2.9 Future Improvements of the Model

Due to the assumptions made in Sec. 5.2.1, there are some recommendations to improve the model. The transfer from Moon to Earth has already been discussed in Sec. 5.2.5. The other improvements will be discussed briefly.

Elliptical Moon Orbit: In reality the Moon orbit around Earth is elliptical instead of circular and has a perigee and apogee of 363300 and 405500 km respectively (eccentricity of 0.0549) [109]. This has a significant influence on the location of the patch point. For example, if the Moon is located at the perigee once the SOI is entered, the patch point needs to be located at 318983 km and during apogee at 361183 km. The optimal Moon orbit discussed in Sec. 5.2.2 also changed due to a different orbital velocity, resulting in an orbit with a lower or higher apogee (of optimal Moon orbit) for the Moon in its perigee and and apogee respectively.

Required Δi: This value is assumed to be 28.58°, which is the worst case. In general, this value varies between $i_E \pm i_M$ or 18.5-28.5°, depending on the location of Earth in its orbit around the Sun. As the required Δi is only a small part of the total required ΔV, it is a valid assumption. However, the total TT is depending on the Δi, and will, for this model, vary between 348 and 354 days for 18.5 and 28.5° respectively.

Perturbations on the CTTV: For the transfer between Earth and Moon, a significant amount of perturbations act on the CTTV. Perturbations include the gravity contribution of various planets [176]. As an improvement of the model, these accelerations should be added, especially the perturbations near the Moon. The Moon has large 'mascons' which make the orbit very unstable.

Include Timing: According to the previous mentioned improvements, timing is part of the model that can be improved. This will include launch mission windows and timing to accurately 'catch' the Moon.

Mirroring: It could be argued that it would not be possible to mirror the transfer and use the same values. The main difficulty would be at the patch point. This might require a different simulation for the trip from Moon to Earth.

5.3 Propulsion

The propulsion system is one of the driving systems behind this transport. As shown in the trade-off in Sec. 4.6, the VASIMR engine was chosen for this purpose. As stated in the conceptual design, the CTTV will use magnetoplasma rockets as a means for propulsion. The VASIMR project is the most developed engine in this field of propulsion, and will form a solid basis for most of the following calculations and design choices.

5.3.1 Thrust Levels

The inherent design of the VASIMR engine allows for, as the name states, a variable specific impulse. This implies the possibility to vary thrust levels if necessary. To explain the principle behind the variable specific impulse, the layout of the engine is shown in Fig. 5.9.

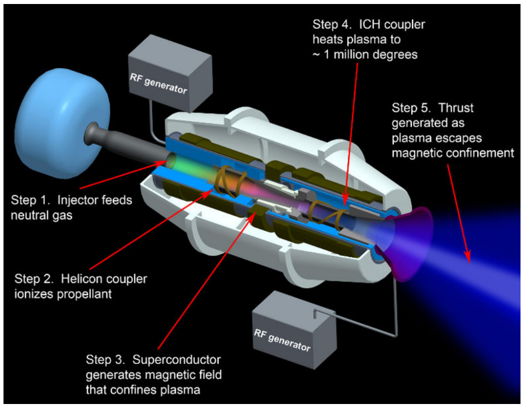


Figure 5.9: Layout of the VASIMR engine [106].

As can be seen in the schematic, step two and four involve heating of the propellant. The level of heating can be varied through energy input, meaning the temperature of the propellant, thus its exhaust velocity and/or mass flow can be changed. In more complex modelling of the orbital trajectory of the CTTV, this possibility for throttling of the engine can be incorporated, increasing the efficiency of the transport even further. However, such advanced modelling is considered to be beyond the scope of this feasibility study.

The current thrust provided by recent engine tests in August 2012 demonstrated the VASIMR engine could provide up to 10 N of thrust [177]. However, these results were obtained through testing of the experimental VX-200 engine. The engine chosen for this concept will be the engine intended for flight, the VF-200, which can, as of now, provide 5.7 N of thrust. However, it is expected that in the future, as research and testing furthers the design of the VASIMR project, efficiency and thrust levels, will increase, as shown by the experimental VX-200 engine [177]. Due to this, it is assumed that 11.4 N of thrust can be delivered by a 200 kW engine, or 5.7 N by a 100 kW model.

5.3.2 Propellant

The propellant required for the VASIMR engine must be a heavy gas, preferably a noble gas, for its low reactivity. However, most of the noble gases are rare on Earth. Indeed, as the atomic number of the atom increases, the natural availability of the noble gas decreases. However, Argon is one of the most abundantly available noble gases on Earth. At 0.93% of the Earth atmosphere, Argon can be readily extracted from air by fractional distillation, a widely-applied industrial technique involving cryogenics. This makes Argon, at 0.4 € per 100 g, a cheap, available propellant, that has already been tested and proven in the VASIMR testbed [106].

5.3.3 Propellant Budget

In a first instance, the mass flow rate of the propellant can be calculated using the following formula.

$$\dot{m} = \frac{T}{I_{sp} \cdot g_0} \tag{5.13}$$

In Eq. 5.13, T is the thrust in N, I_{sp} is the specific impulse in seconds, and g_0 is the Earth gravitational acceleration. Plugging in the standard values of the VASIMR engine, 5.75 N, 5000 s and 9.81 m/s^2 for thrust, specific impulse and gravitational acceleration respectively, a mass flow of 0.0012 kg/s is obtained.

With the mass flow now known, an initial estimate of the propellant budget can be calculated by multiplying the transfer time by the mass flow. This propellant budget was iterated once all mass budgets were known. The final mass propellant budget was estimated to be around 24.76 tons. With 5% contingency, this adds up to 26 tons.

5.3.4 Mass Budget

Using Fig. 5.14, a mass budget can be derived. It is given that the on-board total power required by 10 VASIMR engines will require approximately 1 MW of power. Using Fig. 5.14, a mass to power ratio (α) of 1.7 kg/kW can be derived. Including contingencies, this amounts to a total mass of the VASIMR engine block of 1800 kg.

5.3.5 Cost

Estimating the cost of a single VASIMR engine proved to be difficult. No public cost estimations were made available. Therefore, the cost estimation required a different approach. From a document released in 2011, it was found that while VASIMR was still a NASA-bound project (for approximately 25 years), it received a total funding of 5 M€. After going private, between 2005 and 2010 it received approximately 20 M€ in private investments. By becoming a public corporation, Ad Astra raised another 80 M€, to be able to launch the VASIMR engine as a commercial product by 2014 [178]. This amounts to a total funding of the project, up until commercial launch, of approximately 110 M€, including unmentioned funding contingencies or other possibilities. From that amount of total funding, the cost of a single VASIMR engine can be estimated to be a fraction of that, around 25 M€. This is a very conservative estimate. Especially seeing the fact that one CTTV will require ten VASIMR engines, and that the total expected amount of CTTVs required exceeds 30 units, totaling more than 300 VASIMR engines for large-scale operations. This also does not take into account a production learning curve or other cost-cutting possibilities. With an expected lifetime of 30 years for the CTTV, the annual cost of a single engine would be no more than 1 M€/year, excluding maintenance.

An annual maintenance cost of 3 M€ must also be considered, expecting that maybe a total of one engine must be replaced per year due to major malfunction, or all parts that need replacing approximately equal the cost of one engine. The cost for maintenance was estimated as a multiple of the total annual cost of the VASIMR engine, but actual numbers have not been proposed by Ad Astra.

A total propellant mass of 26 tons is required per year. At a price of 4 €/kg of Argon, the total propellant cost per year is equal to 104 k€.

5.3.6 Summary

One of the many advantages of the VASIMR is that its thrust does not depend on the reactivity of a chemical mixture. Instead, it solely relies on electric power to heat and accelerate the propellant. This makes the VASIMR capable of restarting the engine in case of an engine failure or general malfunctioning. The power supply is excluded in this section and will be discussed in Sec. 5.8.

Table 5.9: Summary table of the CTTV propulsion subsystem.

Element	**Mass [tons]**	**Cost [M€/engine]**
VASIMR VF-200	1.8	25
Argon propellant	26	0.1
Maintenance	-	0.3

5.4 Structures

The primary structure of the CTTV will be preliminary sized. The method followed will deviate slightly from that used in SMAD [60, Sec. 11.6 Structure and Mechanics]. First, the load requirements will be identified, then the configuration (load paths) will be analysed, followed by outlining the design options. Finally the structure is sized to meet the requirements.

5.4.1 Requirements and Assumptions

For the CTTV, it is assumed that the largest loads are experienced during launch and operations, however it should be noted that the spacecraft experiences also other loads [179]. Initial estimates on mass and dimensions show that current launching systems will not be sufficient to launch the CTTV as a whole. However, NASA is developing a new SLS capable of bringing 150 tons with a diameter of 8.3 m [180] into orbit. This is sufficient for the CTTV, but due to a lack of information, it is assumed that the load case will be similar to that of the Ariane V launcher [181]. A safety factor of 1.5 is introduced to account for differences between both launchers.

In Tab. 5.10 the loads in the static and dynamic loads in the longitudinal and lateral direction during launch and operations (transfer) is shown. The loads are most critical during the Solid Rocket Booster (SRB) end of flight phase. During this phase, the maximum dynamic and static load in the longitudinal and lateral direction is -9 and -1.5g respectively. The figure shows, that the lateral loads during phase 1 and 2 is larger than that in phase 3, however the most critical loading case occurs when both loads act simultaneously, hence during phase 3*.

Table 5.10: Load Factors experienced by the CTTV.

Acceleration [g]	Longitudinal		Lateral
Event	*Static*	*Dynamic*	*Static & Dynamic*
1. Lift-off	- 1.7	± 1.5	± 2
2. Maximum Dynamic Pressure	- 2.7	± 0.5	± 2
3. SRB end of flight	- 4.55	± 1.45	± 1
4. Main core thrust tail-off	-0.2	± 1.4	± 0.25
Max. case 3 (1.5 factor)	-9		± 1.5*

Ariane V's user manual specifies minimum values for the payload natural frequency in order to avoid dynamic coupling between frequency dynamics of the CTTV and launcher vehicle. For spacecraft weighing more than 6500 kg, the lateral and longitudinal natural frequency must be greater than 7.5 and 27 Hz respectively [182].

The structure will be clamped at the bottom (VASIMR engines and power plant) in the launchers payload bay. Thus only longitudinal and lateral loads on the upper half (See Fig. 5.13) are considered for the preliminary analysis. The following assumptions have been made.

- For the subsystems, masses are assumed to be point masses at the presumed centre of gravity of each system (See Fig. 5.13).
- The CTTV structure will consist of solid circular beams.
- Stress concentrations in the beams will be neglected.
- It is assumed that the beams will carry all the loads (no additional structural reinforcement).
- All beam ends are hinged.
- Structure is represented by a mass-spring system.
- Damping can be neglected.
- Mass of the beams are neglected during frequency analysis.

5.4.2 Configuration (load paths)

The subsystems have specific masses, dimensions and requirements, which drive the configuration of the CTTV. Fig. 5.13 shows various locations of the subsystems. The ADCS sensors, actuators and propellants, guidance, navigation, telemetry, tracking, command, data handling and thermal control systems will be located inside the upper module (labelled P4 in Fig. 5.13). The module is located far from the nuclear reactor, such that the module can be treated separately for thermal calculations. The rectangular core will provide support for the payload canisters, nuclear reactor, propellant and VASIMR engines. The modules dimensions are thus driven by the size of these components.

During launch, the longitudinal and lateral loads will exert a force on the upper beams, which will be supported by the remaining structure. Bending moments will be counteracted by the additional beams, thus only buckling and normal stresses are computed. In order to satisfy the frequency requirements, the spacecraft will be modelled as a mass-spring system supported in the x, y and z direction, each spring resembling one of the beams.

5.4.3 Material Selection

For the structure of both the CTTV and the LSAM, the available materials should be evaluated. It can then be defined which material suits best for the various structural components. The most common used materials are listed in Tab. 5.11. They yield Aluminium 7075-T6, Titanium Ti 6AL 4V and Structural Steel A36. The use of upcoming composites is another possibility, but for this mission it is left out of the table due to its uni-directional strength properties. Next to that, it is a complicated procedure to construct with composites.

Table 5.11: Material Properties of Aluminium 7075-T6, Titanium Ti 6AL 4V and Structural Steel A36

Material	σ **[MPa]**	**E [GPa]**	ρ [$\frac{kg}{m^3}$]	τ **[MPa]**
Aluminium 7075-T6 [183]	503	71.7	2810	331
Titanium Ti 6AL 4V [184]	1000	110	4420	550
Structural Steel A36 [185][186]	250	200	7850	-

In this table, σ is the yield strength, E is Young's modulus, ρ is the density and τ is the shear strength. As can be seen in the table, the E-modulus for aluminium is lower than that for titanium and structural steel. It implies that a bigger structure would be needed in order to make it equally strong as if titanium or steel were used. However, the density of aluminium is significantly lower. Therefore, aluminium is often used in aerospace application due to the light weight and it saves cost. Next to that, aluminium is very suitable for machining and has a high fracture toughness [60]. Therefore it is decided to use aluminium as main material for the structure of the CTTV and LSAM.

If volumetric issues arise, titanium is a good alternative. Titanium is stronger, meaning that less material is needed for the same strength as compared to aluminium. The main downside of using titanium is the high density and the cost due to expensive extraction techniques [60]. Steel might be, due to the high E-modulus, suitable for only the smallest parts such as bolts and pins. This is however too detailed for the scope of this project and will therefore not be considered.

5.4.4 Sizing

In order to preliminary size the structure, it must be tested/analysed when exposed to the critical loads. First, the buckling and normal stresses in each beam will be determined when exposed to a 9g longitudinal and 1.5g lateral load (Sec. 5.4.1). The structure will then be sized to resist the vibration loads exerted by the launching system and finally the final results are presented. Given that the moment applied on each beam is zero, stresses occurring due to bending will not be considered.

5.4.4.1 Buckling and Normal stress

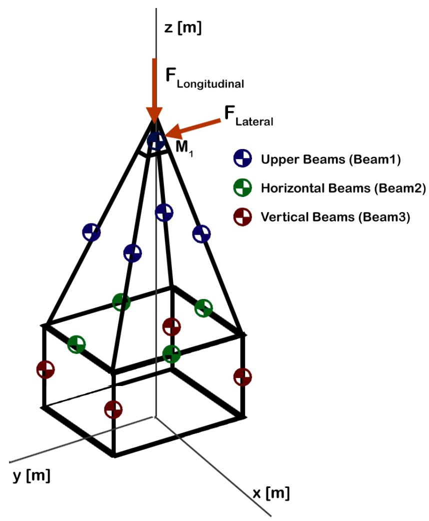


Figure 5.10: Loads on CTTV.

To determine the buckling and yield stresses the maximum force applied onto each beam is determined using statics. To determine the axial stress, the normal force (P) is divided by the surface area (A) where it is acting on [187]. In this equation, the axial force goes through the centroid and the stress is uniform across the entire cross section. The minimum area can be determined by using the yield stress of the material and normal force applied on each beam.

Eq. 5.14 is used to determine the beams required moment of inertia (I) when the critical force (F_{crit}) is applied, given the Young's modulus (E), beam length (L) and column effective length factor (n) [187]. In this structure the ends are hinged, thus n=1.

$$F_{crit} = \frac{n^2 \pi^2 E I}{L^2} \tag{5.14}$$

The results are shown in Tab. 5.12. Noticeable is that buckling is the driving factor for the beam thickness. As explained, E of Aluminium 7075-T6 is used.

Table 5.12: Results: buckling and normal stresses.

Beam	**Length[m]**	**Critical Force[kN]**	**Thickness [mm]** Yield	Buckling
Upper (Beam 1)	13.77	15.5	10.8	113.4
Horizontal (Beam 2)	6.4	7.98 / 4.38*	7.3	7.3
Vertical (Beam 3)	3.5	32.95	14.1	65.2

*: Two possible critical conditions exists on beam 2. One for axial stress and the other for buckling.

5.4.4.2 Vibrations

In order to avoid dynamic coupling between frequency dynamics of the CTTV and launcher vehicle, the structure must be designed to have a natural frequency above 27 and 7.5 Hz in the longitudinal and lateral direction respectively. The spacecraft can be modelled by a mass-spring system with eight springs in the longitudinal direction, four supporting the upper payload and four supporting the upper beams (see Fig. 5.11). In the lateral direction the system can be modelled by two springs (Fig. 5.12).

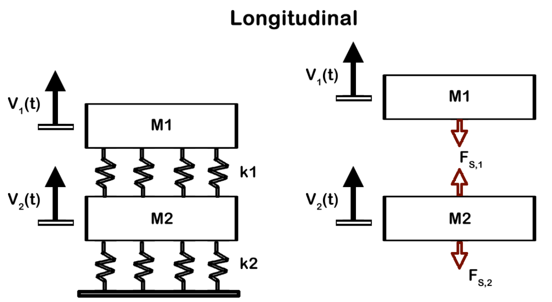


Figure 5.11: Longitudinal Mass-spring system (CTTV).

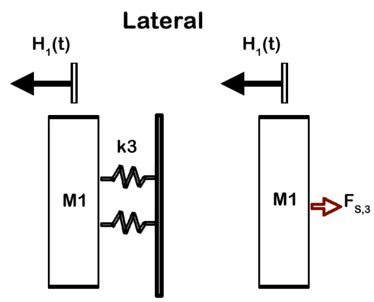


Figure 5.12: Lateral Mass-spring system (CTTV).

It is assumed that the damping coefficients are equal to zero, meaning damping is neglected. To compute the natural frequency of the system the forces $F_{s,1}$, $F_{s,2}$ and $F_{s,3}$ need to be determined. The force exerted by a spring can be calculated by multiplying the spring constant k with the displacement δ of the spring [188][187]. The spring constant of a massless beam can be determined using the materials Young's modulus (E). The Young's modulus of a beam can be found using Eq. 5.15. Here L is the length of the beam, A is the area and δ is the displacement. Force F, can be rewritten as $F = k\delta$. The equation can be rearranged and solved for k.

$$E = \frac{\sigma}{\varepsilon} = \frac{FL}{A\delta} = \frac{kL}{A} \tag{5.15}$$

The system can now be modelled by summing the forces (Eq. 5.16) [187].

$$\Sigma F : M\ddot{x}(t) = p(t) - c\dot{x}(t) - kx(t) \tag{5.16}$$

Here, M is the supported mass; $p(t)$ is the dynamic load; C_{Damp} is the damping coefficient; $x(t)$ is the displacement with time; $\dot{x}(t)$ is the velocity with time, and $\ddot{x}(t)$ is the acceleration with time.

Applying Eq. 5.16 to the longitudinal and lateral model (Fig. 5.11 and 5.12), replacing x(t) with horizontal and vertical displacement, H(t) and V(t) respectively, and rearranging for $p(t)$ yields Eq. 5.17 and 5.18.

$$\begin{aligned} p(t)_{Long} &= M_1\ddot{V}_1(t) - k_1(V_1(t) - V_2(t)) \\ &= M_2\ddot{V}_2(t) + k_1(V_1(t) - V_2(t)) - k_2(V_2(t)) \end{aligned} \tag{5.17}$$

$$p(t)_{Lat} = M_1\ddot{H}_1(t) - k_3H_1(t) \tag{5.18}$$

Assuming a harmonic solution of the form $x(t) = X_1 sin(\omega t - \varepsilon)$ and setting $p(t)$ equal to zero, Eqs. 5.17 and 5.18 become Eqs. 5.19 and 5.20 in matrix formation ($Ax = B$).

$$\begin{vmatrix} (-M_1\omega^2 - k_1) & k_1 \\ k_1 & -(M_2\omega^2 + k_1 + k_2) \end{vmatrix} \begin{vmatrix} V_1 \\ V_2 \end{vmatrix} = 0 \tag{5.19}$$

$$[\, -M_1\omega^2 \quad - k_3 \,][H_1] = 0 \tag{5.20}$$

Eqs. 5.19 and 5.20 can now be solved for ω. Finally the natural frequency is found using Eq. 5.21. Note: if the eigenvalues are complex ($a + ib$), the b is used to obtain the natural frequency.

$$f_n = \frac{\omega}{2\pi} \tag{5.21}$$

A summary of the input and output of this section is given in Tab. 5.13. Since this is a preliminary analysis, the values in Tab. 5.13 are just an indication whether the structure will be sufficient to withstand the vibrational loads. Since the natural frequency in the longitudinal and lateral direction is much larger than the required value, it can be concluded that the structure, with the beams computed in Sec. 5.4.4.1 passes the requirements.

Table 5.13: Results: natural frequency.

Longitudinal

	Length [m]		*Area* [m^2]		*Spring Constant* [N/m]		*Mass* [kg]		*Freq* [kHz]	
Parameters	L1	L2	A_1	A_2	k_1	k_2	M_1	M_2	Req.	Model
...Value	13	3.5	0.0057	0.0022	$4\times4.847\cdot10^7$	$4\times6.882\cdot10^7$	1015	1561	>0.027	140

Lateral

	Length [m]	*Area* [m^2]	*Spring Constant* [N/m]	*Mass* [kg]	*Freq [Hz]*	
Parameters	L_3	A_3	k_3	M_3	Req.	Model
...Value	4.95	0.0057	$2\times2.546\cdot10^8$	1015	>7.5	79

5.4.5 Summary

Table 5.14: Summary of the CTTV's structural preliminary analysis.

Beam	**No. beams** [-]	**Force/beam** [kN]	**Thickness/beam** [mm]	**Length/beam** [m]	**Mass/beam** [kg]
Upper (Beam 1)	4	15.5	13.77	113.4	218
Horizontal (Beam 2)	4	4.38	7.3	52.0	1
Vertical (Beam 3)	4	32.95	65.2	3.5	33

Now that the structure's mass distribution is known, the Mass Moment of Inertia (MMOI) can be computed. The MMOI about the x axis can be found using Eq. 5.22 [187], where I_x is the MMOI about the x axis, m_i is the mass of the i^{th} unit, ȳ is the y-location of the centroid (centre of gravity) and y_i is the y-location of the i^{th} unit. The centroid, also known as the centre of gravity can be computed using Eq. 5.23.

$$I_{x,y} = \sum_{i=1}^{n} m_i(\bar{y,}x - y_i, x_i)^2 \tag{5.22}$$

$$C_{x,y} = \frac{\sum_{i=1}^{n} m_i x_i, y_i}{\sum_{i=1}^{n} m_i} \tag{5.23}$$

Fig. 5.13 gives an overview of the masses and their locations in the structure. Tab. 5.15 contains the input and output of Eqs. 5.22 and 5.23. The MMOI calculated in this section is used to size the ADCS.

Figure 5.13: Preliminary Mass distribution CTTV.

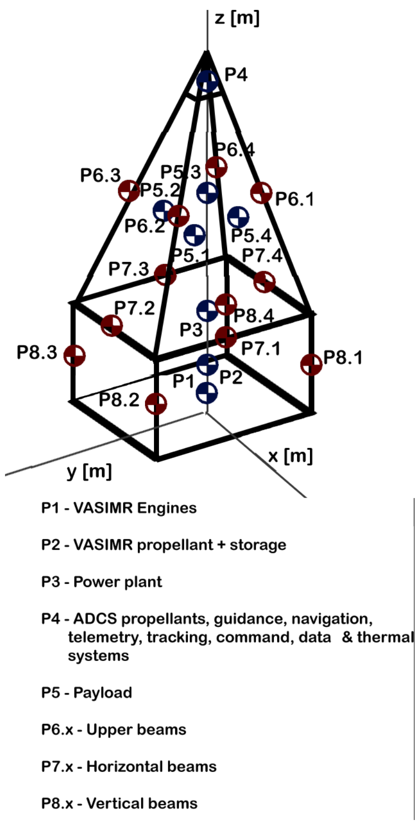


Table 5.15: CTTV mass distribution and MMOI computation.

	Mass [kg]	Centroid [m]		
		x	y	z
P1 Engines	2160	0	0	1
P2 Propellant	27436	0	0	2
P3 Power plant	4225	0	0	2.75
P4 Upper module	1015	0	0	14.5
P5.1 Payload 1	10770	2.34	2.34	8.5
P5.2 Payload 2	10770	-2.34	2.34	8.5
P5.3 Payload 3	10770	-2.34	-2.34	8.5
P5.4 Payload 4	10770	2.34	-2.34	8.5
P6.1 Upper beam 1	390	-1.6	1.6	7.775
P6.2 Upper beam 2	390	1.6	1.6	7.775
P6.3 Upper beam 3	390	1.6	-1.6	7.775
P6.4 Upper beam 4	390	-1.6	-1.6	7.775
P7.1 H beam 1	1	3.2	0	3.5
P7.2 H beam 2	1	0	3.2	3.5
P7.3 H beam 3	1	-3.2	0	3.5
P7.4 H beam 4	1	0	-3.2	3.5
P8.1 V beam 1	33	3.2	-3.2	1.75
P8.2 V beam 2	33	3.2	3.2	1.75
P8.3 V beam 3	33	-3.2	3.2	1.75
P8.4 V beam 4	33	-3.2	-3.2	1.75

	x	y	z
Centroid	0	0	5.292
MMOI [kg m^2]	$1.0397 \cdot 10^6$	$1.0397 \cdot 10^6$	$1.2373 \cdot 10^6$

5.4.6 Verification and Validation

The calculations of the preliminary analysis of the structure were performed by Matlab. The model was verified by hand and finally the output was tested by using a simplified model (a simple cubic structure). Please keep in mind that this is just a preliminary model, the assumptions made in Sec. 5.4.1 simplified the solution, but resulted in a reduction in accuracy (small).

5.5 Thermal Control

Temperature control of the CTTV is crucial for maintaining all the subsystems at an operational temperature. The spacecraft will be subject to the extremes of the space environment. For the thermal subsystem design of this mission, four cases were addressed as listed below. It is assumed the environmental conditions during the transfer do not exceed these cases. First, the equilibrium temperatures are calculated based on the thermal environment. Secondly, the power required to maintain the temperature in the survival range (263 to 298 K) is computed. Finally, the components of the subsystem are identified with a mass and power budget.

- Case I - Hot Earth: Maximum solar radiation in Space Dock orbit.
- Case II - Cold Earth: Behind Earth in Space Dock orbit (i.e. no solar radiation or albedo).
- Case III - Hot Moon: Maximum solar radiation in LLO.
- Case IV - Cold Moon: In Moon shadow in LLO (i.e. no solar radiation or albedo).

5.5.1 Assumptions

The following assumptions have been made, in addition to the assumptions already made in Sec. 4.2.3 (Conceptual Payload Design).

- The CTTV will be treated as 2 independent nodes, 1 Power supply & VASIMR node and one for other subsystems.

- The power supply node will provide its own thermal system which is included in the thermal system.
- The surface area of the subsystem node equals 11.6 m^2.

5.5.2 Equilibrium Temperature

For the cases stated above, the equilibrium temperature of the subsystems node has to be computed. From these temperatures, the power requirements will follow for the thermal subsystem. With the assumptions stated, the equilibrium temperature for the four cases is calculated using Eq. 5.24.

$$T_{eq}^4 = \frac{A_{planetary} \cdot J_p}{A_{surface} \cdot \sigma} + \frac{Q}{A_{surface} \cdot \sigma \cdot \epsilon} + \frac{A_{solar} \cdot J_s + A_{albedo} \cdot J_a}{A_{surface} \cdot \sigma} \cdot \frac{\alpha}{\epsilon} \tag{5.24}$$

In this formula, several intensities (J-factors) need to be defined. The solar radiation intensity J_s is defined with Eq. 5.25 [58].

$$J_s = \frac{P}{4 \cdot \pi \cdot d^2} \tag{5.25}$$

In this equation, P represents the total power output from the Sun and d defines the distance between the spacecraft and the Sun, assumed to be 1 AU. Earth and the Moon reflect the solar radiation they receive partially. This albedo radiation intensity is defined as:

$$J_a = J_s \cdot a \cdot F \tag{5.26}$$

The planetary albedo a is as stated in the assumptions. The parameter F represents the visibility factor. The factor mainly depends on the angle between the local vertical and the Sun's rays. From Fortescue et al., it follows that the maximum F will be approximately 1.5, assuming β equal to zero for maximum albedo [58]. The albedo only affects the non-shadow cases (I&III). Finally, the intensity of the planetary radiation is defined:

$$J_p = I_r \cdot \left(\frac{R_{rad}}{R_{orbit}}\right)^2 \tag{5.27}$$

I_r represents the planetary radiation intensity calculated with $I_r = \sigma \cdot T^4$ (with T being the black body temperature of the Moon) [189][58]. R_{rad} is the effective radiating radius and R_{orbit} equals the height of the orbit measured from the centre of the planet.

Another unknown is Q, the internally dissipated power. It comes from heat generated by other subsystems. The heat from the power supply subsystem is assumed to radiate directly to space and the system is assumed to be isolated from the other subsystems. The electrical subsystems are assumed to have an efficiency of 95%. For the CTTV the dissipated powers are summarised in Tab. 5.16.

Table 5.16: Internally dissipated power CTTV.

Subsystem	**Dissipated power [W]**
Navigation (GNC)	5
ADCS	23.45
TT&C	0.5
C&DH	1.25
Total	30.2

Table 5.17: Equilibrium temperature for the four cases.

Case	**Equilibrium temperature [K]**	**Required power [kW]**
I: Hot Earth	301.6	0
II: Cold Earth	190.5	1.42 (heating)
III: Hot Moon	286.7	0
IV: Cold Moon	203.5	1.16 (heating)

Using Tab. 5.16, an $\frac{\alpha}{\epsilon}$ ratio of 0.57, with an emittance of 0.7 and absorptivity of 0.4, a surface area of 11.6 m^2 and assuming the planetary, albedo and solar area to be a third of that (corresponding to the outer layer of MLI [60]), the equilibrium temperatures can be computed. The results of the equilibrium temperatures are summarised in Tab. 5.17. It must be noted that for the cold cases, the albedo and solar radiation effect are neglected as stated in the assumptions, and the J-factors are different for Earth and Moon.

The equilibrium temperatures found are outside the desired range of -10 to 25°C. To find the power required for heating or cooling the following equation is applied:

$$P_{req} = A_{planetary} \cdot J_p + Q + (A_{solar} \cdot J_s + A_{albedo} \cdot J_a) \cdot \alpha - A_{surface} \cdot \sigma \cdot \epsilon \cdot T^4 \tag{5.28}$$

The temperature in the equation will set the boundaries of the desired temperature range, i.e. for the cold case 263 K and for the hot case 298 K. This results in four power requirements summarised in Tab. 5.17. From these results, the highest power requirement for cooling and heating can be found.

5.5.3 Components Subsystem

There are multiple options to control the temperature of the system. The main options yield passive and active temperature control. Passive control does not require a power input, which would be desirable for the CTTV. The downside is that it only covers a small temperature range, while an active control can cover large ranges using a power input. For the extreme cases, active control might be a necessity for the CTTV. To control the temperature, the ϵ and α can be varied, or a power required for the subsystem can be found as shown above. Firstly, passive control will be applied using a 15 layer MLI on the inside of the spacecraft. The insulation causes internally dissipated power to heat up the subsystem node from the inside. The MLI weighs 0.73 kg/m^2 and with an area of 11.6 m^2 this results in a weight of 8.47 kg. MLI reduces the power requirement for heating to 1.37 kW, which will be provided by a heater using active thermal control. Patch heaters will be used with a capacity of 1 kW per sheet. Two patches of 25 by 25 cm will be installed for the subsystem node. Since the patches are thin and flexible, their mass is assumed to be negligible. The cost for the MLI layers equals 11347 €, assuming 15 layers with a cost of 65.21 € per m^2 [62]. The patch heaters will cost 68.5 € per patch, resulting in a cost of 137 €. The total cost is estimated at 11500 €.

5.6 Attitude Determination and Control

For the Attitude Determination and Control System (ADCS) of the CTTV, a stepwise approach has been used, in which initially the different attitude control modes have been identified. After that, the corresponding requirements of the ADCS have been defined. From these requirements the method for the attitude sensing and control could be chosen, since each method has specific specifications. The ADCS hardware is then sized according to the mission profile and the actual hardware is proposed.

5.6.1 Attitude Control Modes

For different missions, different control modes are required. For the CTTV these control modes are as follows, as defined in SMAD [60]:

- **Acquisition:** The initial determination of the spacecraft attitude and it stabilisation.
- **Normal, On-Station:** ADCS general operations during the mission.
- **Slew:** Reorientation of the spacecraft whenever required.
- **Contingency or Safe:** May be used in case of failure or to include a safety margin on the subsystem.

5.6.2 Selection of the ADCS Hardware

From the required high slew-rates for the docking procedures and the specific orbit in which the CTTV transfers towards the Moon, the sensors and actuators for the ADCS are chosen. Some apply more to the specific mission profile than others, which will be elaborated upon in this subsection.

First, the selected sensors will be discussed. For the ADCS two sensors are used, the star tracker and an Inertial Measurement Unit (IMU) to support the star trackers. Two of each for redundancy. In addition to this the ADCS could possibly make use of the sensors which were discussed in Sec. 5.7. The star trackers were chosen over sun sensors and horizon scanners due to a constant view on the stars. The other two scanners would not work during eclipse or far away from Earth. Next to that, a magnetometer was excluded because it does not have a high attitude accuracy. As a placeholder for the star tracker the CT-602 from Ball Aerospace & Technologies Corp. [190] is used. It has a 12-15 years lifetime and is proven from LEO to deep space missions. The accuracy is in the range of 0.05 - 0.2° as it degrades over time, this is just above the required 0.1° but expected to increase by 2040. The CT-602 has a weight of 5.5 kg and an average power consumption of 9 W. The cost of one star tracker is estimated at 390000 € [191].

For the IMUs the HG-1700 from Honeywell is set as a placeholder, having a mass of only 0.9 kg each and a power consumption of 8 W [192]. The total mass and power for the sensors will be 12.8 kg and as there is only one of the two operative the total power consumption is 17 W. The cost for one IMU is estimated at 7 k€ [193].

As for the actuators, thrusters are chosen. They have no constraints in pointing options and have an accuracy in the range of 0.1 - 5°, as three-axis control is required [60]. The CTTV has a larger mass (81.5 tons) than common spacecraft discussed in SMAD [60]. Therefore, the thrusters are the only options to deliver the performance that is needed. The chosen thrusters will be discussed in the next section. In addition to the thrusters, Control Momentum Gyroscopes (CMGs) are needed to operate within a close range to the LSAM and provide small corrections.

5.6.3 Sizing and Choice of ADCS Hardware

To be able to choose the ADCS hardware, some of the actuators should be sized to see how large their required output is. Different sizing equations are used for the different estimations. The torque that the reaction wheels are required to deliver is shown in Eq. 5.29.

$$T = 4\theta \frac{I}{t^2} \tag{5.29}$$

Here θ is the required angle that should be covered in time t, which in the case of the reaction wheels is 1 deg/min, since they are required for minor disturbance and manoeuvre adjustments. The highest Moment of Inertia I for the CTTV is I_x, which yields the maximum required torque from the reaction wheels. They should be able to deliver a torque of 20.2 $N \cdot m$. For a torque of this magnitude the use of CMG is preferred over reaction wheels, so the CMG M50 of Honeywell is proposed [194]. Unfortunately the choice in CMGs is not very elaborate, so the M50 is slightly oversized with its maximum torque of 75 N·m. It would be advisable to revise the choice when more alternatives are available.

The location of the thrusters is shown in Fig. 5.1. The eight thrusters located at the bottom and the top of the CTTV account for the rotations around the x- and z- axis. For the rotations around the y-axis the eight thrusters just below the payload canisters will be used. The required force level of the thrusters is determined using Eq. 5.30.

$$F = \frac{I\ddot{\theta}}{L} \tag{5.30}$$

Here I is the Moment of Inertia of the respective axis around which should be rotated, which is approximately $1.04 \cdot 10^6$ kg·m^2 for the x- and y-axis and $1.24 \cdot 10^5$ kg·m^2 for the z-axis. The angular acceleration $\ddot{\theta}$ comes from the required slew rate. This is set at 10 deg/min for slewing using the thrusters. Slewing is done by accelerating to a certain speed, coasting and afterwards decelerating again. With 10 deg/min the average speed is 0.167 deg/s^2. It is assumed that the CTTV is accelerated during 10 s, which brings the acceleration to 0.0167 deg/s. The moment arm L is 2.5 m when rotating about the y-axis times four, since the rotation can be done in double coupled thrusting. When rotating about the x- and z-axis, the arm of the thrusters are 5.8 and 9.8 m for the lower and upper thrusters, respectively. Using this the required thrust is set at 9.4 N for the thrusters for rotation around the z-axis and at 20.1 N for rotation around the x- and y-axis. The proposed thrusters are the 20 N Hydrazine thrusters of EADS Astrium [195], see the specification is Tab. 5.18. The cost are estimated at 62 k€ [196].

Table 5.18: Specifications EADS Astrium 20 N Hydrazine thruster [195].

Characteristic	**Value**	**Unit**		**Characteristic**	**Value**	**Unit**
Thrust range	7.9 - 24.6	N		Mass	0.395	kg
I_{sp} vacuum	224 - 230	s		Length	195	mm
Cycle life	93130	cycles		Cost	62	k€

To determine the thruster pulse life, it should be estimated what number of pulses are required per year. Every time the CTTV passes the patch point, the yaw has to be adjusted. To divide these pulses between the thrusters, the CTTV is rotated by 90 degrees after every year. It is assumed that for every docking manoeuvre 40 pulses per axis are necessary. For orbit corrections one adjustment per axis per day is assumed. For every time a thrust is induced, a decelerating thrust is needed as well, doubling the amount of pulses. These numbers are only estimations and may therefore be subject to changes. A contingency of two is included to ensure a safe estimation. With all the numbers combined, the thrusters at the top and bottom of the CTTV are expected to have 2820 pulses per year, while the eight thrusters in the middle deliver 1630 pulses per year. With a an expected lifetime of 93130 pulses, this implies the thrusters can be used for several decades [195].

The final propellant mass per year can now be determined using Eq. 5.31. The total pulse duration t is the pulse length multiplied with the number of pulses. Since not always a pulse duration of 10 s is needed and minor adjustments can require shorter impulses, an average pulse length of 6 s is assumed. The proposed Hydrazine thruster has a specific impulse I_{sp} of 224 s. From the estimations and calculations, a total required propellant mass of 1.57 tons will be required per year for all the pulses of all thrusters.

$$M_p = \frac{F \cdot t}{I_{sp} g} \tag{5.31}$$

5.6.4 Budgets

In the breakdown of Tab. 5.19, the cost, mass and power of a single component are given. For the total budgets the total amount of components has been taken into account. The propellant that is used by the thrusters is included as a separate entry in the table and has a cost of 136 €/kg [197].

Table 5.19: CTTV ADCS budget breakdown.

Component	Amount	Cost [k€]	Mass [kg]	Power [W]
IMU	2	7	0.9	8
Star Tracker	2	390	5.5	9
Thruster	16	62	0.395	-
CMG	4	153	28	113
Total	-	2398	131.1	356
Propellant	-	214	1572	-

5.7 Guidance and Navigation

For the Guidance and Navigation System (GNS) guidance refers to adjusting the orbit, where navigation refers to determining the attitude and position of the spacecraft [60]. Normally, the control part is related to the ADCS and therefore these two subsystems are strongly related. First, the accuracy requirements for different phases of the mission will be defined, after which the orbital control and maintenance will be discussed. Then a suitable navigation method is chosen. The results are listed in Tab. 5.20. The Autonomous Navigation System (ANS) will be used for general navigation, whereas the Autonomous Transfer Vehicle (ATV)-derived Rendezvous and Docking System (R&DS) will be used for docking procedures.

Table 5.20: Overview GNS of the CTTV.

Component	Operating mode	Accuracy range [m]	Power [W]	Mass [kg]	Cost [M€]
ANS	Autonomous	100-400	<8	<2.5	≈ 0.235
ATV-derived R&DS	Autonomous	<0.01 m	<150	≈ 38	≈ 1
Total	-	-	≈ 150	≈ 40	≈ 1.235

5.7.1 Accuracy Requirements

From the top-level requirements of the GNS defined in Sec. 5.1, preliminary (position) accuracy requirements can be defined. This will be done in several phases of the mission. For the first phase, the LEO docking phase, it is recommended that the position accuracy is smaller than 7 m. This is due to the fact that the CTTV has to be in range of the Payload Exchange Robotic Arm (PERA) for the docking procedures near Earth. In the second phase, the transfer can be performed autonomously and does not need an accurate position. However, the position accuracy at the patch point of the Moon SOI needs to be in the range of ±500 m in order to determine further actions, as has been discussed in Sec. 5.2. Lastly, the docking with the LSAM needs a high positional accuracy because the PERAs will not be used for docking. The diameter of the NASA Docking System (NDS) is 1.72 m, see Sec. 4.9.2, and the preliminary position accuracy is determined to be less than 0.5 m.

5.7.2 Guidance

Orbit control or guidance is related to adjusting the orbit to meet some predetermined conditions. Orbit maintenance is only related to maintaining this predetermined orbit [60]. For this mission, orbit control and maintenance is needed to counteract the perturbations present in the transfer trajectory and especially the Moon orbit. Next to this, orbit control is needed to target an end orbit. As the time spent in LEO and LLO orbit is relatively short, the orbit maintenance is not relevant compared to the orbit control. Additionally, autonomous orbit control will be provided, reducing the life-cycle cost and risks.

5.7.3 Navigation

The navigation data is needed at multiple locations, for example at the Space Dock and lunar base. Real-time navigation data will be needed on board of the CTTV and by intervals communicated to the ground station. This data will be required using semi-autonomous[1] navigation. The navigation method will be an ANS due to the use of highly reliable Earth, Moon and Sun observations [60]. It can be compared to Microcosm Autonomous Navigation System (MANS). The characteristics can be seen in Tab. 5.20. It uses only a single sensor and has already been tested [60]. This system can be combined with components of the ADCS. On an additional note, it requires a computer with a mass of around 5 kg and 30 W of power [198]. This computer will be included in the command and data handling subsystem. The cost of this system is about 235000 € [199]. This navigation method has an accuracy below the required accuracies mentioned in Tab. 5.1. Therefore, an additional system (R&DS) derived from ESAs ATV is used during the docking procedures [200]. This system uses two videometers (VDMs)[2] [201]. One VDM will weigh

[1] Semi-autonomous can be compared to a navigation constellation like the Global Navigation Satellite System (GNSS)
[2] A VDM emits pulsed laser beams, which are reflected and the reflected image is analysed.

about 5 kg and will consume about 36 W [202]. Next to this, there are two telegoniometers[3] (TGMs) to determine the attitude and position with respect to the LSAM or Space Dock [201]. The TGMs are used to add redundancy and a safety margin. One TGM will weigh about 14 kg and will consume about 35 W [203]. Two of each are required for redundancy, but are active during rendezvous. The cost of this system has not been found, but is estimated at 1 M€. This system can be used if two vehicles are within a range of 250 m, which can be determined by the ANS. The light beam sent by the VDMs is reflected by reflectors which only have a deviation of 3 mm on a distance of 300 m [200]. On June 18^{th} 2013, the ATV has been docked to the ISS with an 11 mm precision, which corresponds to the required position accuracy [204].

5.8 Power Supply

The CTTV uses ten VASIMR engines for propulsion, each consuming 100 kW power as stated in Sec. 5.3. This accumulates to a total of 1 MW of required power, meaning that power budgets of remaining subsystems are almost negligible. Only the payload canisters have a non-negligible influence. In total 1146 kW will be required per CTTV. To include a contingency in the requirement, it is assumed that to power the whole CTTV a total of 1.2 MW is needed. With a required power of 1.2 MW and a lifetime of multiple years, the desired power source will be a nuclear reactor [205].

Very recent proposed concepts of a nuclear power source for the VASIMR engines include a NEP (Nuclear Electric Power) system, which consists of a closed cycle MHD (Magneto-Hydro-Dynamic) nuclear plant, that is characterised by a multi-megawatt power level and a low specific mass α, expressed in kg/kW [107]. The NEP consists of a nuclear fission reactor that generates thermal energy, fuelled by Uranium-235 [206]. This thermal energy is transferred to the MHD generator, which converts the heat from the reactor partly to electric energy and partly to waste heat, using He/Xe Frozen Inert Plasma (FIP). The electric energy is mainly used to power the VASIMR engines, while the remaining electric energy is used later in the closed cycle again. The waste heat from the MHD generator is partly transferred to the high pressure inert gas entering the fission reactor again. The low-pressure cooled gas rejects more heat using a radiation cooler before entering the compressors. These are powered using part of the electric energy from the MHD generator output. Using intermediate radiators between the different stages of the compressor, the gas is compressed at an as low as possible temperature to maximise the compressor efficiency. The pressurised gas is then heated using the aforementioned waste heat from the MHD generator before entering the fission reactor again [207][107].

5.8.1 Operations

In order for the power plant to deliver the required 1.2 MW of power, the nuclear fission plant should deliver a certain power output in the form of heat. From the proposed closed cycle concept it is assumed that 61% of the electric output of the MHD generator is effectively used for the propulsion and the other subsystems, while the remaining electric power is used to optimise the closed cycles' efficiency [207][107]. This would imply that the MHD generator should have an output of 1.97 MW to generate 1.2 MW. The sources expect an achievable enthalpy extraction rate of 35%, which by other sources is even expected to reach 40% [208]. To have a conservative estimate, an enthalpy extraction rate of 35% is assumed yielding a required power output of 5.62 MW. Using this, the required amount of Uranium-235 can be computed.

During Uranium-235 fission multiple fission reactions can occur, which release on average 202.5 MeV per individual reaction [209][210]. Given that 1 MeV equals $1.602177 \cdot 10^{-13}$ Joule and using Avogadro's number (N), the energy per mole of U-235 can be computed as shown in Eq. 5.32:

$$6.0221413 \cdot 10^{23} \cdot (202.5 \cdot 1.602177 \cdot 10^{-13}) = 1.9538 \cdot 10^{13} \ \frac{J}{mol} \tag{5.32}$$

The equation yields that per mol of fused Uranium-235, 5427.3 MWh of energy in the form of heat is generated. From the previously determined required amount of 5.62 MW from the nuclear plant, it follows that the plant can meet this requirement for $5427.3/5.62 = 965.6$ hours per mol of Uranium-235. Its molar mass is 235.044 g/mol, so the CTTV will be supplied with power for 171 days for every kg of Uranium-235. Since refueling and disposing the fuel waste from the power plants on the CTTVs would impose large complexity and risk to the mission, it is chosen to supply the power plants with sufficient fuel to last throughout the full mission duration. This can be in the order of years or even decades without needing too large masses of Uranium on board, as was shown in the calculations in this section.

The CTTV relies entirely on the nuclear power plant for its power supply. Adding a redundant power system is not possible given the large size and mass of the power plants. To ensure communication and tracking of the CTTV in the very unlikely case that the power plant fails, a primary battery system is included. For the communication and tracking system, 10.1 W of power is required (see Sec. 5.9), which should last for a duration of 2 months. This timespan is derived from the expected time to have a (pre-planned) recovery mission executed, together with the worst-case distance at which the CTTV could be located. With 10.1 W for 2 months, 14544 W·hr is required from the

[3] A TGM determines the attitude and position with respect to the other spacecraft by means of radio waves.

battery. Given the current specific energy density of primary batteries of 590 W·hr/kg [211], the battery should have a mass of approximately 25 kg. In this way the valuable payload of the CTTV can be obtained even in the case of a power failure.

A challenge that should be addressed for the CTTV segment is the end-of-life disposal of the spacecraft. Since they will contain a certain amount of radioactive waste, returning them to Earth or putting them in a graveyard orbit around Earth will impose an unacceptable risk. Other options for disposal include the storage of the old CTTVs on or beneath the lunar surface or sending them into deep space or towards the Sun, but the final optimal choice will depend on the ethical and political situation in 2040.

5.8.2 System Budgets

The contribution of the power system to the budgets of the total CTTV system is determined in this subsection. Needless to say that its contribution to the power budget is not relevant, given that it delivers the full power budget. For the mass contribution of the power system the estimations in Fig. 5.14 can be used.

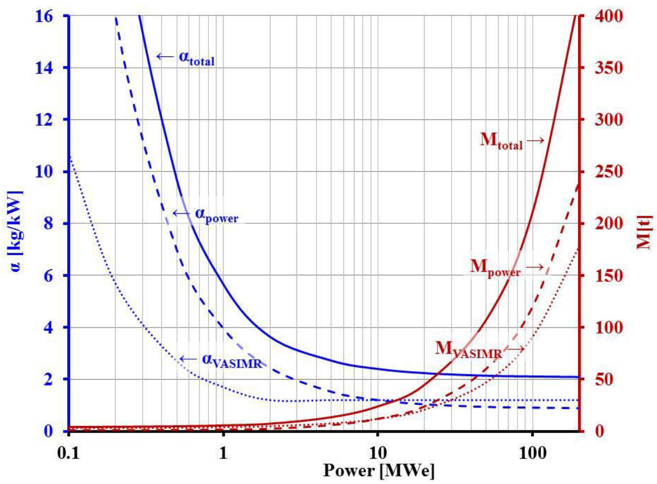


Figure 5.14: Specific and Total Mass for the Nuclear MHD - VASIMR System [107].

For a net output of 1.2 MW, a specific mass for the power system of 3.2 kg/kW can be assumed, and with a contingency factor included, a specific mass of 3.5 kg/kW is used. This results in a total mass of 4.2 tons for the power system.

At this stage of development, no cost estimations have been made on the proposed concept from the sources. To give a rough estimate on the acquisition costs of the power system, estimations of another nuclear electric propulsion mission are used. Given the great changes that may occur in the decades to come, a contingency of 50% is included, resulting in an assumed cost of 168 M€ for the nuclear power system [212].

5.8.3 Electric Diagram

In Fig. 5.15, a schematic of the electric configuration of the CTTV can be found.

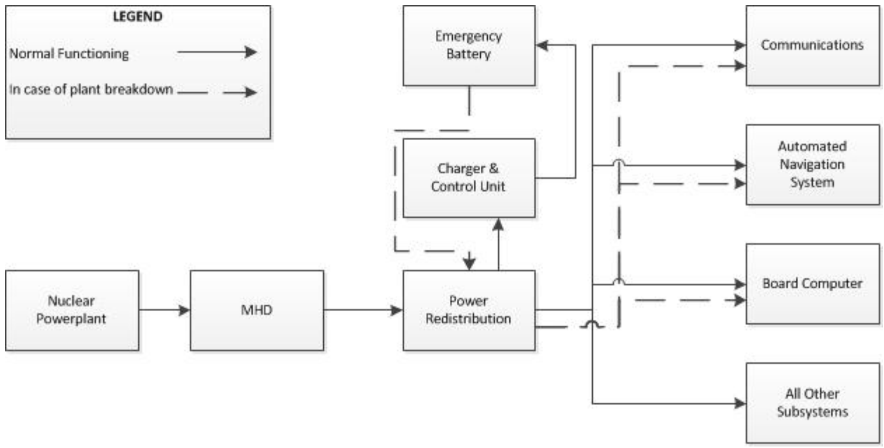


Figure 5.15: Electric block diagram of the CTT.

5.9 Telemetry, Tracking and Command

Another important subsystem is the communication subsystem. For a successful mission, it is important that the CTTV is able to both transmit data and receive commands. In this section, the detailed design of the communication subsystem of the CTTV is presented. First of all, the assumptions made will be addressed. Then a communication architecture will be described. Subsequently, ground stations will be selected. A link budget will be provided, the required power for the antennas on board of the satellite will be estimated and a mass budget estimation will be shown.

5.9.1 Assumptions

- Direct Earth-CTTV communication until patch point, then via relay satellites.
- Communication equipment for docking negligible in terms of mass and power.
- Relay Satellites are similar to NASA's TDRSS.
- A needed downlink of 400 kbps (See Sec. 4.3).
- A needed uplink of 40 kbps (See Sec. 4.3).
- Low Bit Error Rate (BER) of 10^{-6} (See SMAD [60]).
- Polarisation and propagation loss of -0.6 dB (See SMAD [60]).
- Transmitter line loss of -1 dB (See SMAD [60]).
- Pointing error is half the half-power beamwidth (SMAD [60]).
- Antenna efficiency of 0.55 (SMAD [60]).

5.9.2 Communication Architecture

As mentioned, the communication subsystem of the CTTV plays an important role in the mission. As addressed in Sec. 4.3, the CTTV has three main communication links: a link with the ground segment and links with both relay satellites in LL1 and LL2, respectively. Next to that, a direct link will be established during docking procedures between the CTTV and the LSAM and between CTTV and the Space Dock, respectively. A direct link between the CTTV and the ground segment will be present as long as the CTTV has not reached the patch point. Once the CTTV passed this point, communications with the Earth will pass through one of the relay satellites. Communications with the lunar base will occur via the relay satellites.

5.9.3 Ground Station and Relay Satellite Selection

As explained in Sec. 4.3, a continuous link between Earth and the CTTV is desired because of the complexity of the mission. For a continuous link, three ground stations should be used (equally spaced around the equator). As ESA is the customer, it was decided to use the ESA Tracking Station Network (ESTRACK) ground stations [213]. ESTRACK provides a continuous link with the CTTV. Next to that, the ground station network enables selection of the desired frequency bands as explained in Sec. 5.9.4. Three of the ESTRACK ground stations are Deep Space Antennas (DSA), which are placed in New Norcia (Australia), Cebreror (Spain) and Malarge (Argentina) and can therefore be used for deep space communications [214]. They are capable of communicating both relay satellites. As redundancy, it might be possible to use, for example, NASA's Deep Space Network (DSN) [215]. ESA's DSAs are 35 m in diameter, which will be used in Sec. 5.9.5 to determine the receiver gain.

Concerning the relay satellites, possible reference systems are the European Data Relay Satellite System (EDRSS) and NASA's TDRSS [216]. The latter is already operational and will be used as primary reference. The TDRSS satellites have an antenna diameter of 4.5 m.

5.9.4 Frequency, Data Rate and Memory Size

With the used ground stations and relay satellites known, suitable frequencies need to be selected for uplink and downlink signals. In Tab. 5.21, various uplink and downlink frequencies with corresponding frequency bands, as defined by the International Communications Union (ITU), can be found [217]. The ground stations from ESA support both S-band and X-band frequencies. Both frequencies have the advantage of being less affected by atmospheric attenuation. The newer and (in bandwidth) more abundant X-band will be used for the communication between the CTTV and the ground stations. The uplink and downlink frequencies are selected as 7.9 GHz and 7.25 GHz, respectively.

For communication between the relay satellites and the CTTV, the Ku-band is selected due to a large available bandwidth [216]. The disadvantage of using Ku-band is the high sensitivity to atmospheric attenuation, which is in this case not a problem because there will be no atmospheric barrier between the relay satellites and the CTTV. The selected frequencies are based on the satellites as used in the TDRSS, yielding 14 GHz for the forward link and 12.75 GHz for the return link.

During docking, the CTTV also has to communicate with the LSAM or the Space Dock. These links, however, have to be established rarely and only if the CTTV is close to one of these systems. Therefore the Ultra High Frequency (UHF) band will be used for this communications. As the antennas for short distances of this UHF band are small in terms of size and power, they are neglected for the further design of the communication subsystem [217].

Finally a data rate needs to be assigned to the various links. For all links, the uplink and downlink data rates are set at 40 and 400 kbps respectively, as explained in Sec. 4.3. Because of the continuous link and the multiple redundant ground stations, there is no data storage required. A small memory may be added to the CTTV to account for unexpected problems with the communication links.

Table 5.21: Frequency bands [217].

Frequency band	**Uplink frequency [GHz]**	**Downlink frequency [GHz]**
UHF	0.2-0.45	0.2 - 0.45
L	1.635 - 1.66	1.535 - 1.58
S	2.65 - 2.69	2.5 - 2.54
C	5.9 - 8.4	3.7 - 4.2
X	7.9 - 8.4	7.25 - 7.75
K_u	14 - 14.5	12.5 - 12.75
K_a	27.5 - 31	17.7 - 19.7

5.9.5 Link Budget

A link budget needs to be determined for the links of the previous section. To ensure a low probability of error in the received data of the CTTV, a certain Signal to Noise (SNR) ratio is required. The SNR is dependent on the applied modulation and coding technique. It is not known which modulation technique will be available, therefore a high required SNR will be chosen. Thus, different modulation techniques can be applied and it is assumed that the worst-case modulation technique is used (which is Frequency Shift Keying (FSK)) [60]. Based on SMAD [60], the required SNR is set at 13 dB, thereby assuming a low Bit Error Rate (BER). Next to the required SNR, there is the actual SNR which has to be calculated. In Eq. 5.33, it is shown that the difference between these SNR's is the link margin. The link margin should be higher than 0 dB, such that communications are still possible if unforeseen attenuation increases the losses.

$$LM = \frac{E_b}{N_0} - (\frac{E_b}{N_0})_{Req} \tag{5.33}$$

Before the link margin can be determined, the actual SNR needs to be calculated. This is done using Eq. 5.34. In this equation, EIRP is the Effective Isotropic Radiated Power, obtained from Eq. 5.35, L_{pr} is the receive antenna pointing loss obtained from Eq. 5.36, L_s is the space loss obtained from Eq. 5.38, L_a is the propagation and polarisation loss estimated at -0.6 dB [60], G_r is the receiver antenna gain obtained from Eq. 5.39, T_s is the system noise temperature in dB-K, obtained from SMAD [60] and R is the data rate in kbps.

$$\frac{E_b}{N_0} = EIRP + L_{pr} + L_s + L_a + G_r + 228.6 - 10 \cdot log(T_s) - 10 \cdot log(R) \tag{5.34}$$

By using the equation below, the EIRP is calculated. In this equation, P_T is the transmitter power, which is iterated (together with the diameter of the antennas), to obtain a suitable link margin. G_T is the transmitter gain, calculated using Eq. 5.39. L_l is the transmitter line loss, estimated to be -1 dB based on SMAD [60].

$$EIRP = P_T + G_T + L_l \tag{5.35}$$

With Eq. 5.36, the pointing loss can be calculated. In this equation, e is the pointing error, assumed to be half the half-power beamwidth, yielding an L_{pr} of -3 dB. The half-power beamwidth itself can be calculated using Eq. 5.37, where D is the diameter and f is the frequency in GHz.

$$L_{pr} = -12 \cdot (\frac{e}{\theta})^2 \tag{5.36}$$

$$\theta = \frac{21}{D \cdot f} \tag{5.37}$$

Eq. 5.38 yields the space loss. In this equation, S is the distance from the receiver to the transmitter and f yields the frequency in Hz at which data is transmitted and received. The distance between the CTTV and the ground segment was set at 340083 km, which is the distance from Earth to the patch point. After that, the CTTV will communicate via the relay satellites. The worst possible distance from the patch point to the relay satellites is the distance from the Moon to LL2 plus the distance from the patch point to the Moon, yielding a total distance of 125398 km. It is not taken into account that the distance to the relay satellite differs because of its orbiting motion, due to the small impact on the corresponding losses. The same counts for the deviation in distance from the Earth ground stations to the CTTV.

$$L_s = 147.55 - 20 \cdot log(S) - 20 \cdot log(f) \tag{5.38}$$

In Eq. 5.39, it is shown how the gain (G) is calculated. In this equation, D is the diameter of the antenna. The diameter is also iterated, just like the transmitted power (as these are the two variables in the link budget design). The frequency f is in Hz, and η is the efficiency of the antenna, assumed to be 0.55 based on SMAD [60].

$$G = -159.59 + 20 \cdot log(D) + 20 \cdot log(f) + 10 \cdot log(\eta) \tag{5.39}$$

5.9.6 Power, Mass and Cost Budget

The results of the calculations of the previous section can be found in Tab. 5.22. The input variables, power input and antenna diameter, were iterated. The bigger the antenna, the higher the gain and the lower the input power required. However, the downside of the high gain antenna is the small half-power beamwidth and corresponding required pointing accuracy. A low gain antenna on the other hand, requires more power input.

Because of the low data rates which have to be transmitted and received, the relatively low gain antennas are considered to be most suitable. They require less pointing accuracy. As can be seen in Tab. 5.22, the antenna diameter for communications between the CTTV and Earth is 0.1 m. The required power input is only 0.1 W. This yields a link margin of 28.3 dB, such that unforeseen losses will not be a problem. For redundancy, two of these antennas will be placed on the CTTV, yielding a total required power of 0.2 W. For an estimation of the corresponding mass, reference satellites from SMAD have been used [60]. A single antenna with transmitter would have a mass of 1.8 kg, yielding a total mass of 3.6 kg.

For communications between the CTTV and relay satellites, a 0.05 m antenna has been selected, requiring a power of 5 W. The reason for the small antenna is the higher half-power beamwidth, requiring a lower pointing accuracy to communicate with the relay satellites. The link margin is approximately 29 dB. For the worst case it is assumed that the CTTV has to communicate with both relay satellites at the same time. One antenna for redundancy yields a total of three antennas, requiring a total power of 15 W. The masses are again estimated using SMAD [60], yielding a total mass of 15.6 kg. Therefore, the total mass of the communication subsystem will be 19.2 kg. For the worst case, one of the antennas for communication with the ground station and two antennas for communication with the relay satellites have to be provided with power. This yields a worst case total power requirement of 10.1 W.

The cost of the communication subsystem carries a high uncertainty. It is assumed that the majority of the communication subsystem cost comes from the transponder. NASA has developed a transponder for deep space communications, which have a purchase cost of 0.587 M€ on provision of an order of at least five. As for this mission at least five of them would be needed (for the LSAM and the CTTV), the total cost for the CTTV communication subsystem would be 2.9 M€ as shown in Tab. 5.22.

Table 5.22: Link, mass and power budget for CTTV communication subsystem.

	Symbol	Units	Downlink to Earth	Intersatellite Link to LL1/LL2
Frequency	f	GHz	7.25	15
Antenna Efficiency	η	-	0.55	0.55
Transmit Antenna Diameter	D_t	m	0.1	0.05
Transmitter Power	P_t	W	0.1	5
Transmitter Power	P_t	dBW	-10	7
Transmitter Gain	G_t	dBi	15	14.7
Transmit Antenna Beamwidth	θ	deg	29	30
Transmitter Line Loss	L_l	dB	-1	-1
Equivalent Isotropic Radiated Power	EIRP	dBW	4	20.7
Free Space Path Length	S	km	340083	125398
Space Loss	L_s	dB	-220.3	-217.3
Propagation and Polarisation Loss	L_a	dB	-0.6	-0.6
Receiver Antenna Pointing Loss	L_{pr}	dB	-3	-3
Receiver Antenna Diameter	D_r	m	35	4.5
Receiver Antenna Gain	G_r	dB	68.9	51.1
System Noise Temperature	T_s	dB-K	21.3	27.9
Data Rate	R	kbps	400	400
Signal to Noise Ratio	$\frac{E_b}{N_0}$	dB	41.3	42
Required Signal to Noise Ratio	$(\frac{E_b}{N_0})_{Req}$	dB	13	13
Link Margin	-	dB	28.3	29
Total Power (incl. Redundancy)	P	W	0.2	15
Total Mass (incl. Redundancy)	m	kg	3.6	15.6
Total Cost (incl. Redundancy)	-	M€	1.2	1.7

5.10 Command and Data Handling

The on-board computer of the CTTV must handle many inputs, convert them to useful data, prepare them for downlink, and many other activities. Along with payload interaction control, the C&DH aspect of the CTTV will form its "brain". With such an important function to fulfil, it is important that the computer handling the command and data aspect is designed as redundantly and safely as possible.

5.10.1 Sizing

Before detailing the entire design of the C&DH system, a few elaborations on the requirements must be stated. It is important to stress that the CTTV has an expected lifetime of 30 years. This means that most systems on board are expected to function almost flawlessly for a very extended period of time. While maintenance of the CTTV at the Space Dock is planned for, this maintenance is not expected to be able to perform the most complex tasks. The astronauts performing the maintenance will be in bulky EVA suits that may hardly fit inside the CTTV, let alone perform maintenance tasks on sensitive systems. In order to avoid the need for these complex maintenance activities, a high level of redundancy and robustness must be included in the design. This makes for a very complex design, but one that is also able to withstand 30 years of space operations. An additional important factor for robustness of the system is the fact that the CTTV carries a considerable value of payload at once. A failing system would cause a huge financial loss, let alone a nearly dramatic loss of fuel for energy generation.

With these considerations in mind, the design of the C&DH system was performed according to the defined procedure [60]. During this procedure, it was systematically opted for the cases where the C&DH system is a very complex one. Therefore, the resultant system is the heaviest, most voluminous and power-drawing system. The results of the design are presented below.

5.10.2 Cost

According to the Lawrence Berkeley National Laboratory, complex C&DH can form a considerable cost driver for the mission. In the case of the proposed mission, the required C&DH system is not only complex, but must resist a very hostile environment for a long time, as explained above. According to LBNL, costs can quickly rise to 3 M€ [218]. On the other hand, CPUs used aboard the Mars Curiosity Rover cost upwards of 200 k€, while the ISS runs on simple i386 processors mostly [219]. Therefore, the cost of the on-board computer is estimated to be around 2 M€, including contingencies. This is a very pessimistic estimate, but the computer is essential to the functioning of the transport and mission.

5.10.3 Diagrams

In Fig. 5.16, the interaction of the C&DH board computer is shown with the rest of the system, along with fault checking protocols. Additionally, in Fig. 5.17, the data handling of the C&DH system is shown.

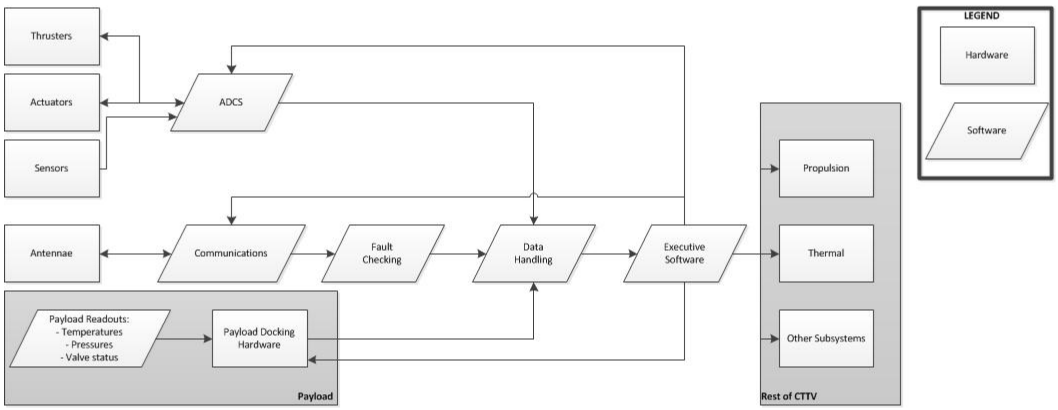
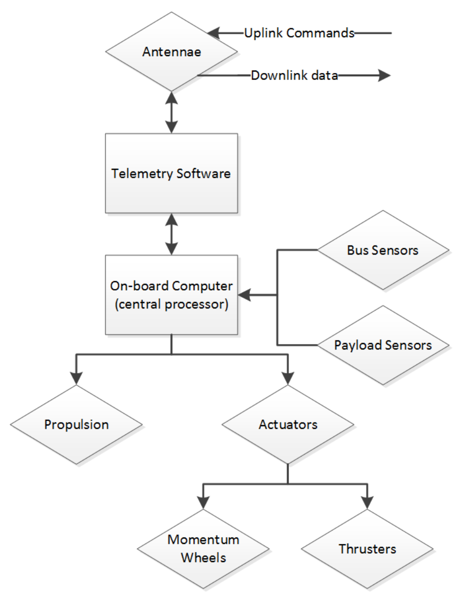


Figure 5.16: Hardware/software diagram for C&DH interactions in the CTTV.

Table 5.23: Overview of C&DH system properties.

Property	Value	Unit
Size	13 - 15	dm^3
Mass	9.5 - 10.5	kg
Power	15 - 25	W
Cost	2	M€

Figure 5.17: Data handling by CTTV C&DH board computer.



5.10.4 Summary

In Tab. 5.23, an overview is given of the properties of the command and data handling system. It must be noted that during the iterations of the mass and power budgets, the worst-case values were taken from the table.

5.11 Docking Procedures and Payload Handling

The CTTV is the passive vehicle for all docking operations. For docking of the LSAM to CTTV, see Sec. 6.11. The docking procedures with the Space Dock are shown in Sec. 4.9. Docking is necessary for payload transfer, maintenance, refuelling, and other operations.

The docking of CTTV to the Space Dock will only require berthing, not docking. One of the Space Dock's Canadarm2s will grab the CTTV once it is within reach. The CTTV will then be berthed to the docking port. For docking with the LSAM, the CTTV only needs to hold its position. The LSAM then docks using ATV-derived technology. For both docking links, an NDS is used. Using the same docking system for both operations reduces mass. The NDS has a mass of 340 kg.

5.12 Summary

The objective of the CTTV is to pick up four He-3 canisters from two LSAMs in LLO, transfer to LEO to exchange the four He-3 canisters with four empty canisters at the Space Dock and transfer back again taking supplies. This is a continuous operation where the refuelling will be performed at the Space Dock. For the overview of the mass and power budgets, see Sec. 7.2. The cost can be found in Sec. 7.4.

The CTTV has a dry mass of 55.68 tons and is able to carry four canisters of 43.08 tons in total. The total propellant mass needed for a round trip is 27.44 tons, of which 1.57 tons is the Hydrazine for the ADCS. The wet mass of the CTTV leaving the Space Dock is 81.54 tons. The total power consumption is 1.13 MW. Starting the transfer at 500 km LEO orbit with a 0° inclination the CTTV spirals in a circular orbit towards the patch point, simultaneously changing the inclination and semi-major axis in order to get in line with the Moon orbit around Earth. From the patch point the CTTV will get into an elliptical orbit, by slowly thrusting at certain intervals the final LLO orbit at 100 km altitude is obtained. The total required ΔV for LEO to LLO is 9 km/s, resulting in a ΔV of 18 km/s for the round trip. The total round trip takes about 354 days. This will require 22 CTTVs to be in orbit to deliver 88 canisters per year. The CTTV is equipped with 10 VASIMR engines, based on the design of the VF-200 engine. The total mass of the engines is 1.8 tons and the required propellant (Argon) for one round trip is about 26 tons. The CTTV's structure holds the entire system and subsystems together. Loads experienced during operations were minor compared to those during launch from Earth, therefore, the CTTV's structure was sized to endure these longitudinal and lateral loads. Frequency analysis showed that the natural frequency was high enough to endure vibrations exerted by the launch vehicle. It was determined that Aluminium 7075-T6 is the most favourable material for the structure.

The Thermal Control will keep the CTTV between 263 and 298 K using 15 layers of MLI for passive control. The active control is provided by two patch heaters of 25 by 25 cm to provide the additional 1.37 kW. For attitude determination and control the ADCS uses two types of sensors and two of each for redundancy, the CT-602 start tracker and the HG-1700 IMU. To adjust the position 20 N Hydrazine thrusters are used and for close proximity to the LSAM or Space Dock CMGs will be used. The GNS of the CTTV will determine the position using an ANS accurate to about 100m. In addition to this ATV-derived R&DS will be used for docking procedures. This system will use VGMs and TGMs to determine the position accurate to 15 mm. The power subsystem consists of a nuclear power plant and an MHD generator, providing 1 MW for the VASIMR engines and an additional 200 kW for the subsystems. An additional battery of 25 kg is needed to provide power in case of a failure. The total mass of the power plant is 4.2 tons. For communication the power, mass and cost are negligible compared to the total CTTV. UHF will be used to communicate with the LSAM and Space Dock, X-band for communications with the ground stations and relay satellites. The data rates are 400 kbps for downlink and 40 kbps for uplink. The C&DH will consist of an on-board computer which processes all the inputs and converts them to useful data. The size will be about 13 - 15 dm^3.

For the docking with the LSAM and Space Dock the CTTV contains two systems. The ASDS and a passive CBS will be used. This results in a structural connection between the two vehicles where the payload can be exchanged by the PERAs.

5.12.1 Scalability

The design discussed in this chapter is based on the 10% of the global energy demand by 2040 requirement, resulting in an annual He-3 supply of 200 tons. Concerning the scalability up to 50% this is achieved by upscaling the number of CTTVs in orbit to 110. This will require docking in LEO and LLO about every three days. For the scalability to 1 and 0.1% the same procedure as in the conceptual design of the CTTV will be used, see Sec. 4.6. For the 1%, or 20 tons He-3, requirement the design of the CTTV will be the same as for the 10% requirement. Only three CTTVs will be in orbit, reducing the annual amount of propellant required. For 0.1% the CTTV has to be redesigned to accommodate only one He-3 canister, as just one is needed on an annual basis (2 tons). This will include scaling down the propulsion system from 10 VASIMRS to 4-6, which increases the transfer time but reduces the overall mass.

6. Detailed Design: LSAM

For the detailed design of the CTTV, this chapter elaborates on the conceptual design of the LSAM as presented in Sec. 4.7. Conceptual values were used to design the subsystems. Multiple iterations yielded the results presented here. This chapter has the same setup as the presentation of the CTTV detailed design, starting with the subsystem requirements. The astrodynamic characteristics of ascent and descent from the Moon to LLO will be discussed, followed by the design of all the subsystems. A summary including a discussion on the scalability will be given. An illustration of the LSAM can be seen in Fig. 6.1. The integrated system is shown in Fig. 6.2

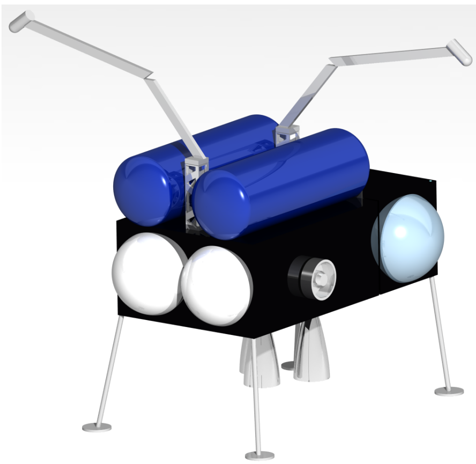


Figure 6.1: Illustration of the LSAM external layout.

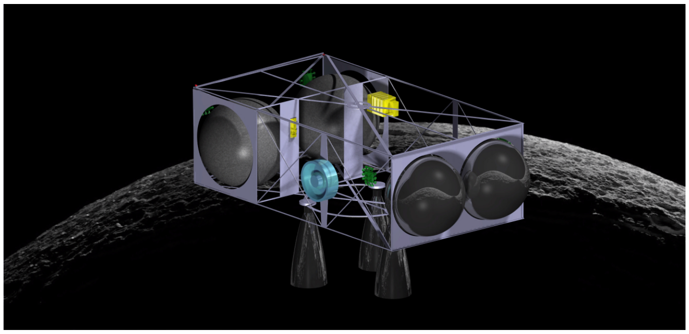


Figure 6.2: System integration of the LSAM. Docking system, CPU, Power, ADCS thrusters, Propellant tanks, Antennas (in the back), and Engines are shown. Payload and robotic arms are not shown.

6.1 Subsystem Requirements

Tab. 6.1 gives an overview of all subsystem requirements that have been identified throughout the detailed design process.

Table 6.1: Overview of all subsystem requirements for the LSAM.

Requirement-ID	Requirement description
LSAM-Prop-1	The propulsion unit shall provide enough thrust to provide a net launch acceleration of 1.36 m/s^2.
LSAM-Prop-2	The propulsion unit shall provide enough thrust to provide a net landing acceleration of 6.29 m/s^2.
LSAM-Prop-3	The propulsion unit shall provide a ΔV of 3810 m/s.
LSAM-Prop-4	The propulsion unit shall be restartable.
LSAM-Prop-5	The engines shall support throttle settings between 60% and 100%.
LSAM-Prop-6	The propulsion unit shall use liquid bi-propellants from in-situ resources.
LSAM-Prop-7	The oxidiser and fuel shall be stored separately as a liquid.
LSAM-Prop-8	The propulsion unit shall have contingency for one-engine-out failure.
LSAM-Prop-9	The propulsion unit shall operate all engines at lower than 100% throttle in nominal conditions.
LSAM-Struc-1	The structure shall provide structural support for the payload, propellant, and other subsystems.
LSAM-Struc-2	The structure shall sustain all loads acting on the structure over the course of the LSAM mission.
LSAM-Struc-3	The structure shall be mass-efficient.
LSAM-Struc-4	The structure shall have a safety factor of 1.5.
LSAM-Struc-5	The structure shall have natural longitudinal and lateral frequencies higher than frequencies of relevant rocket launchers.
LSAM-Struc-6	The structure shall provide load paths for the loads acting on the structure.
LSAM-Struc-7	The structure shall be easy to assemble on the Moon in EVA suits.
LSAM-Struc-8	The sstructure shall withstand landing on the lunar surface with a maximum velocity of 5 m/s.
LSAM-Therm-1	The thermal control system shall keep the spacecraft within the temperature range of 263-298 K.
LSAM-Therm-2	The thermal control system shall be able to provide adequate heating throughout the mission for all the various physical stages.
LSAM-Therm-3	Thermal control shall mainly be delivered by passive thermal control components.
LSAM-Therm-4	Thermal regulation shall be applied to all stages.
LSAM-ADCS-1	The ADCS shall contain enough propellant for the mission, including a margin factor of two.

LSAM-ADCS-2	The ADCS shall include contingency for the actuators and sensors.
LSAM-ADCS-3	The ADCS shall include separate sensors for rough and precise attitude determination.
LSAM-ADCS-4	The ADCS shall include different types of sensors.
LSAM-ADCS-5	The ADCS's thruster plumes shall not damage LSAM or CTTV.
LSAM-ADCS-6	The ADCS actuators shall be placed such to minimise unwanted rotations about other axes during rotation about an axis.
LSAM-ADCS-7	The ADCS shall have a determination accuracy of 0.1°.
LSAM-ADCS-8	The ADCS shall have a full determination range (360°).
LSAM-ADCS-9	The ADCS shall have a control accuracy of 0.1°.
LSAM-ADCS-10	The ADCS shall have a full control range (360°).
LSAM-ADCS-11	The ADCS shall limit jitter to 0.1°.
LSAM-ADCS-12	The ADCS shall limit drift to 4°.
LSAM-ADCS-13	The ADCS shall have a setting rate of 10°/min.
LSAM-GNS-1	The location of the LSAM shall be known with an accuracy of 100 m
LSAM-GNS-2	The LSAM shall have a rendezvous accuracy of 0.5 m
LSAM-GNS-3	The LSAM shall have a docking accuracy of less than 0.1 m
LSAM-GNS-4	The LSAM shall have a landing accuracy of 50 m
LSAM-Power-1	The power system shall deliver 66 kW for payload support and additional power for all the subsystems.
LSAM-Power-2	The power system shall support peak power demands with a secondary power source.
LSAM-Power-3	The power system shall use In-Situ Resources wherever possible
LSAM-Power-4	The power system shall include a regulation and charging system
LSAM-Power-5	The power system shall be capable to operate in a high-radiation environment for at least 7 years.
LSAM-TT&C-1	The subsystem shall be able to communicate with the ground stations.
LSAM-TT&C-2	The subsystem shall be able to communicate with the relay satellites.
LSAM-TT&C-3	The subsystem shall be able to communicate with the Space Dock/LSAM during docking procedures.
LSAM-TT&C-4	The subsystem shall be able to transmit data at a rate of 800 kbps.
LSAM-TT&C-5	The subsystem shall be able to receive data at a rate of 60 kbps.
LSAM-TT&C-6	The subsystem shall be able to communicate with a redundant band.
LSAM-C&DH-1	The data processing shall be done onboard.
LSAM-C&DH-2	The data processing shall accept external verification commands.
LSAM-C&DH-3	The data processing shall accommodate navigation data provided by the ANS.
LSAM-C&DH-4	The data processing shall accommodate navigation data provided by the VDMs.
LSAM-C&DH-5	The data processing shall accommodate navigation data provided by the TGMs.
LSAM-C&DH-6	The data processing shall accommodate continuous communication links.
LSAM-Dock-1	The docking shall establish a structural connection with the CTTV.
LSAM-Dock-2	The docking system shall exchange the two He-3 canisters with those from the CTTV.
LSAM-Dock-3	The docking system shall be placed such that exchange of payload canisters is possible.
LSAM-Dock-4	Rendezvous, docking and payload handling shall take no longer than 3.5 hours.
LSAM-Dock-5	The payload exchange shall take no longer than one hour.
LSAM-Dock-6	The docking system shall be compatible with CTTV docking system.
LSAM-Dock-7	The docking system shall be autonomously operable
LSAM-Dock-8	The docking system shall allow transfer of additional payload (supplies) from CTTV to LSAM.

6.2 Ascent and Descent Trajectory Design

This section will elaborate on the trajectories the LSAM will travel along. The objective is to deliver payload to the CTTV, which is orbiting the Moon in a circular 100 km altitude orbit. The LSAM reaches this orbit through a series of different manoeuvres. In orbit, payloads will be exchanged in roughly 3.5 hours (nearly 2 orbits of $P = 2\pi\sqrt{\frac{1838^3}{4902.8}}$ =7070 s each). Subsequently, the descent will be performed. Ascent and descent follow the same kind of trajectory. Accelerations during descent are higher than during ascent. Both ascent and descent are performed between altitudes of 0 and 20 km with the horizontal velocity being changed between 0 and 1688 m/s. The iterated values will be used here, which means that the LSAM has a lift-off mass of 108.7 tons and a dry mass of 39.1 tons. This dry mass includes payload and a propellant contingency of 1.9 tons. The dry mass differs by a few tens of kilogram from the final value, which is a negligible leftover iteration error. Furthermore, the LSAM is equipped with three 180 kN thrusters, delivering a total thrust of 540 kN at maximum throttle. All engines have to be throttled down to 60% for, yielding a thrust of 324 kN. The fuel has an I_{sp} of 380 s, yielding a mass flow of 86.9 kg/s. Thrust angle θ represents the angle between the horizontal and the thrust vector.

A model has been written to define both manoeuvres and to find parameters such as duration and required propellant. In this model, the mass varies over the mission and the flat surface assumption has been discarded. Additionally, the throttle setting is constant over the mission and the angles have been defined with very high accuracy, in the order of 0.01 degrees in some segments. This model uses an Euler integration with δt=0.1 s.

A massless object orbiting a celestial body is pulled towards the centre such that it will keep going around the centre of gravity of the body. This can be modelled as an object flying over a surface which exercises no gravity. As the object then decelerates, the gravity will gradually become more apparent. A relation was found between the relative apparent gravity and the horizontal velocity of the object by comparing the distance before the spacecraft would hit the surface with both the flat surface constant gravity assumption and relative gravity due to an elliptical orbit. Fig. 6.3 shows the results of this comparison. An average relative gravity from both approaches has been used in the model. Note that only velocities up to 1660 m/s were considered, due to discretisation errors. This also causes the right part of the bottom graph to diverge from the expected line. This was corrected manually as the focus is mainly on the higher velocities.

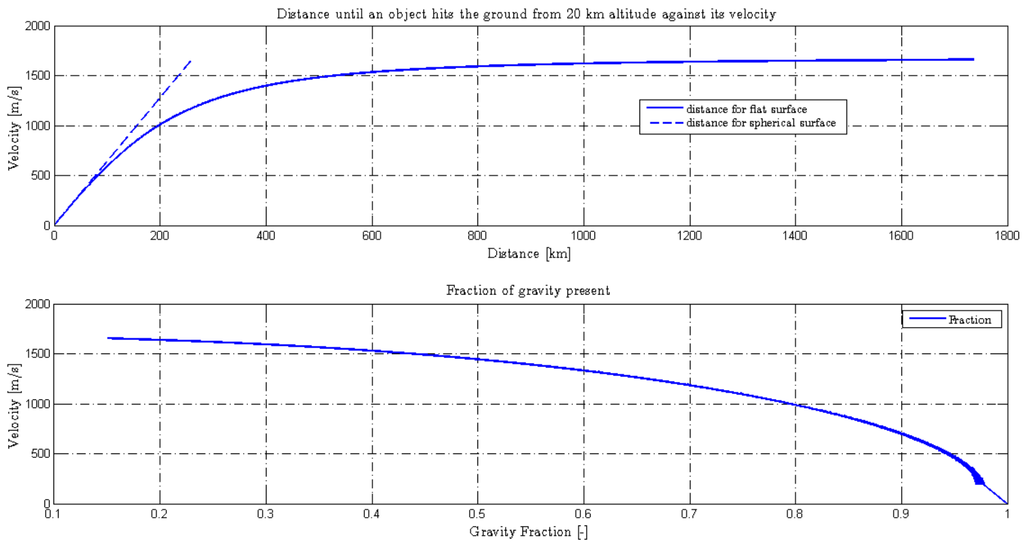


Figure 6.3: Comparison of gravity models.

Ascent: The mission starts with a vertical ascent to 150 m, after which it will direct its thrust angle 60.5° from the horizontal. This setting is then kept until an altitude of 1000 m. Then, the thrust angle is directed such that the vertical thrust component exactly counteracts the gravity pull. The LSAM continues coasting vertically. Horizontally, it accelerates to 1200 m/s. At the end of this phase the gravity pull is so small that it is neglected. It can easily be counteracted by a very small thrust angle ($< 10°$). Acceleration continues till a horizontal velocity of 1688 m/s at an altitude of 20 km is reached. At this point, the vehicle is at the perigee of the targeted 20 x 100 km orbit. It will travel half an orbit, before performing the last manoeuvre to reach LLO. There it will approach the CTTV for docking manoeuvres.

Descent: After the payload exchange has been performed, the LSAM will return back to the lunar base. It will descend to perigee of the 20 x 100 km orbit, where it will decrease its horizontal velocity to 800 m/s. During this phase, an average of the gravity pull has been chosen to be 60% of total gravity. The thrust is directed such that it exactly counteracts gravity, keeping the vertical velocity constant. When the horizontal velocity has been decreased to zero the vehicle is at 1500 meters altitude with a vertical velocity of about 100 m/s. With 60% thruster setting, the last phase should end at 0 m with a velocity of 0 m/s. One can see that there will be a small remaining height and velocity due to discretisation errors (1 m and 1 m/s respectively).

Results of both analyses are summarised in Tab. 6.3. Tab. 6.3 shows a summary of the key values of the model.

Table 6.2: Breakdown of both ascent and descent manoeuvres of the LSAM after iteration.

Segment [-]	h_0 **[m]**	h_1 **[m]**	V_{x_0} **[m/s]**	V_{x_1} **[m/s]**	V_{y_0} **[m/s]**	V_{y_1} **[m/s]**	**Time** [s]	**ΔV** **[m/s]**	m_{exp} **[kg]**	θ **[deg]**	**Throttle** [%]
Ascent											
1	0	150	0	0	0	20.18	14.9	44.3	1292	90	100
2	150	1004	0	38.6	20.18	46.6	25.7	78.3	2238	60.5	100
3	1004	17621	38.6	1200.2	46.6	46.6	356.2	1299	30959	31.79 - 21.83	100
4	17621	20018	1200.2	1688.0	46.6	-1.0	104.9	488.4	9117	2.56-2.24	100
5	20018	100000	1688.0	1633.2	-1.0	0	0	18.3	320	0	100
Total	-	-	-	-	-	-	**501.7**	**1928.3**	**43926**	-	-
Descent											
1	100000	20000	1633.2	1688.4	0	0	0	18.3	319	0	100
2	20000	12337	1688.4	799.7	0	-93.9	158	892.5	13732	5.2	100
3	12337	1497	799.7	-0.2	-93.9	-93.9	115.5	821.3	10039	14.72-11.77	100
4	1497	148	0	0	-93.9	-26.9	22.1	103.0	1109	90	57.5
5	148	1.0	0	0	-26.9	0.1	10.8	44.5	470	90	50
Total	-	-	-	-	-	-	**307.0**	**1879.6**	**25668**	-	-

Table 6.3: Summary of key simulation values.

General		**Variable**	**Ascent**	**Descent**
δt	0.1 s	ΔV	1928.3 m/s	1879.6 m/s
I_{sp}	380 s	M_0	108734 kg	64812 kg
$F_{T_{max}}$	540 kN	M_{end}	64812 kg	39144 kg
Throttle	60%	Total time	501.7 s	307.0 s
		Horizontally Travelled Distance	356.9 km	247.5 km

Verification of the Model

The model presented above has been verified. First of all, different sections were tested for input values of which the output was known or analytically calculated. For example testing the model for thrusting horizontally with constant mass (hence acceleration) until V_x=0 m/s would cause the spacecraft to hit the lunar surface after 157 seconds, which equals the time to drop 20 km. Perturbations were added to the values. The outcomes changed as expected. Furthermore, the output ΔV's can be compared to similar missions, like the Apollo landing modules. Apollo 17's lander required 2036.2 m/s for descent and 1863.4 m/s for ascent [220]. The offset is due to different accelerations. The required fuel mass given by the model was compared with values from Tsiolkovsky's equation, which yielded 43926 kg and 25667 kg of propellant for ascent and descent, respectively. This closely corresponds to the model output.

6.3 Propulsion

The LSAM brings two He-3 canisters into LLO, each weighting 10.77 tons. Annually, 44 of these LSAM launches are required for 200 tons of He-3. These values drive the choice of engines (in terms of thrust) and total propellant mass (and the fuel-to-oxidiser ratio in case of liquid bi-propellants). Additional astrodynamic specifications for this orbit can be found in Tab. 6.4. Each LSAM will launch multiple times during its lifetime. This introduces the requirement of using robust restartable engines.

Table 6.4: Orbit specifications for the ascent/descent trajectory (see Sec. 6.2).

Specification	Value	Unit
Inclination	9	deg
Height	100	km
Period	117.75	min
Velocity	1634	m/s
Eccentricity	0	-
Ascent ΔV	1928.3	m/s
Descent ΔV	1879.6	m/s
Required total thrust	324	kN

Table 6.5: Exemplary RD-185 engine and propellant specifications [168][221].

Specification	Value	Unit
Wet mass	415	kg
I_{sp}	380	s
Thrust	180	kN
Oxidiser-to-fuel ratio	3.45	-
Chamber pressure	146	bar
Overall diameter	1.5	m
Overall height	3.3	m

6.3.1 Propellant Requirement

Currently, only few available space propulsion types are available that provide enough thrust to launch heavy payloads. These are either chemical or nuclear thermal. Nuclear thermal propulsion uses the energy generated by a nuclear fission reaction to heat a working fluid, which is expelled. The I_{sp} range is 900 to 1100 s. This option will be discarded as to exclude the risk of radioactive waste/fallout and all supplementary safety issues. Chemical propulsion breaks up in either solid or liquid fuels. Solid propellants generate thrust from 50 N to 5 MN, at a specific impulse of 300 s. Liquid propellants are an alternative, generating similar thrust, but with higher I_{sp} (320-460 s), depending on which oxidiser-fuel combination is used [60]. Solid motors are simple, reliable, and relatively cheap, but the performance is not refillable with new propellant. Liquid engines provide additional choices concerning the propellant used. Either mono-propellant, bi-propellant or dual mode can be chosen [60]. Since dual mode is toxic and dangerous, and since mono-propellants have lower performance, bi-propellants will be used. A variety of chemical elements are possible as a fuel/oxidiser combination.

A design choice was made to use LO2/methane (CH4) as propellant. Such rocket engines have already been developed, and appear promising for future Moon and Mars missions. Research, ground testing and development have been performed in Russia, Japan and the USA in recent years [222]. Since the 90's Russia developed several LO2/methane engines, such as the RD-185 [222]. The United States have been testing its RS-18 to use LO2/methane instead of the intended Nitrogen Tetroxide (NTO)/Aerozine-50 [223]. XCOR Aerospace performed tests on the XR-5M15 methane engine in December 2006. There are several other reasons for choosing the liquid methane as a working fluid instead of other previous mentioned fuels:

- Methane is much more easier to handle than the well performing LH2. Boiling points of LO2 (90 K) and LCH4 (111 K) are closer to each other, which eases handling, insulation requirements, and engine design [166].
- LCH4 has a density of 422.62 kg/m^3 as compared to 71 kg/m^3 for LH2 thus allowing for more compact fuel tanks and overall bus structure.
- Methane has a better performance than most other bi-propellant fuels (such as MMH, aerozine, RP-1, etc.) which lowers propellant mass [221].
- Methane is available from ISRU from the lunar mining operations. While less abundant than H2, the considerable amount of total volatiles mined per hour provides sufficient methane[1].
- In addition to the propellants being non-toxic, the LO2/methane combustion process also provides "cleaner" combustion products (CO, CO2, and H2O) as compared to hyperbolic propellant combustion products [223].

One drawback of the LO2/LCH4 combination is that the performance is lower than that of LO2/LH2. The ideal specific vacuum impulse is roughly 372 s [221][225]. As of today, the use of staged combustion cycles can improve this value by some 5%, as high as 390 s [58]. Throughout the analysis, an I_{sp} of 380 s is assumed. Considering restartability of the engines, developments at, among others, TNO (Dutch Organisation for Applied Scientific Research) include promising ignition technologies for liquid propellants [226].

6.3.2 Component Selection and Sizing

It was already established in Sec. 6.2 that a total thrust of 324 kN (0.6·540 kN) is needed. This value was found using the average lunar gravity of 1.622 m/s^2. However, in the worse-case the gravity acceleration can be as high as 1.636 m/s^2 [227]. This would require an additional 2.8 kN. The RD-185 was taken as a reference, since it is the only fully developed methane engine of sufficiently good performance available today. Its capabilities are summarised in Tab. 6.5 [168][221].

[1] roughly 70 tons/h of H2 as compared to roughly 18 tons/h of CH4. 44 launches per year would translate into an equivalent of 199 hours of mined volatiles per LSAM flight. This would mean that 13 854 tons of H2 could be available per flight, or 3 634 tons of CH4. Theoretically 20 000 tons of oxygen is present in the lunar volatiles, albeit in the form of H2O, CO2 and CO. Colson and Haskin elaborate on how oxygen can be produced with the use of so-called "silicate melt electrolysis" [224].

The engine configuration will be such that a redundant engine is included to account for a one-engine-out scenario during flight. Therefore, the engines must be capable of a range of throttle settings. An alternative to dealing with an engine failure is to let the remaining engines run longer, consuming more propellants. As rendezvous with the CTTV and precise de-orbiting to return to the base are time-critical mission parts, contingency for engine-out is not included in propellant mass, but in the number of engines. Three engines are mounted on the LSAM. All three are active during ascent and descent. They have to be able to throttle from 60 to 100%. The total mass is 1245 kg. The success rate for liquid cryogenic rocket engines can be assumed to be 96.64% [228]. If the LSAM were to be designed with two engines[2], three LSAM engines would fail in the worst case. Without contingency, up to 4.56 tons of He-3 would be lost or irrecoverable (6.8% of the total amount of He-3). The engines are gimbal-mounted.

6.3.3 Propellant Tank Volume

Knowing the I_{sp} of the chosen propellant (380 s), the propellant masses can be computed. Tab. 6.6 summarises these. They were found by iteration, as explained in Sec. 6.2. Additional specification for the oxidiser and fuel are also given, as well as their abundance. Also, 1.9 tons extra propellant is included as a contingency.

Table 6.6: Propellant characteristics and mass breakdown.

Element	Value	Unit
Propellant ascent	45.21	tons
Propellant descent	26.37	tons
Total propellant	71.58	tons
Liquid oxygen boiling point	90	K
Liquid methane boiling point	111	K
isru amount of oxygen from H2O, CO2 and CO	20 000	tons/flight
isru amount of methane available for 1 launch	3 356	tons/flight

Table 6.7: Propellant mass and tank volumes for different number of tanks.

Specification	Value	Unit
Oxidiser mass	55.49	t
Fuel mass	16.09	t
Density oxidiser	1141	kg/m^3
Density fuel	442.62	kg/m^3
Oxidiser volume	48.64	m^3
Fuel volume	36.34	m^3
Two tanks each		
Oxidiser tank radius	1.80	m
Fuel tank radius	1.63	m
Oxidiser tank mass	114	kg
Fuel tank mass	94	kg
Total tank mass	416	kg

The fuel and oxidiser mass and volume are found by assuming an oxidiser-to-fuel ratio of 3.45 (from Table 6.5), as specified for the developed RD-185 engine and confirmed by the respective optimal combustion process [221]. The pressure will be taken as one atm and spherical fuel tanks will be used. Tab. 6.7 summarises the results.

It was decided to user two tanks for fuel and oxidiser each. The structural mass of the tanks has to be found. The minimal thickness was found from contained pressure only, using Eq. 6.1. The pressure was assumed to be one atm, simplifying the storage of the fuel and oxidiser as liquids. The yield strength (stress) was taken as 275 MPa (Al 7075-0, [229]) and the radius was taken from Tab. 6.7.

$$t = \frac{pa}{2\sigma} \tag{6.1}$$

Substituting values results in a thickness of 0.33 mm. For contingency, one mm was used, corresponding to a oxidiser tank mass of 114 kg and a fuel tank mass of 94 kg. The tanks masses amount to 416 kg. It is acknowledged that these values are only the absolute minimum, as usually liquid oxidiser and fuel tanks take up around 7% of the total propellant mass and that propellants are usually stored at higher pressures and/or with the addition of turbopumps to the feed system [60].

6.4 Structures

The structural subsystem of the LSAM has four main elements. These are the landing legs, support of the payload, support of the robotic arms, and primary and secondary bus structure. Initial values for the subsystem design are taken from the conceptual design. After multiple iterations of the LSAM subsystems, the values presented in this section were found. The relevant input values are shown in Tab. 6.8.

[2] 44 flights, 2 engines used each flight, 144 engines used in total

Table 6.8: LSAM input values for subsystem design, showing conceptual values and last iteration values.

Element	Conceptual value	Final iteration input	Units
Launch Mass	116.8	108.73	tons
Dry Mass	46.1	39.05	tons
Launch Acceleration	2.5	1.4	m/s^2
Landing Acceleration	6.5	6.3	m/s^2
Thrust	3x 110	3x 180	kN
Payload Mass	23.98	23.98	tons
Robotic Arm Mass	2x 0.8	2x 0.950	tons

6.4.1 Assumptions

- The loads are quasi-static.
- Maximum thrust acts without an opposing gravity contribution (i.e. thrusting while in orbit).
- Attitude control loads are negligible.
- Loads from subsystems other than engines, propellant tanks, propellant, and payload are negligible.
- Loads due to perturbations, docking, thrust vectoring, and thrust misalignment are negligible.
- External shear loads are negligible.

6.4.2 Load Cases

The mission of the LSAM has three segments, launch, docking, and landing, with different quasi-static loads. At launch, the LSAM is accelerated by a force of 152.2 kN. At landing, the dry mass (all propellant burned) of 39.05 tons has an acceleration of 6.3 m/s^2. This gives a force of 246.0 kN. With a gravitational acceleration of 1.62 m/s^2 the ground force is maximum for the fully fuelled LSAM [230]. The force is 176.1 kN. Forces at burn-out and de-orbiting are lower than launch and landing loads. However, the maximum acceleration occurs for landing, and mandates loads for separate support systems (such as payload support). In the following calculations, a safety factor of 1.5 is applied.

Furthermore, loads act on the structural: docking connection with the CTTV due to disturbance forces acting differentially on LSAM and CTTV. The effect of disturbance forces are not considered here. During payload exchange, torques of the robotic arms act on the support structure. As shown in Sec. 6.11, the torques will be in the order of 16 Nm. Another type of load, other than quasi-static loads, requires attention for the LSAM. Vibrations can cause severe damage and structural failure for system operating in an atmosphere-free environment. Natural frequencies should not correspond to vibrations imposed on the structure by the thrust. As these imposed vibrations of the engines are unknown, the structures cannot be sized for vibrations. A check of the natural frequency of the LSAM with respect to Ariane V vibrations will be conducted at the end of this section. For material selection, the same three materials (Aluminium, Titanium, and Steel) as for the CTTV structure are considered (see Sec. 5.4).

6.4.3 LSAM Landing Legs

The most prominent loads acting on the landing legs are the launch load and landing impact load. A reasonable maximum value for landing velocity is 5 m/s, based on the Apollo Lunar Lander [231]. With the landing mass (39.05 tons) this gives an impulse of 195 255 kg·m/s. In particular, the amount of time over which the landing structures dissipates the impulse drives the imposed force. The LSAM shall dissipate the 5 m/s landing within 0.1 seconds, without putting the rest of the structure under high stress (i.e. by using dampeners). This yields to a force of 1 952.6 kN.

The landing legs are placed under an angle of 10° to increase stability. Correspondingly, normal- and shear forces act on each of the four landing legs (total normal force 2884.3 kN, total shear force 508.6 kN, not including safety factor). The legs will be solid cylindrical rods, bridging a vertical distance of five meter (5.08 m length, accounting for the angle). This gives enough margin for the engines. As materials, Aluminium and Titanium have been considered. Types of failure are compressive/tensile failure and buckling failure. Bending is neglected due to the low angle of 10° and low shear force. It was found that buckling is the critical failure mode. Designing for buckling gives, for each leg, a radius of 6.4 cm for Aluminium (183.3 kg) and 5.7 cm for Titanium (232.8 kg). Due to the lower mass, Aluminium legs will be used.

6.4.4 Payload Structural Support

The payload canisters are supported by a panel near the centre of the LSAM, which is connected to the bottom of the canister, and a pin at the head of the canister. Each payload canister has a maximum mass of 11.99 tons (10.77 payload & 1.22 redundancy) and a length of 8.55 m. The maximum load on each canister due to acceleration at landing is 113.2 kN (including safety factor). This shear force acts on the support panel. Assuming uniform mass distribution of the payload gives a moment of 483.8 kNm. Precise determination of the distribution of forces and moments on the two connections (pin near head of canister and connection at bottom) is beyond the scope of this detailed design.

The load case is approximated by assuming that half the force is carried by the pin. This counteracts the moment, as the pin force is acting over almost the entire length of the canister. The area over which the shear force at the support panel acts is approximately equal to the diameter of the connector. The connector is identical to the one on the CTTV. The distance of the centre of the connector to the LSAM bus structure is 1.5 m. Each payload support panel extends down to the bottom of the bus structure. The stress depends on the total area and thus on the thickness of the support panel. It was found that buckling is the critical load case. Sizing for buckling gives, for Aluminium and Titanium, thicknesses of 1.3 cm and 1.1 cm, respectively. The other dimensions are one meter in width, and six meters in height (height of the main bus plus distance to centre of connector plus half distance to upper end of connector). Corresponding masses are 217.9 kg for Aluminium and 297.2 kg for Titanium. Aluminium is chosen for the payload support panel. The pin near the head of the canister has to be sturdy enough to sustain the loads and transfer them to the underlying structure. The load is 56.6 kN (including safety factor). Structural Steel A36 is used. Only compressional/tensional stresses are considered due to the limited length of the pin. The required diameter of the pin is 1.7 cm. Length and mass are negligible.

6.4.5 Robotic Arms Structural Support

Both Payload Exchange Robotic Arms (PERAs) have to be be supported by a structure. During launch and landing, compressional/tensile loads act on the support. During in-orbit operations, only torques act on the support. Each PERA has a mass of 950 kg. With a maximum acceleration of 6.3 m/s^2 this gives a load of 9.0 kN (including safety factor). The failure mode of this structure is buckling. It is modelled as a solid circular rod of three meters height (equal to canister diameter). For Aluminium, this gives a diameter of 3.3 cm with a mass of 7.1 kg. For Titanium, the values are 2.9 cm and 9.1 kg. Before choosing either of the two materials, the moment has to be considered. Typical torque during operations is 16 Nm (see Sec. 6.11). The maximum stress occurs at the edge of the cross section. With above dimensions, the bending stress for an Aluminium rod is 2.3 MPa. A Titanium rod has a maximum bending stress of 0.6 MPa. Aluminium is chosen due to the resulting structure's lower mass.

6.4.6 Bus Structure

Structural sizing of the bus needs to take into account placement and attachment of other subsystems. Only subsystems with significant masses have to be considered. These are engines, propellant tanks, propellant, and payload (with the load being transferred to the bus via the structure presented in Sec. 6.4.4). The layout of the main bus structure can be seen in Fig. 6.4. The bus structure consists of rods or beams along the edges. One of the side panels has additional support struts for attaching the docking mechanism. Front and back panels have additional support that enclose the propellant tanks. Note that fuel tanks are mounted forward and oxidiser tanks are mounted in the rear. Each tank is also connected to adjoining struts of the top and bottom panels (not shown). Each tank is supported by six connections in total. The three engines' main mounting points are along a circular structure in the bottom panel, which connect, via a structural cone (not shown), to a smaller circular structure on the top panel. This configuration distributes the thrust more evenly over the bus structure. Engine mounting ring of both top and bottom panel connect to the top and bottom outer frame via six beams. The skin only carries shear loads. This allows for easy assembly of the disassembled LSAM on the Moon as well as design flexibility for cut-outs and other interfaces from the main bus to the outside (such as thermal subsystem, maintenance interfaces, etc.). Shear forces are not considered here, as they occur due to (negligible) thrust misalignment's or perturbations.

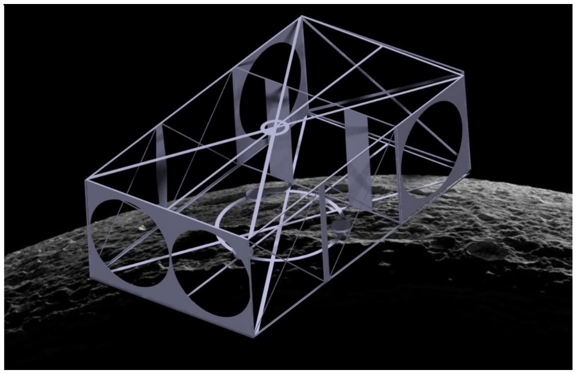


Figure 6.4: Layout of the LSAM main bus structure. The pads in the bottom panel indicate the engine positions.

The maximum thrust during the LSAM mission is 324 kN (worst case, in orbit, thus no gravity contribution, see Sec. 6.3). Via the central cone structure, approximately one-third of this force is transferred to the top panel. Struts of the top panel, the side panel, and the vertical struts of the frame are presented separately. In the following, a safety factor of 1.5 is applied. If not explicitly stated otherwise, Aluminium is used as material. Titanium is generally more expensive and harder to manufacture and yields, for this structure, very small mass savings (in the order of tens of kilogram). Cross sections of the struts are square. This allows easier assembly on the Moon[3] and easier attachment of skin panels than circular cross sections .

Top Panel: The top panel distributes a thrust force of 108.0 kN (including safety factor). Each of the six struts connecting to the engine mounting ring carries approximately one-sixth of the load. The moment arm of each strut is measured from the centre of the engine mounting ring. Additional loads act on the struts, creating moments. The straight struts support one-sixth of the load of both fuel tanks (strut to the front) or oxidiser tanks (strut to the rear), respectively. Each diagonal strut forward transports the load of the payload connector pin and one-sixth of one fuel tank. Diagonal struts rearward transport half of the load of the payload support panel (the other half is carried by the bottom panel) and one-sixth of one oxidiser tank.

Table 6.9: Characteristics of the top panel struts.

Strut	**Loads**	**Side length [cm]**	**Mass [kg]**
Straight to front	Thrust, two fuel tanks	11.0	210.8
Straight to back	Thrust, two oxidiser tanks	11.0	210.8
Diagonal forward x2	Thrust, payload pin connector, one fuel tank	4.2	35.0
Diagonal rearward x2	Thrust, half of payload panel connector, one oxidiser tank	9.5	179.2

Bottom Panel: The bottom panel distributes a thrust force of 216.0 kN (including safety factor). Each of the six struts connecting to the engine mounting ring carries approximately one-sixth of the load. The moment arm of each strut is measured from its intersection with the engine mounting ring. Additional loads act on the struts, creating moments. The straight struts support one-sixth of the load of both fuel tanks (strut to the front) or oxidiser tanks (strut to the rear), respectively. Each diagonal strut forward transports one-sixth of the load of one fuel tank. Diagonal struts rearward transport half of the load of the payload support panel (the other half is carried by the top panel) and one-sixth of one oxidiser tank.

Table 6.10: Characteristics of the bottom panel struts.

Strut	**Loads**	**Side length [cm]**	**Mass [kg]**
Straight to front	Thrust, two fuel tanks	10.9	99.8
Straight to back	Thrust, two oxidiser tanks	10.9	99.8
Diagonal forward x2	Thrust, one fuel tank	12.7	218.5
Diagonal rearward x2	Thrust, half of payload panel connector, one oxidiser tank	6.8	62.4

6.4.7 Vertical Struts

Due to symmetry both front vertical struts and both rear vertical struts carry the same normal load. Only the four corner struts carry the load; side panel vertical struts are not considered. For bending load, front and rear struts carry different loads. Each vertical corner strut of four meters length carries one-fourth of the top panel thrust in tension and one-fourth of the bottom panel thrust in compression. As thrust is split one-third and two-third between top and bottom panel, the resulting force is the difference between both forces (27.0 kN), and acts in compression. Additionally, the moments of the top diagonal and bottom diagonal act on the strut. The strut is designed such that it carries all of the resulting bending load. The cumulative moment on each front vertical strut is 166.7 kNm. Bending is the critical load (as oppose to normal force or buckling). For Aluminium, a square cross section of side length 12.6 cm is required (177.7 kg). Titanium would require 10.0 cm side length (176.8 kg). The cumulative moment on each rear vertical strut is 97.4 kNm. This relatively low bending moment is due to the oxidiser tank mass and payload support panel introducing a moment opposite to the thrust moment in the top and bottom panel. Bending is the critical load. For Aluminium, a square cross section of side length 10.5 cm is required (124.2 kg). Titanium would require 8.4 cm side length (123.6 kg).

6.4.8 Overview of Structural Components

The primary structural components have been designed for quasi-static loads. The components are the landing legs, the payload support (panel and support pin), support structure for the PERAs, and the main structural links in the bus.

[3] Most probably, the LSAM will have to be transported in modules, as it is too large to fit in any existing payload fairing. Some LTRs could be reconverted to "assembly rovers" as to facilitate this assembly.

Table 6.11: Overview of LSAM structural components.

Element	**Dimensions**	**Mass**	**Material [kg]**
Landing Legs x4	radius 6.4cm x 5.01m	183.3	Aluminium
Payload Support Panel x2	1m x 6m x 1.3cm	217.9	Aluminium
Payload Support Pin x1	Diameter 1.7cm	negligible	Steel A36
Robotic Arm Support x2	Diameter 3.3cm Height 3m	7.1	Aluminium
Top Bus Panel straight front	side length 11.0cm x 6.2m	210.8	Aluminium
Top Bus Panel straight back	side length 11.0cm x 6.2m	210.8	Aluminium
Top Bus Panel diagonal forward 2x	side length 4.2cm x 7.1m	35.0	Aluminium
Top Bus Panel diagonal rearward 2x	side length 9.5cm x 7.1m	179.2	Aluminium
Bottom Bus Panel straight front	side length 10.9cm x 3m	99.8	Aluminium
Bottom Bus Panel straight back	side length 10.9cm x 3m	99.8	Aluminium
Bottom Bus Panel diagonal forward 2x	side length 12.7cm x 4.8m	218.5	Aluminium
Bottom Bus Panel diagonal rearward 2x	side length 6.8cm x 4.8m	62.4	Aluminium
Vertical Strut Front 2x	side length 12.6cm x 4m	177.7	Aluminium
Vertical Strut Rear 2x	side length 10.5cm x 4m	124.2	Aluminium

Struts other than the ones presented above are not main load-carrying. Loads acting on them are secondary, such as side forces due to ADCS thrusting. Based on the dimensions of - and loads on - the primary structural elements (Tab. 6.11), a side length of six centimetres for secondary structural elements is reasonable. These struts cover a total length of 36.25 m. With Aluminium as material, the total mass of the secondary bus structure is 1420.3 kg, with a centre of mass in the centre of the bus. Total mass of the structural subsystem adds up to 4818.8 kg.

6.4.9 Vibrations

In the following, the natural frequency of the LSAM is computed and compared with frequency values of the Ariane V, as it represents the most prominent European launch technology. The Ariane V launch vehicle has a longitudinal frequency of 27 Hz and a lateral frequency of 7.5 Hz (see Sec. 5.4). The natural frequency of the LSAM needs to be higher than these values. Natural frequency of the LSAM is computed with Eq. 6.2.

$$f = \frac{1}{2 * \pi} \cdot \sqrt{\frac{k}{M}} \tag{6.2}$$

Here, M is the mass of the LSAM. The highest mass corresponds to the lowest frequency. The LSAM is modelled as a block of mass 108 tons, supported by three springs (corresponding to three engine suspensions), each with stiffness k. Total stiffness is given by Eq. 6.3

$$k = 3 \cdot \frac{EA}{L} \tag{6.3}$$

Here, L is the distance to the centre of gravity (approximately 3 m from the rocket engines). The Young's Modulus E is set at the value for Aluminium (71.7 GPa). The variable A denotes the area of the structure. The value for the primary bottom panel structure is 206.1 m^2. A factor three is added to account for the three rocket engines. Combining the above equations gives a natural frequency (longitudinal) of 5204 Hz. Lateral natural frequency is modelled in the same way (three rocket engines, side area of primary structure). This gives 458 Hz. Both longitudinal and lateral natural frequencies of the LSAM are well above the constraints estimated from Ariane V, and possibly well above most other rocket engines' vibrations. These frequency calculations are approximations of the highly complicated vibration environment of the structural subsystem. For example, the interaction of bottom and top engine mounting and the conical connection element are not taken into account. These elements increase stiffness of the structure, increasing the natural frequency. These approximations indicate that the overall LSAM structure is not very sensitive to vibrations.

6.5 Thermal Control

For the LSAM vehicle a thermal analysis and design has to be performed as the vehicle is exposed to different types of radiation. For the thermal analysis of the payload the same method is applied as for the detailed design in Sec. 5.5. As the LSAM is only present in lunar orbit or at the lunar surface, the cases observed will be the hot and the cold case for the Moon only. The temperature in which the LSAM has to operate is from 263 K to 298 K as this is the operating limit of the batteries which are the critical elements on board. The LSAM thermal analysis is analysed as a 2 node spacecraft, where the propellant tanks are separated from the rest of the spacecraft vehicle, since the propellant tank has to be sustained under low temperatures of 90 K. The power system of the LSAM is given to be a hotspot in the design as a large amount of power dissipates here. The working temperature of the fuel cell of 353 K might be used for heating the other components of the spacecraft. The surface area exposed for the LSAM is determined to be 115 square meters, since the total surface area is 345 square meters.

6.5.1 Assumptions

- The payload will be treated as a double node.
- The propellant storage is assumed to have a separate thermal control system to maintain its low temperatures.
- Moon albedo factor equals 0.07 [58].
- Surface Area exposed to albedo, solar radiation and planetary radiation is 0.33 percent of the total surface area.
- Distance Moon-Sun is assumed to be 1 AU.
- The effective radiating surface is assumed to be equal to the radius of Earth/Moon.
- Solar radiation and Albedo will equal to 0 in the shadow cases.
- Other celestial body radiation is neglected (only Moon and Earth considered).
- The LSAM will be stored at the lunar surface to prevent contamination of the surfaces and protect. against the lunar temperatures ranging from 90 to 400 K.
- Total dissipated power (in form of heat) is assumed to be 3500 W (5% of total electrical power).
- Good surface fitting between the heat pipes and the LSAM panels is achievable.
- Contamination of the surfaces and finishing will not be taken into account as the LSAM is sheltered on the Moon and the LSAM will not have surfaces sensitive to contamination.

6.5.2 Thermal Subsystem Design

Using the method described in Sec. 5.5 the equilibrium temperature of the LSAM can be calculated, with the results presented in Tab. 6.12. First an analysis is run without accounting for the internally dissipated power as this is assumed to be small, with the purpose to design for the passive thermal control. This results in a range of absorptivity over emissivity ratios, which have to be in the range of 0.4 to 0.8 in order to get a cold case temperature of 207 K and a hot case temperature of 298 K (which is the maximum allowable temperature). Afterwards the internally dissipated power can be included, which provides the range of temperatures for different ratios. In a list of several surfaces and finishes for thermal control [58], it can be found that a value of 0.63 for absorptivity over emissivity can be only provided by an aluminized kapton (kapton outside) surface.

Table 6.12: Equilibrium temperature payload canister.

Case	**Equilibrium temperature [K]**
I: Hot	283.5
II: Cold	207.4

Table 6.13: Power requirement thermal subsystem.

Case	**Power required [kW]**
I: Hot	0
II: Cold	11.5 (heating)

As mentioned earlier, the thermal design was critical for the batteries that had to operate between 263 and 298 K. The LSAM temperature with aluminized kapton surface will not exceed the maximum allowable temperature. However, the coldest scenario for the LSAM is not within the limits. Therefore heating equipment is needed for the cold case. Using the formula's stated in Sec. 5.5 the power requirement can be calculated for the heater, the results are given in Tab. 6.13.

The internally dissipated power is assumed to be 5% for electronic devices, resulting in 3755 W for the subsystems altogether. According to the thermal balance equation mentioned in section 5.5 the heating power required equals 11.5 kW to cope with the cold case operating aspects of the LSAM mission. This heating power will be delivered by active control components. Passive thermal control is applied to the spacecraft by using MLI. For large surface areas the emissivity of these layers will be in the range of 0.015 - 0.03. This depends on the amount of insulation layers applied. In order to obtain an effective emissivity of 0.015, 20 layers are used [60], approaching the optimum for practical values. The heat kept within the LSAM by the MLI is then calculated to be 312 W.

An active thermal control component is the fuel cell system waste water pipe. This heater pipe is made out of highly thermal conductive material, which is chosen to be copper for this configuration. The waste water of the fuel cell system has a temperature of 353 K, since this is the working temperature of the fuel cell. These conductive pipes are led towards the outer panels of the LSAM structure to deliver the heat along the spacecraft. Through the conductivity equation of two panels that fit well it follows that another 4.2 kW of heat is feeded by the heat pipes. The length of this pipe is brought to 96 m to cover the four of the 12 m long side panels of the spacecraft.

$$Q_{pipe} = \frac{T_{req} - T_{cold}}{s_1/k_1 A + s_2/k_2 A} \tag{6.4}$$

In this equation s is the thickness of both attached material, A is the heat transfer area and k is the thermal conductivity of the material. This is 401 W/mK for a copper water-filled pipe and 0.06 for aluminized kapton. The thermal conductivity of aluminized kapton was given by $4.638 \cdot 10^{-4}$ $T^{0.5678}$ where T is the desired minimum temperature [232]. The remaining 7 kW has to be delivered by active thermal control. The often used option is the use of heaters. Again patch heaters of 25 by 25 cm that can provide a power of 1 kW each are used [59]. For the LSAM this comes to eight installed patch heaters, because of 95% efficiency and redundancy perspective.

6.5.3 Budgets Thermal Subsystem

The power requirement of the thermal subsystem follows from the coldest temperature encountered and the power needed for heating, which was 7 kW. For the mass of the MLI a density of 0.73 kg/m^2 was used, which leads to a mass of 70 kg. For the 96 m long copper pipe with a diameter of 0.03 m and density of 8933 kg/m^3 a mass contribution of approximately 600 kg is the outcome. The patch heaters require a total area of 8·0.25·0.25 m^2= 0.5 m^2. With a small thickness, the mass contribution of the heaters is assumed to be negligible compared to that of the heat pipe and MLI components.

The costs per heater will be around 68.5 € and the heat pipe costs are based on the copper price in 2013, resulting in 3150 € for the heat pipe. Kepton polyamide tape is used for the MLI, which costs 65.21 €/m^2 [62]. Covering the total internal surface area with 20 layers will imply a cost of 450 k€. It should be noted that the thermal subsystem indirectly induces high costs due to the high power requirement, this will translate in higher costs for the power subsystem. The costs for the thermal subsystem amounts to 450 k€.

Table 6.14: Summary thermal subsystem budgets.

Budget	Value	Unit
Power	7	kW
Mass	670	kg
Cost	450	k€

6.5.4 Thermal System Layout

The eight heaters will be located at the interior of the outer panels, two at each of the side panels. The heat pipe needs to be attached to the power system and should be led around the LSAM, preferably to the outer panels as these will be cooled down. The outer surface of the LSAM will be completely covered with 20 layer MLI material to maintain the inside temperatures for the cold case. For further research it is recommended that the thermal subsystem will be split up in nodes, such that the heats within the spacecraft can be distributed among all the subsystems, instead of being restricted to the LSAM overall temperature during the thermal design.

6.6 Attitude Determination and Control

Two segments of the LSAM mission profile require high-precision ADCS: the docking with the CTTV and landing at the lunar base. The latter uses a separate set of instruments and thrust vectoring of the gimbal-mounted engines and is not considered here (see Sec. 6.7). Launch from the lunar surface, orbit insertion and de-orbiting (exclusive the actual descent phase) do not require high-precision ADCS. The ADCS primarily has to be designed for docking. The requirements for the docking are similar to the ADCS requirements of the CTTV (see Sec. 5.6). Attitude determination requirements call for precise star trackers [60]. As the LSAM is the active vehicle during docking, the used IMUs are more complex and sensitive than the ones used for the CTTV. Additionally, (Moon) horizon sensors, sun sensors, and a visual identification system (for lunar topographic landmarks) will be used. The latter is especially relevant for descent and landing (also see Sec. 6.7). The setting time requirement translates to a slewing requirement of 0.167°/s. With 20 s pulse time for acceleration and deceleration, respectively, this gives a required acceleration of 30 arcsec/s^2. This and the other demanding requirements make it necessary to install a three-axis control system, using thrusters [60]. ADCS actuators are placed such that they maximise the distance to the centre of gravity and can be fired such that the net force on the LSAM is zero and only rotation occurs. However, thruster plumes must not hit the external parts of the propellant tanks. Thrusters will be placed in blocks. Each thruster block (or Reaction Control Block, RCB) will feature four thrusters on the outside and oriented in the same plane but perpendicular to each other. One RCB each will be placed near the front of each side panel. Two more RCBs will be placed on either side of the rear panel. The front RCBs feature an additional thruster, pointing perpendicular to the side panel. These are required to balance the net force produced by the rear thrusters for rotations about the z-axis. The thrusters are the only actuator that can meet the requirements. For contingency, each thrusting direction of each RCB will be equipped with an additional thruster.

The position of the centre of gravity varies over the mission due to propellant consumption. For designing the actuators, the worst-case distance for each RCB will be used. One Control Moment Gyro (CMG) is located near the power subsystem (thus reducing resistance power loss). A redundant CMG is placed near the docking system. Note that the reaction wheels will primarily be used for attitude control in close proximity to the CTTV, when some of the thrusters cannot be fired due to the plume hitting the CTTV.

6.6.1 Center of Gravity and RCB placement

Viewed from the side, the centre of gravity is located between 5.12 and 7.44 m from the front and between 2.91 and 4.3 m from the bottom. Viewed from the top, the centre of gravity is located between 3.74 and 3.76 m from the docking side. The side-panel-RCBs are placed at 0.3 m distance from the front and 3.5 m from the bottom. The rear-panel RCBs are placed 0.3 m from the respective side edge and 3.5 m from the bottom. Vertical placement of the RCBs is relevant to reduce undesired torques over the other two axes during manoeuvres over the third axis. However, this undesired torque cannot be completely avoided due to the shift of the centre of gravity. Additional ADCS propellant has to be accounted for in order to counter these rotations.

Rotation about the x-axis: The front RCBs have a worst-case distance of 3.24 and 3.74 m from the x-axis. The rear RCBs have a distance of 2.94 and 3.44 m. The worst-case (full propellant tanks) moment of inertia was computed using the subsystem masses and the parallel axis theorem. A value of 394423.7 kg-m^2 was found. To meet the requirements, each of the four thrusters has to deliver 246 N of thrust.

Rotation about the y-axis: The front RCBs have a worst-case distance of 4.82 m from the y-axis. The rear RCBs have a distance of 4.96 m. With a worst-case (full propellant tanks) moment of inertia of 36990.1 kg-m^2 each of the four thrusters has to deliver 16 N of thrust.

Rotation about the z-axis: The front RCBs have a worst-case distance of 3.24 and 3.74 m from the z-axis. The rear RCBs have a distance of 4.96 m. With a worst-case (full propellant tanks) moment of inertia of 250412.5 kg-m^2 each of the four thrusters has to deliver 123 N of thrust.

6.6.2 Actuator Selection

As the same thrusters of the RCBs have to fire for different rotations, the highest required thrust is chosen as design parameter. The highest thrust is 246 N. Reference ADCS thrusters are 200 N thrusters from EADS Astrium, used on the ATV [233]. They can run for a total of 12.9 hours with 160000 pulses. These thrusters have to be adapted for methane/oxygen propellant and their thrust has to be increased. Possibly, a new thruster system has to be developed. Next, the propellant mass for the ADCS has to be found. It depends on thrusters performance, thrust delivered per thruster, number of thrusters firing for each pulse, number of pulses and pulse duration. The I_{sp} of the thrusters is estimated to be 350 seconds, based on feasible values for specific impulse of methane/oxygen combustion (see Section **??**). Thrust delivered per thruster is shown above. For each pulse, four thrusters fire. The number of pulses is split between docking (estimated 65 pulses per axis) and other operations (estimated 20 pulses per axis). These values are multiplied by two, as each pulse has an accelerating and a decelerating burn. A margin factor of two is applied to account for disturbances and momentum-dumping, both of which are not treated here. This gives a total number of pulses of 340 per mission. Lifetime of the thrusters is thus approximately 470 missions (or over 20 years, accounting for two LSAMs flying 44 missions per year). The nominal pulse time is 20 s. However, minor attitude changes will require lower pulse durations. A factor of 0.6 is applied, giving average pulse times of 12 s. These values yield a total ADCS propellant mass of 1756.4 kg. Note that few pulses are required during non-docking operations, at the gimbal-mounted engines provide a main part of the attitude control during launch, de-orbiting, and landing.

Mass per thruster is estimated to be 5.9 kg. With contingency thrusters, the front RCBs have a mass of 59.0 kg, the rear RCBs have a mass of 47.2 kg. These values are based on other thrusters from EADS Astrium [234], scaled up to 200 N. Two dual-gimbal CMGs are installed on the LSAM, of which one is active and one is for contingency. Each CMG can generate 500 Nm about any axis, with a required power of 150 W and an estimated mass of 40 kg [60].

6.6.3 Sensor Selection

For redundancy, three IMUs are used. Each one weighs about 10 kg. The active high-precision IMU consumes about 50 W of power [60]. Two sun sensors provide coarse attitude determination. Each weighs about 1 kg. The active sensor consumes 2 W [60]. Fixed-head Moon horizon sensors are used for medium to coarse attitude determination. Two are placed on the LSAM. Each weighs about 2 kg. The active sensor requires 6 W. High-precision attitude determination is done via star trackers. Three are active at the same time, with two more as redundancy. Each star tracker weighs 3 kg and requires 10 W when active.

6.6.4 Costs of ADCS Parts

Considering atittude sensors costs, the required accuracy and precision has a major influence. IMUs cost about 10.1 k€ each [193]. Costs of sun sensors are set at 3 k€ per sensor [235]. Horizon sensors and star trackers each cost 140 k€ and 389.2 k€, respectively [235][191]. Total costs for sensors add up to 370.9 k€. Very little information can be found on costs of actuators. CMG costs are set at 150 k€, based on SMAD [60]. Costs of thrusters and feed system parts are set at 260 k€, taking into account differences in thrust from reference thrusters and the required systems for the LSAM [236]. Propellant costs can be neglected, as ISRU at the lunar base produces the required propellants. Note that these cost values are approximations. Also, no development costs were accounted for. Total costs for actuators add up to 9.66 M€. Note that costs of the LSAM ADCS are higher than for the CTTV ADCS. This is due to more redundancy and additional sensors, which are required by the ascent/descent mission part. Also, the LSAM is the active vehicle during docking.

6.6.5 Overview of the ADCS

The values presented in this section are summarised in Tabs. 6.15, 6.17, and 6.16. In total, the ADCS sensors have a mass of 51 kg and consume 88 W. The actuators add up to 292.4 kg. The active CMG requires 150 W of power. ADCS propellant mass is 1756.4 kg. The ADCS system costs about 10 M€.

Table 6.15: Overview of LSAM ADCS elements.

Element	No.	Tot. Mass [kg]	Power active [W]	Cost/part [k€]
IMU	3	30	50	10.1
Sun Sensor	2	2	2	3.0
Horizon Sensor	2	4	6	70.0
Star Tracker	5	15	30	38.9

Table 6.16: Overview of LSAM ADCS.

Element	Tot. mass [kg]	Tot. power [W]	Tot. Cost [k€]
ADCS sensors	51	88	370.8
ADCS actuators	292.4	150	9660.0
ADCS propellant	1756.4	-	-

Table 6.17: Overview of LSAM attitude control elements.

Element	Active	Contingency	Performance per Element	Mass per Element [kg]	Tot. Mass [kg]	Cost per Element [k€]
CMG	1	1	500 Nm torque at 150 W power	40	80	150
Front RCB	2	-	5 thrusters + 5 contingency	59.0	118.0	2600
Rear RCB	2	-	4 thrusters + 4 contingency	47.2	94.4	2080
RCB thrusters	18	18	250 N; installed in RCBs (see above)	5.9	212.4	260
Propellant	-	-	-	-	1756.4	-

6.7 Guidance and Navigation

This section will elaborate on the choices made with respect to guidance and navigation systems for the LSAM. Tab. 5.20 from Sec. 5.7 summarises the chosen configuration, with the exception of the operating mode of the R&DS, which operates under supervised autonomy, and the additional landing guidance systems, of which nothing in particular is known yet.

6.7.1 Accuracy Requirements

As the LSAM will be involved in launch and landing procedures, this induces certain requirements for the accuracy in order to complete these manoeuvring tasks, which can be found in Tab. 6.18. Furthermore, both rendezvous and docking procedures should be accounted for. Based on these requirements navigation information, orbit maintenance and orbit control will be treated later on.

Table 6.18: Accuracy overview of LSAM's GNS.

Phase	Accuracy	Unit
Launch	100	m
Rendezvous	0.5	m
Dock	<0.1	m
Landing	50	m

The launch accuracy was set at 100 m. The LSAM will be launched to an altitude of 100 km, where it should get within the mentioned range of the CTTV. During the rendezvous, the LSAM approaches the CTTV until they are close enough to initiate the docking procedure. The docking will be done by the use of a direct docking system as described earlier in Sec. 4.9.2. The NDS would be used for this, having a reach of 0.46 m. The docking accuracy itself should be in the order of centimetres, as recently the Autonomous Transfer Vehicle (ATV) called *"Albert Einstein"* showed possible [204]. For the landing phase, a circular landing clearance zone of roughly 10000 m^2 is required, which would result in a landing accuracy need of 57 m. 50 m will be taken as a final value as an added contingency on the area.

6.7.2 Guidance

The orbital control and maintenance requires data capability to arrive at rendezvous and perform the docking and landing procedures. Choosing for autonomous maintenance and control reduces life-cycle costs and risks for many missions. Nevertheless, since the LSAM exhibits a rather high thrust (around 500 kN) while performing launch and rendezvous, a fully autonomous system might endanger the CTTV without operator oversight. Considering the importance of the payloads, a more traditional approach (ground based) is desirable. It was therefore established that a supervised autonomy would suit the operational needs better. In this configuration, the maintenance manoeuvres are computed on board, while ground verification would be required to proceed with the operations. As the mission progresses throughout the years, no crew would be present any longer, causing all operations to be optimised and fully automated. Furthermore, only orbit control will be considered for guidance of the LSAM. Since the LSAM orbit maintenance (if ever needed in the worse case, as it is initially not meant to orbit long on its own), could be taken care of by the ADCS systems, or even by the CTTV while docked. The R&DS currently available on the ATV will be used (consisting of VGMs and TGMs), as explained previously for the CTTV in Sec. 5.7.

6.7.3 Navigation

The navigation (or alternatively called the orbit determination) is primarily needed to assure that the rendezvous ultimately will take place, by providing real-time orbit determination. The latter -together with possible surface recognition (imaging) hardware, and additional laser range-finders- is required to allow for more precise landing manoeuvres. ESA has been developing Lunar Lander's guidance, navigation and control for several years [237]. While studies are undergoing exploring the solutions, no specific information was found yet to include them in the budgets [238][239]. The navigation data would be needed on board, since it could provide for easier data distribution with other systems through links if needed (mission control, Space Dock, CTTV or lunar base).

An ANS will be used as navigation method. It is a fully autonomous system, and the CTTV already included it into its inventory which would allow for less diversity throughout the mission elements. Furthermore, this system has a high reliability and a low cost. Apart from being autonomous, it can use data from additional star sensors, gyros and accelerometers seen in ADCS systems. This system is capable of determining the orbit, attitude, ground look point, sun direction, all which are basically sufficient for lunar launches, rendezvous and landings. It does however require an additional spacecraft clock and computer (accounted for in the C&DH section). Its accuracy is stated to be 100 m, which would satisfy the launch accuracy. The landing accuracy does comply on the other hand, but might very well be attained by either combining the ANS with complementary sensors, gyros or accelerometers or using the aforementioned additional visual ranging/surface mapping hardware, or even both. The costs and mass estimations for both the ANS and R&DS are the same as described in Tab. 5.20.

6.8 Power Supply

To come up with the mass budgets, specifications and types of electrical power subsystems the different modes of the LSAM first have to be analysed. After that the selection and sizing process of the power sources can take place. Then the electrical system needs to be completed through power management and transfer devices (Power Control Unit (PCU), wiring etc.). As a final step the volume, mass, power delivered and the costs can be established for the power subsystem, which will be used as input values in the design iteration process.

6.8.1 Power Subsystem Modes

As described in the section on functions of the LSAM, the LSAM spacecraft has different phases of operating and thus different use of its subsystems. The different modes, which are of interest to power subsystem related features are:

- **Orbiting mode:** The total average operating power is based on this mode, when the LSAM orbits in a 100 km altitude LLO orbit. Including redundancy, this leads to an average power consumption of 9100 W for this mode, which is scaled by the ratio between the Altair mass to the preliminary LSAM mass. The two payload canisters consume another 66 kW together, which brings the total average operating power requirement to 75.1 kW.
- **Docking with CTTV mode:** During this mode extra communications and C&DH needs to be available as close interaction with the CTTV is necessary. During this phase the robot arms and manoeuvring thrusters consume extra power compared to average as well.
- **Descent mode:** At the beginning of the lunar descent it is stated that the peak power consumption is reached for the LSAM comparable vehicle Altair [240]. ACDS, communication and propulsion subsystems need to operate full power to ensure that the ΔV manoeuvre will be initiated correctly. The peak power of the LSAM is estimated

to be 5 kW higher, which leads to a peak power requirement of 80.1 kW

- **Ascent mode:** The ascent from the Moon is a mode in which the propulsion subsystem is the main factor. Communications and ACDS consume less power than the descent phase.
- **Lunar surface mode:** The passive mode where the vehicle is stored at the lunar surface. Only the subsystem and optionally the payload require power for radiation protection and storage of the canisters.

6.8.2 Eclipses

For mission times exceeding one month, it can be derived that photovoltaic, radioisotope thermoelectric and solar and nuclear dynamic systems might be used as an electrical power system. One of the most often used systems is the use of a solar power system. A drawback of the use of this system is that it will not be able to generate power when an eclipse occurs. An eclipse is the partial or complete obscuring, relative to a designated observer, of one celestial body by another. Whether the spacecraft is located in shadow or sunlight influences not only this power system, but also the thermal subsystem. Therefore the eclipse time needs to be calculated for the spacecraft orbiting around the Moon.

This spacecraft deals with multiple eclipses: A lunar eclipse, which occurs when the Moon passes directly behind the Earth into its shadow (umbra) and a spacecraft eclipse, which the eclipse that occurs when the LSAM is located in the shadow of the Moon. For the preliminary eclipse calculation the following assumptions are made:

- The orbit is considered to be 2D for simplicity purposes. Therefore the Moon's inclination angle and the solar exclamation angle are not considered.
- The celestial body's shadow is given the shape of a cylinder shadow, as the Sun is seen as a one-point object at infinite distance with solar rays approaching the celestial bodies in parallel. The worst case scenario is analysed, as the power system might generate some power when the spacecraft is in a prenumbra phase.
- The lunar eclipses and the spacecraft eclipses are treated separately from each other. The calculation for the eclipse time does not hold a combination of these two events. The spacecraft orbiting around the Moon might for example be located in the sunlight, when the Moon is already in the Earths shadow. These scenarios are discarded from the analysis as a rough estimation on the eclipse time will suffice for this detail of the design.

The eclipse time is calculation according to the equation:

$$T_{ecl} = \frac{2\lambda}{360} \cdot 2\pi\sqrt{\frac{a^3}{\mu}} \; where \; sin(\lambda) = \frac{R}{a} \tag{6.5}$$

In this equation T_{ecl} is the eclipse duration per orbit in seconds, 2λ is the angle between the centre of the shadowing body and the shadowed body at the start and the end of the lunar eclipse, a is the semi-major axis, R is the radius of the shadowing planet and μ is the gravitational parameter for the shadowing planet, which is Moon or Earth in this case.

The most important outcomes, which are shown in Tab. 6.19, are an eclipse time of 208 minutes for the lunar eclipse (large eclipse), which orbital period is 39531 minutes and an eclipse time of 46 minutes for the spacecraft eclipse (small eclipse) with an orbital period of 117.8 minutes. The maximum duration of the spacecraft to be out of the sunlight will hold one where a small eclipse occurs adjacent to a large eclipse, after which half the time of a small eclipse will follow. Therefore it is from now on dealt with two eclipse times, the longest eclipse time of 277 minutes and the shortest eclipse time of 46 minutes, which occurs multiple times in one orbit. It should again be mentioned that during these eclipses there will still be sunlight received by the spacecraft, due to the prenumbra phenomena and planetary reflection.

Table 6.19: Eclipse outcomes.

	Lunar eclipse (long)	**Spacecraft eclipse (short)**
T_{orb} [min]	39531	117.8
T_{ecl} [min]	208	46
T_{ecl}/T_{orb} [%]	0.53	39.4
Number of eclipses per day	0.036	12.2
Longest consecutive eclipse [min]	277	-

6.8.3 Selection and Sizing of the Power Sources

The selection of the primary power source will be based on the power required by the mission and the mission lifetime. Based on proven power densities, efficiencies and system components the power source will then be sized.

6.8.3.1 Primary Power Source

The primary power source for the LSAM will be selected through the criteria of required power, mission lifetime, system mass and costs. The first two elements are the main drivers. It was seen that the required average power to be delivered was 9100 W as the LSAM is scaled up with respect to the Altair. Including payload tanks the required average power is 75.1 kW. The primary power source is designed such that the primary power source should take care of the average loads, and the secondary power source should deal with eclipse periods and periods when peak power needs to be delivered. The mission lifetime is set at the depletion time of the Mare Tranquilitatis, which is seven years. This long mission life implies extra redundancy design, independent battery charging and larger capacity batteries.

In the previous Sec. 6.8.2 it was already mentioned which power systems are able to provide this required power and last reasonably for long mission times. As the power requirement is not high, there is no need for a nuclear power system. The power systems which qualify for this moderate power requirement are solar and fuel cells. The major drawback of solar cells is the irregular power supply due to eclipses, where for fuel cells the limited lifetime is a weak point. However, nowadays it is stated by manufacturers that some fuel cells have a lifetime of several years. The main advantage of fuel cells will be that their power density is high compared to other power systems, in other words: this is the most lightweight option. The choice for a fuel cell power system also allows to use in-situ resources from the Moon to operate (i.e. hydrogen). The operating temperature of the fuel cell subsystem also influences the thermal subsystem and the waste products (water) might be used for heating or cooling other components, which will be discussed in a later section. Other landers, such as the Apollo landers and the Altair also used fuel cells for descent and batteries for ascent.

As there are different types of fuel cells available, the PEM (Proton Exchange Membrane) fuel cells are selected as these fuel cells are regenerative and are able to already deliver powers of 10 kW for over 10000 hours [241]. The working temperature of the fuel cells is in the range of 80-120 degrees Celsius, this parameter may be of influence for the thermal subsystem design. The waste water from the fuel cell reaction can be used for heating or cooling components to increase efficiency. The schematic of the PEM fuel cell is shown in Fig. 6.5. The principle is that any easily oxidised fuel (i.e. hydrogen) is fed to an anode, while oxygen is fed to the cathode. Charged protons from the anode can pass trough the membrane and through that electrons travel along a circuit outside the fuel cell, providing power. The waste products are the depleted fuels, depleted oxidant, water and heat.

Sizing of the Fuel Cell System

The fuel cell power subsystem is divided into three components: the fuel cell stack, cooling system and tanks. The BOP (Balance of Plant), which holds the cooling system and the tanks, mainly determines the size and mass of the total fuel cell power system. For an order of magnitude, a fuel cell for designed for delivering 100 kW has a total mass of 250 kg, where the stack, cooling and tanks weigh 100, 40 and 80 kg respectively. The other 30 kg of mass are used for connecting the components [241].

It has to be taken into account that about 20% of the power has to be transferred to the BOP itself, which causes a significant loss in the net power output of the system. Also a lifetime degradation of 0.9 can be used for the fuel cell over the entire mission (which includes catalyst, membrane, bipolar plates, environment/contaminant/operation-induced and transport layer degradation) [242]. The required End-of-Life (EOL) power and the corresponding Beginning-of-Life (BOL) power, taking into account the lifetime degradation, are given by:

$$P_{EOL,av} = \frac{75.1}{0.80} = 94\ kW \tag{6.6}$$

$$P_{BOL,av} = \frac{P_{EOL}}{L_d} \tag{6.7}$$

This will result in an required average BOL power of roughly 104 kW for the fuel cells. The peak power for EOL is assumed to be 5 kW higher than the average EOL power. For PEM fuel cell stacks a power density in the range of 250-350 W/kg is available. It is assumed that by 2040 a power density of 350 W/kg can be taken, as in the year 2000 the power density was just about 150 W/kg. The space shuttle used of fuel stacks with a power density of 275 W/kg [243]. With the power density of 350 W/kg and the BOL power of 104.3 kW this leads to a fuel stack mass of 300 kg. Given the mass ratio between the cooling system, storage tanks and the fuel stack, the overall mass of the fuel cell system is determined to be 660 kg. The storage tanks of 240 kg for the moment are assumed to contain enough resources to operate for at least one LSAM descent/ascent cycle. It is recommended that calculations will be done on the tank sizing for hydrogen and oxygen and the consumption of these fuels during LSAM operations.

Figure 6.5: Schematic of a PEM fuel cell.

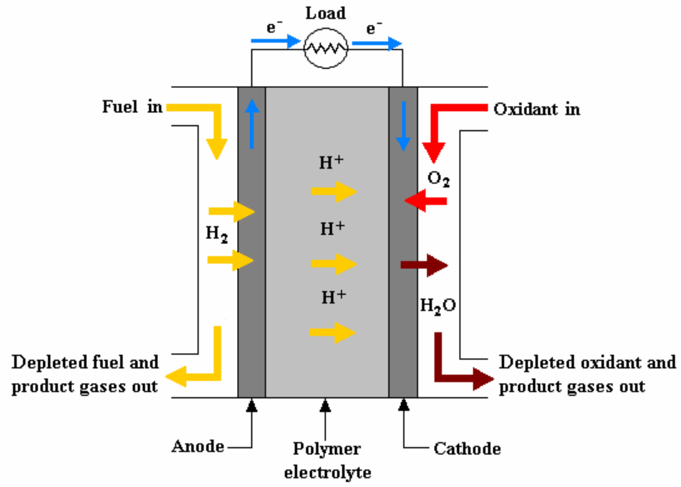


Figure 6.6: Fuel cell system masses.

Component	Value	Unit
PEM Fuel cell stack mass	300	kg
Cooling system mass	120	kg
Storage tank mass	240	kg
Total fuel cell system mass	660	kg

6.8.3.2 Secondary Power Source

As the primary power system is designed based on the average operating power requirement, a supplementary secondary power source is needed to provide power during peak power requirements. The peak power adds an additional 5 kW to the 104.3 kW average operating power (BOL). The peak power is then 109.3 kW (BOL). The peak power duration is expected not to last longer than half an hour, as the high power-consuming descent phase and ascent phase remain within this time limit. From SMAD p. 420 [60] the characteristics of secondary batteries are shown, which state that Litium-Ion and Sodium-Sulphur batteries are still under development. A conservative choice for Nickel-Hydrogen batteries with a specific energy density of 43-57 Wh/kg is made as this power source is already space-qualified. The Depth-of-Discharge (DOD) is determined with respect to the cycle life of the batteries. A higher depth of discharge means less battery mass, but shorter lifetime. 48 flights of the LSAM are needed per year, and 2 peak power battery cycles are assumed per LSAM flight. This leads to 1000 cycles per year, and 10000 cycles for the entire mission lifetime of ten years. From the DOD versus cycle life graph at p.421 in SMAD it is determined that a NiH2 battery DOD of 55% is needed to allow for 10000 cycles. A load transmission efficiency, n, of 0.9 is assumed for the spacecraft in LLO. These battery characteristics are listed in Tab. 6.20. The mass of the battery is obtained by combining the specific energy density with the battery capacity. The battery capacity C is given by:

$$C = \frac{PT}{(DOD)n} \tag{6.8}$$

P is the required power to be supplied by the batteries, T is the time that the batteries operate during one cycle, DOD is the Depth-of-Discharge and n is the load transmission efficiency. This results in a battery capacity of 5050 Wh and a corresponding battery mass of 87 kg.

Table 6.20: Secondary battery characteristics.

Battery type	Nickel-Hydrogen (single pressure vessel design)
Specific energy density	57 W·h/kg
Cycle life	7000-10000 cycles
DOD	55%
Load transmission efficiency n	0.9
Required electrical power	5 kW
Maximum battery cycle time	0.5 h
Battery capacity	5050 Wh
Battery mass	87 kg

6.8.4 Power Regulation and Control

Next to the primary and secondary power sources a wiring system and regulation and control system need to be present. The functions of the power regulation system is controlling the fuel cell system, regulating the bus voltage and charging the secondary battery. Electrical power needs to be controlled to prevent undesired spacecraft heating. The two main power control techniques are a Peak-Power Tracker (PPT) and a Direct-Energy-Transfer (DET) subsystem. The PPT is only active when peak powers occur, the DET constantly delivers power, even if there are no power loads. As the PPT uses a significant amount of the total power and is more suitable for missions under five years where power at the BOL is required than at EOL. Therefore the dissipated DET subsystem is used [60]. There are however some measures to avoid dissipation of power not used by the loads, such as shunt regulators, which will not be further discussed upon in this report. The control of the electrical-power subsystems and the bus voltage can be performed unregulated, quasi-regulated or fully regulated. For this system a quasi-regulated system is chosen, which only becomes unregulated during the discharge: for higher levels of regulation the system becomes more inefficient, but it should be ensured that the battery charging is regulated properly. The mission lifetime is usually limited by the batteries. For mission lifetimes over five years individual charging of the batteries has to be performed. The basic layout for a Direct Energy Transfer System with individual charging is shown in Fig. 6.7. The wiring and regulators mass and power loss will be discussed in the Sec. 6.8.5.

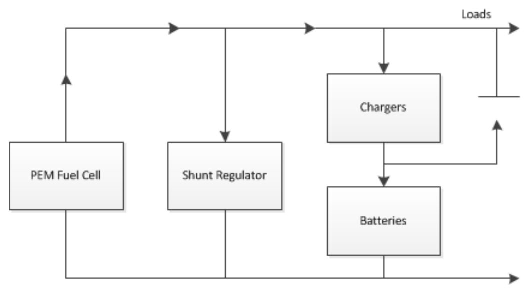


Figure 6.7: Schematic block diagram of the power management system.

6.8.5 Mass and Power Budget

From mass estimation tools in SMAD p.334 [60] the mass of the different power components can be determined. Also the power consumption of the regulators and wiring are shown in Tab. 6.21. The wiring is given as a percentage of the dry mass. The mass and power budgets are defined as a fraction of the controlled or operating power. Within these ranges of budgets, it is stated that the mass of the power system is 3920 kg and the power consumption of the power subsystem is 20.5 kW. The volume of the fuel cell system was estimated by comparison with existing PEM fuel cells. A 75 kW system has a mass of 140 kg and a volume of 0.05 m^3 [244]. An energy system density of 2980 kg/m^3. The total power system will then have a volume of 1.32 m^3.

Table 6.21: Mass and power budgets of the power subsystem components.

Component	**Weight [kg]**	**Power required [kW]**	**Fraction used for the calculation**
Fuel cell system	660	18.8	20% EOL power
Batteries	90	-	C/specific energy density
Power Control Unit	2080	-	0.02 P (controlled power)
Regulator/Converters	125	1.5	0.025 P for mass budget, 0.02 P for power budget (converted
Wiring	390 - 1560	0.1-0.25	0.01 - 0.04 M_{dry} for mass, 0.02 - 0.05 P for power
Total range	3330 - 4500	20.4-20.55	
Total	3920	20.5	

6.8.6 Results

The conclusive power subsystem features for the LSAM are listed in Tab. 6.22. It should be mentioned that the power supplied in this table has already accounted for the power demand of the power subsystem itself.

Table 6.22: Features from the LSAM power subsystem.

Primary power source	PEM fuel cells
Secondary power source	Nickel-Hydrogen (single pressure vessel design)
Electrical power control system	Quasi-regulated DET
Total weight power subsystem	3920 kg
Total power consumption power subsystem	20.5 kW
Average operating power EOL supplied	94 kW
Peak power EOL supplied	99 kW
Fuel cell system mass	660 kg
Battery mass	87 kg
Power subsystem volume	1.32 m^3
Power subsystem cost	2.05 M€

6.8.7 Costs

In an energy policy article on fuel cell economics [245] it was stated that the costs for a 5 kW fuel cell system are in the order of 923 €/kg and 538 €/kg for a 250 kW commercial fuel cell system. The costs per kW for the 104 kW system (including BOP and degradation), is determined to be 763 €/kg. For the total fuel cell system mass of 660 kg, a total fuel cell system cost of 0.5 M€ is obtained. The costs for the PCU, regulators and wiring are generally in the range of 308-770 €/kW [246], which will be fixed to 540 €/kW. This holds a cost for the PCU of about 56 k€. The yearly operational costs include some uncertainty, as these costs consist of the fuels used by the fuel cell. The availability of these materials heavily depends on the scale at which ISRU is available. For the operational costs 25% of the fixed costs are assumed, which match 150 k€ annually. For the LSAM lifetime of ten years this brings the costs of the total power subsystem to approximately 2.05 M€.

As seen in the conceptual power allocation of the LSAM, the payload canisters had a relatively large power demand of 80% of the total operating power. It was an option not to deliver any power to the canister compressors, which would decrease the power use of the LSAM drastically. In order to apply this decrease in power, 1.5% of the He-3 mass per day has to be sacrificed. To make a trade-off whether this is acceptable, a small value analysis can be performed. Given the fact that the time from lunar ascent until the docking with the CTTV might take 3.8 hours in the worst case and the He-3 mass per canister is 2.28 tons, this will hold a loss of five kg of He-3 per canister. With the He-3 price of 5 M€/kg, this leads to a loss of 2.200 B€ annually. This amount of money does not offset the cost of the power subsystem and therefore the high power demand of the canisters is accepted.

6.9 Telemetry, Tracking and Command

The detailed design of the communication subsystem of the LSAM is presented. Below the made assumptions are given that have been used while obtaining the numbers.

- Direct LSAM-lunar base communication when possible, otherwise via relay satellites.
- Lunar base communication system equipment is comparable to TDRSS equipment.
- Communication equipment for docking negligible in terms of mass and power.
- Relay Satellites are similar to NASA's TDRSS.
- A needed downlink of 800 kbps.
- A needed uplink of 60 kbps.
- Low Bit Error Rate (BER) of 10^{-6} (See SMAD [60]).
- Polarisation and propagation loss of -0.6 dB (See SMAD [60]).
- Transmitter line loss of -1 dB (See SMAD [60]).
- Pointing error is half the half power beamwidth (SMAD [60]).
- Antenna efficiency of 0.55 (SMAD [60]).
- SNR is assumed to be as stated in 6.23

Communication Architecture: The architecture for the communications of the LSAM does not deviate significantly from the architecture of the CTTV. As long as the LSAM is in the field of view of the Lunar Base Communication Station (LBCS), it will communicate directly with the lunar base. As soon as the LSAM leaves the field of view of the lunar base, it will communicate with the base via one of the relay satellites. Only for docking procedures the LSAM communicates with the CTTV.

6.9.1 Ground Station and Relay Satellites Selection

In the beginning of the mission there will be only one lunar base, therefore there will also be only one LBCS. The LBCS so far has not been developed. It is assumed that the LBCS will be of similar size as the relay satellites used in LL1 and LL2. This yields an antenna diameter of the LBCS of 4.5 m as mentioned in Sec. 5.9.3. For the communication with the relay satellites obviously the same satellites will be used as mentioned in Sec. 5.9.3.

6.9.2 Frequency, Data Rate and Memory Size

The TDRSS relay satellites of NASA, reference for the relay satllites, use, as addressed in Sec. 5.9.4, the Ku-band frequency. The corresponding frequencies can be found in Tab. 5.21. To avoid interference with the communications between the CTTV and the relay satellites, different frequencies have to be selected for the communications between the LSAM and these satellites. The selected frequencies, based on TDRSS [216], are 14.4 GHz for forward link and 12.6 GHz for return link. For the communication between the LBCS and the LSAM, it is assumed that the same specifications as for the relay satellites can be used. This is reasonable, as there is no atmospheric barrier between the Moon and the LSAM, meaning that the communication subsystem will not suffer from atmospheric attenuation. Again Ku-band is assumed and a frequency of 14.2 GHz for uplink and 12.5 GHz for downlink is selected. The diameter of the lunar base receiving antenna is assumed to be 4.5 m, similar to the antennas of TDRSS [216]. The data rates for LSAM communication differ from the CTTV. The data rate of commands from the base are estimated to be higher than for the CTTV, since the robotic arms need to be monitored and/or controlled from the base. 60 kbps is a reasonable data rate. Also, the data transmitter data rate is estimated to be higher than for the CTTV. A video stream is desirable during the payload handling from the LSAM to the CTTV and more data due to the robotic arms needs to be transmitted. This data rate is estimated to be 800 kbps. For docking procedures with the CTTV, it is assumed that the UHF-band will be applied. As explained in Sec. 5.9.4, it is assumed that the mass and power budget of this link are negligible. Like the CTTV, the LSAM has a continuous link with the lunar base, meaning no significant data memory is required.

6.9.3 Link Budget

With the input values discussed in the previous section, a link budget will be provided. The same equations apply as in Sec. 5.9.5. The same input variables are assumed for $(\frac{E_b}{N_0})_{req}$, L_a, L_l, L_{pr} and η. The distance S between the transmitter and receiver antenna for the CTTV was obtained by neglecting the orbiting motion of the relay satellites. For the distance S between the LSAM and the LBCS this can not be assumed, as the changes in space losses do have a significant impact on the design of the communication subsystem.

The distance S is calculated using Eq. 6.9. In this equation, R is the radius of the Moon which is 1737 km and h is the altitude of the LSAM, which is 100 km. The principle is shown in Fig. 6.8. It is thereby assumed that the LBCS can receive and send signals parallel with the horizon. This yields a maximum distance of 397.8 km between the LSAM and the LBCS. The worst possible distance between the LSAM and LL1 is LLO distance plus the diameter of the Moon and the distance from the Moon's surface to LL1. This is a total distance of about 70000 km. For this distance again the motion of the relay satellites is not taken into account due to its low impact on the space losses.

$$S = \sqrt{2 \cdot h \cdot R + h^2} \tag{6.9}$$

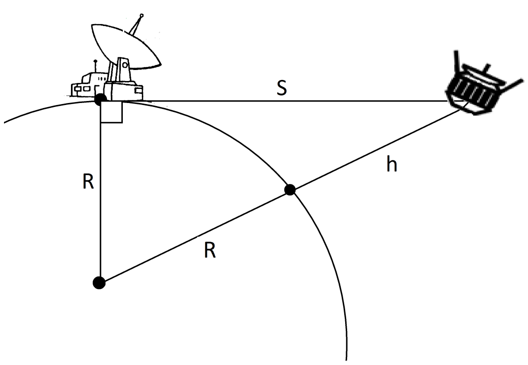


Figure 6.8: Maximum distance S from LBCS to LSAM.

6.9.4 Power and Mass Budget

For all assumptions and input values the results of the link budget calculations as adressed in the previous section can be found in Tab. 6.23. Again because of the low data rates, low gain antenna's will be used. Because of the very limited space losses for the communications between the LBCS and the LSAM, the required antenna size, power and mass are only 1 cm, 3 mW and 0.12 kg respectively, and thus very (negligible) small. Masses are estimated using SMAD [60]. For the communications with the relay satellites, low gain antennas are used to require a less strict pointing accuracy. The downside of this design choice is the higher required power. However, the total required power for the subsystem is still very small in comparison with the total power required for the LSAM. For redundancy, two antennas are placed for the link between LSAM and LBCS. For communications with the relay satellites, also two antennas are placed. This yields one redundant antenna, because the LSAM will have to communicate with at most one of the relay satellites in the same time.

To conclude, for the worst case if the LSAM communicates with the base via a relay satellite, the total power required is 50 W. The total mass of this LSAM subsystem is 12.62 kg. The cost of the LSAM communication subsystem carries a high uncertainty. The costs are estimated similar as explained in Sec. 5.9.6. This yields a cost of 1.2 M€ for communications with the base and 1.2 M€ for communications with the relay satellites, implying a total cost of 2.4 M€ as shown in Tab. 5.22.

Table 6.23: Link, mass and power budget for LSAM communication subsystem.

	Symbol	**Units**	**Downlink to LBCS**	**Intersatellite Link to LL1/LL2**
Frequency	f	GHz	14.2	14.4
Antenna Efficiency	η	-	0.55	0.55
Transmit Antenna Diameter	D_t	m	0.01	0.01
Transmitter Power	P_t	W	0.003	50
Transmitter Power	P_t	dBW	-25.2	17
Transmitter Gain	G_t	dBi	0.9	1
Transmit Antenna Beamwidth	θ	deg	147.9	145.8
Transmitter Line Loss	L_l	dB	-1	-1
Equivalent Isotropic Radiated Power	EIRP	dBW	-25.4	17
Free Space Path Length	S	km	397.8	62834
Space Loss	L_s	dB	-167.5	-211.6
Propagation and Polarisation Loss	L_a	dB	-0.6	-0.6
Receiver Antenna Pointing Loss	L_{pr}	dB	-3	-3
Receiver Antenna Diameter	D_r	m	4.5	4.5
Receiver Antenna Gain	G_r	dB	51.1	51.1
System Noise Temperature	T_s	dB-K	27.9	27.9
Data Rate	R	kbps	800	800
Signal to Noise Ratio	$\frac{E_b}{N_0}$	dB	42.7	41
Required Signal to Noise Ratio	$(\frac{E_b}{N_0})_{req}$	dB	13	13
Link Margin	-	dB	29.7	28
Total Power (incl Redundancy)	P	W	0.006	100
Total Mass (incl Redundancy)	m	kg	0.12	12.5
Total Cost (incl Redundancy)	-	M€	1.2	1.2

6.10 Command and Data Handling

The C&DH system will be estimated in term of size, mass and power. This will be done by considering the complexity of the processing operations to be carried out. Initially the functions will be established, followed by requirements and constraints.

Firstly, it has been identified that the C&DH should mainly be needed for data processing, command processing but also partly for data storage as the LSAM will operate autonomously. This means that an on-board computer will be needed. Furthermore telemetry processing is also of importance, mainly due to the need of feedback of on-board control systems. Also, the LSAM will use additional sensors in order to succeed in its landing, which will require more processing. However, the complexity is not as great as i.e. scientific mission telemetry requirements. Additional functions which the C&DH should provide for are: time awareness (needed for launch / rendezvous / docking / landing / guidance and navigation), computer watchdog (needed to keep track of the condition of the on-board central processor, even more so since the system should operate under supervised autonomy), and some spare channels (in case automation would induce an unforeseen growth of requirements).

Several constraints are identified. For the spacecraft bus, integrated C&DH systems would be required, considering the autonomous nature of the LSAM. At a cost of higher software and programming requirements, this would allow for a reduced hardware requirement overall (as command, telemetry, flight processing and attitude control is integrated in one system). Furthermore, this would allow for easier supervision through data links from Earth or Moon. Another requirement is that reliability should be high, due to the importance of the payload. Redundant systems are costly though, and parts quality increases with cost as well. Nevertheless, since only nine LSAMs would operate during 30 years (no critical mishaps included), the investment is well made. In terms of radiation, it is important to account for the degradation and limitation of part types. As pointed out in SMAD: "Environment severity may double system development time, and increase parts cost by a factor 10" [60]. It was established that the radiation near the Moon surface would be 0.82 mSv. Radiation may cause the equipment to deteriorate significantly.

Using Tab. 1.28 from SMAD, the complexity was assessed for the command and telemetry functions. Furthermore, the system was combined with additional functions, therefore the characteristics of a combined system were used. The overall complexity was found to be between typical and complex. It was decided that for contingency reasons, the combined system behaves as a complex system. Tab. 6.24 shows the outcome. It must be pointed out that these are still first-order estimations. These values may increase if complexity grows even more.

Table 6.24: Estimated size, mass and power usage for a combined system [60].

Component	**Size [cm^3]**	**Mass [kg]**	**Power usage [W]**
Combined system	14 000	12	30

The C&DH diagram obtained in Sec. 5.10 is rather similar to the one for the LSAM. Therefore, due to the similarity, no diagram will be given here. The only difference is that the LSAM will have additional landing telemetry devices and thus a slightly higher processing requirement.

6.11 Docking Procedures and Payload Handling

LSAM and CTTV have to dock before transferring payload. Once docked, the two systems act as one inertial body; the relative distances between the payload bays of the two vehicles remains the same, no matter the external disturbance force. This simplifies the payload exchange. The main challenge of designing the docking process is to allow good access to the LSAM's and the CTTV's payload canisters for the LSAM's payload exchange robotic arms (PERAs). Two PERAs are required to facilitate the payload exchange, as all four canister spots involved in the payload exchange are occupied. Two full He-3 canisters have to be exchanged with two empty canisters. Additional supplies may be exchanged as well.

The PERAs are located on a support structure between the two payload bays on top of the LSAM. The required docking port of the CTTV is located on the bus, just below the canister mounting. The docking port of the LSAM is placed on one of the LSAM bus sides parallel to the canisters. Thus, the PERAs can easily grab any of the two nearest canisters (see Fig. 6.9). The same docking system as for the docking CTTV-Space Dock is used (NASA Docking System, NDS). Only a purely structural connection is established between the vehicles. No other connections (power, fuel lines, etc.) are included in the link. The docking system has a mass of about 340 kg [167] (see also Sec. 5.11).

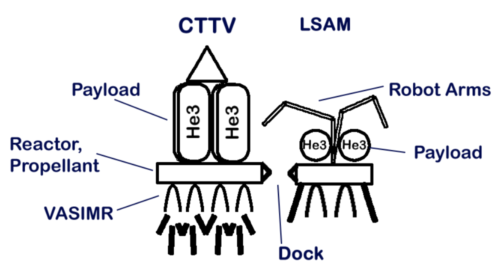


Figure 6.9: Sketch of docking between CTTV and LSAM.

An overview of the docking procedures and payload handling operations can be found in Tab. 6.25. The total mission of the LSAM may take no longer than 3.5 hours. This is equivalent to two orbits, allowing a timely return of the LSAM to the lunar base, without additional time spent in orbit to reach the correct de-orbiting position. The entire procedure has to be done twice per CTTV, to exchange all four payload canisters. Thus, two LSAMs are required. The second LSAM will only launch after the first one has landed. This reduces the need for extensive on-orbit proximity and collision avoidance operations between three vehicles; only two vehicles have to be coordinated on-orbit.

When the LSAM is within 250 m distance from the CTTV, the automated docking procedure begins. Using European ATV-derived technology the LSAM approaches the side of the CTTV. VDMs and TGMs on board the LSAM constantly measure the distance and orientation. When the docking connection is established and secured, the robotic arms are activated. Note that each canister has an adaptor to connect to the arms. Additional supplies may be taken from the CTTV after exchange of the canisters. This will not occur for every payload exchange. The supplies, in a container, will be stored in a structure between the two robotic arm supports on the LSAM.

The robotic arms consist of two limbs and one 'hand', which is highly mobile and contains the adaptor mechanism. Designing each limb such that it can bridge the horizontal and vertical distance and adding a margin of 0.4 m for each limb gives a total limb length of 10.15 m. The base of the arm can rotate in a half sphere of 180 degrees. The connection of the first limb to the base (Connection I) and the connection of limbs one and two (Connection II) can each rotate only around one axis. Robotic arms of similar capabilities have already been developed (European Robotic Arm ERA, Canadarm, Canadarm2). In particular, ERA has similar specifications as required for the payload exchange. With a total length of 11.3 m, a mass of 630 kg, average power of 475 W (800 W peak), and maximum payload handling capacity of 8 tons ERA is used as reference [247]. Linearly scaling up the ERA parameters to the required payload handling of 12 tons gives a robotic arm mass of 950 kg and average power of 320 W (peak 540 W). Costs of ERA are not yet published. For order of magnitude estimation, costs of Canadarm were about $108 M (assuming FY80), and mass was 431 kg (for 14.5 kg payload capacity) [248]. The costs of each robotic arm for the LSAM is set at 240 M€ (FY12). The major part of these costs is research and development costs. Production costs are estimated at 2 M€. See Tab. 6.26 for the PERA specifications.

Table 6.25: Operations performed during docking and payload handling, in sequence.

Sequence number	Operation
1	Reach proximity of CTTV.
2	Initiate automated docking procedure at 250 m distance.
3	Dock with CTTV.
4	Perform systems check, unfold Arm I and II.
5	Attach Arm I to CTTV canister (empty).
6	Lift Arm I CTTV canister out of payload bay.
7	Attach Arm II to LSAM canister (full).
8	Lift Arm II LSAM canister into empty CTTV payload bay, secure canister.
9	Lift Arm I CTTV canister into empty LSAM payload bay, secure canister.
10	Repeat steps 5 to 9 for second canister pair.
11	Retract/Fold Arm I and II, check secureness of payload.
12	Undock from CTTV and leave proximity.
13	Correct LSAM attitude and descend to lunar base.

To estimate the required torque for the robotic arms, the manoeuvre of lifting one canister out of its CTTV payload bay is considered. It is assumed that only Connection I (at the arm base) acts during this operation. After attachment to the canister the arm is moved to a position parallel with respect to the LSAM vertical body axis (that is, by an angle of 90°). With a distance from arm base to canister connection point of 6.8 m, the described arc has a length of 10.7 m. If the total exchange of payload should take no more than one hour, the described arm manoeuvre takes no more than 300 seconds (taking into account the other arm operations). Half of the time the arm accelerates, half of the time it comes to halt. This gives a required acceleration/deceleration of $3.5 \cdot 10^{-5}$ rad/s^2 (7.2 arcsec/s^2). The inertia of the accelerated mass is approximately the inertia of the moved canister. Taking the moment of inertia of a 9.7 tons empty canister around the centre of rotation (distance 6.8 m) is about 448528 kg·m^2. With the previously calculated acceleration, this gives a required torque of 16 Nm. This torque can be applied by available electromotors.

Table 6.26: Parameters of the robotic arms.

Length [m]	Mass [kg]	Handling Mass [tons]	Costs [M€] (FY12)
10.15	950	12	240

6.12 Summary

The objective of the LSAM is to deliver two full He-3 canisters to LLO, dock with the CTTV, and exchange the payload. Subsequently, the LSAM de-orbits and lands at the lunar base, carrying two empty canisters received from the CTTV. For an overview of the mass and budget, be referred to Sec. 7.2. The cost of the LSAM can be found in Sec. 7.4.

The different aspects of the LSAM mission have been presented, with the subsystems designed in detail. Overall, the LSAM has a dry mass of 37.12 tons, of which 21.54 tons are payload and 1.76 tons are ADCS propellant. Propellant mass for ascent and descent adds up to 71.58 tons, including 1.9 tons contingency. The total launch mass from the lunar base is 108.70 tons. Total power consumption of the subsystems, excluding power required by the power system itself, is 75.1 kW, of which 66 kW are for support of the payload canisters. From lift-off, the LSAM follows a trajectory that initially brings it into a 20 km x 100 km orbit, and then to the LLO CTTV orbit of 100 km x 100 km. Ascent requires a ΔV of 1928 m/s and takes about 8.4 minutes, descent requires 1879.6 m/s and takes 5.1 minutes. The LSAM is equipped with three RD-185-derived engines, with a maximum thrust of 180 kN each. Specific impulse is 380 seconds. The engines are throttled to 60%. In case of an engine failure, the remaining engines are throttled up. For propellant, methane and oxygen from ISRU is used. The structural subsystem of the LSAM supports the payload, two robotic arms for payload exchange and other subsystems. Four legs withstand the impact of landing. The main bus structure distributes the thrust of the engines. Aluminium was chosen for most structural elements. Thrust-induced vibrations will not harm the structure.

Thermal control keeps the LSAM temperature within a range from 263 to 298 K. Heating elements and MLI are applied. The system also uses 353 K waste water from the fuel cells. The LSAMs ADCS controls the spacecraft's motion during docking and most other operations. The gimbal-mounted engines provide attitude control during ascent and descent. The LSAM uses a total of 18 thrusters for attitude control. Additional CMGs provide control in close proximity of the CTTV, when some thrusters cannot be fired. A variety of IMUs and sun-, horizon-, and star sensors provide attitude determination. The GNC subsystems ensures that the LSAM reaches orbit with an accuracy of 100 m and lands with an accuracy of 50 m. For docking with the CTTV, ATV-derived technology is used. Landing requires technologies currently under development for ESA's Lunar Lander. The power subsystem can provide 104.3 kW at BOL. At EOL, 93.9 kW are supplied. Power will be provided by fuel cells, which are advantageous over other power sources for short mission durations. Reliable, space-flight-proven batteries are used as a secondary power source to provide peak power during ascent, descent, and docking and payload exchange. The LSAM uses X-band to communicate with the lunar base and the relay satellites in LL1 or LL2. UHF is used for proximity communication with the CTTV. Peak data rates, occuring during docking and payload handling, are 60 kbps for uplink and 800 kbps for downlink. Two directional X-band antennas and two UHF receivers/transmitters are used. The command and data handling system allows the LSAM to operate largely autonomous. Sensor data is used as input for docking, payload handling, and ascent and descent trajectories. Docking with the CTTV establishes a purely structural connection between the two vehicles. The LSAM docking port is located on the side of the main bus, allowing the robotic arms easy access to both the LSAM's payload canisters and the CTTV's payload. Two robotic arms facilitate the payload exchange.

6.12.1 Scalability

The LSAM detailed design is considering a mission that supplies 200 tons of He-3 per year to provide 10% of the global energy demand in 2040. As each LSAM carries two He-3 canisters, two LSAMs will be launched for every CTTV arriving in LLO. A third LSAM will remain on the lunar surface for contingency. Per year, 44 LSAM launches are required to supply 200 tons of He-3, with just above 16 days available between launches to service and maintain the two active vehicles.

The LSAM element of the overall mission can be easily scaled up or down. A higher energy market share, such as 50%, would require a total 220 LSAM launches per year. More LSAMs would have to be maintained on the Moon, as the launch frequency would be higher than the turnaround time of each LSAM. However, a higher amount of LSAMs reduces the costs of production. Available propellant is not a restriction to scalability, as increasing the He-3 mining rate also increases the available methane. Oxygen, however, may be too scarce to supply a large fleet of LSAMs with enough oxidiser. Availability of oxygen on the Moon (i.e. as water ice near the poles) requires further research. Scaling the mission down to 1% market share (20 tons of He-3) corresponds to 2.2 LSAM payloads per year. Thus, either the payload canisters need to be re-designed for higher He-3 capacity, or three LSAM launches have to be conducted annually. One LSAM (plus one for contingency) would be sufficient to deliver the He-3. For a mission scaled down to 0.1% market share (2 tons of He-3), the LSAM needs to be redesigned. The configuration presented above would be over-designed for such a small amount of He-3 per year.

7. Mission Evaluation

This chapter will evaluate the designs which are discussed in Chs. 4 - 6. First, the requirements are checked with the use of a compliance matrix. Secondly, the budget breakdown of the end-to-end system will be given. The technical risk assessment will follow thereafter. Concerning the cost, the Life Cycle Cost (LCC) and the Return on Investments (RoI) will be discussed in Sections 7.4 and 7.5. After this, the RAMS evaluation and the sensitivity will be elaborated on. At the end the operations and logistics of the missions is discussed.

7.1 Compliance Matrix

The compliance matrix is a table of all requirements set for the mission, along with confirmation, or, if that is the case, along with an explanation of why the requirements were not met. Requirements of all major systems and subsystems have been stated Secs 4.1 5.1 and **??**. It suffices to confirm that all requirements have been met, including the top-level requirements, except for one, which will be covered below. However, some conditions are involved with fulfilling the stated requirements. These conditions will be elaborated upon in Ch. 8 and recommendations will be given in Ch. 10.

One of the top-level requirements was not addressed in detail. The requirement states *"The system must be scalable to adapt to 50% of the world energy demand"*. It was agreed to put less emphasis on this requirement, but instead focus on evaluating down-scaling the mission. Above requirements will be evaluated in the following.

While in itself, scaling up is possible, the proposed mission becomes incredibly complex and nearly impossible to manage, rendering the above requirement a killer requirement. To illustrate this complexity: every other day docking would have to occur in LEO, and every day in LLO. This is one example of aspects of the mission that would induce significant complexity.

7.2 Budget Breakdown

Throughout the mission description presented in the previous chapters, the different budgets in terms of mass, power, cost, and propellant have been set up. To give an overview of the numbers associated with the mission profile, these budgets have been summarised in this section. Below a table is shown with the mass, power, and propellant budgets, for the cost breakdown and -budget the reader is referred to Sec. 7.4. For references and reasoning on the presented numbers the corresponding sections and chapters can be consulted.

It should be noted that the budget breakdown in Tab. 7.1 is per individual 'unit' listed, while for the yearly operations a multitude may be operational. To establish a total budget, this multitude has been accounted for. For the mass budget the total amount during a mission duration of 30 years was used, i.e. Skylon has a lifetime of 1.5 years, so with 1 operational Skylon at any time, a total of 20 vehicles is required. This is to account for the different lifetimes of the components of the mission.

The total power budget comes from the required power on any instant during the operational phase of the mission. This means for example that the required power from one Skylon is used, while the power of 22 CTTVs is accumulated, since this is the amount that is in operation each year. With the CTTV lifetime estimated at 30 years, the mass of 22 CTTVs is used for the mass budget over the mission lifetime. The total propellant budget is not included, since different propellants are used for different applications. The budget breakdown is shown in Tab. 7.1.

Table 7.1: Budget breakdown of the entire mission. Values are stated per unit.

Segment	**Mass [kg]**	**Power [W]**	**Propellant [kg]**	**Amount / yr**	**Amount / 30 yrs**	**Lifetime [yrs]**
Skylon	275 000	44 977 000	32 116 000	1	23	1.35
Space Dock	107 700	294 500	1725	1	1	30
CTTV	11 026	1 001 916	27 432	22	22	30
... Propulsion	2 160	-	-	-	-	-
... Propellant tanks	2 586	-	-	-	-	-
... Structures	1845	-	-	-	-	-
... Thermal control	9	1370	-	-	-	-
... ADCS	131	356	1 572	-	-	-
... GNS	40	150	-	-	-	-
... Power supply	4 225	-	-	-	-	-
... TTC	19	15	-	-	-	-
... C&DH	11	25	-	-	-	-
LSAM	13 824	9069	73 332	3	9	10
... Propulsion	1 245	-	71 576	-	-	-
... Propellant tanks	409	-	-	-	-	-
... Structures	7 059	-	-	-	-	-

... Thermal control	670	7000	-	-	-	-
... ADCS	343	238	1 756	-	-	-
... GNS	40	150	-	-	-	-
... Power supply	4 033	1 551	-	-	-	-
... TTC	13	100	-	-	-	-
... C&DH	12	30	-	-	-	-
Payload	10 770	33 000	-	88	264	10
Lunar base	75 000	3 200 000	-	1	6	5
Lunar operations	217 090 000	38 906 800 000	-	-	-	30
Manned missions	1 450 000	4000	1 383 000	3	90	
Total	**357 683 000**	**38 980 000 000**	-	-	-	30

Though the values have been estimated with varying reliability (due to the available info on the data), above table enables the comparison of the orders of magnitude associated with the different budgets. What becomes clear from the table is the major impact of the lunar operations and the manned missions on the budgets. The final mass budget is mainly driven by these two segments, though it must be noted that they have a greatly differing impact. The mass of the lunar operations has to be delivered to the lunar surface, while the mass of the manned mission mainly consists of the mass that is required for the high number of launches. For the power budget, the only driver is the lunar operations segment, which adds several orders of magnitude to the final power budget.

7.3 Technical Risk Assessment

As part of the mission evaluation the technical risks of the proposed mission are covered. While any given space mission is subject to a large number of risks, this section covers the most important and general ones which are applicable to the mission layout as proposed. First, a list of the most important risks will be established, after which a risk map will be shown representing the severity of each risk.

7.3.1 Technical Risk List

This section will show a list of possible risks to which the mission might be exposed, summarised in Tab. 7.2. Along with a description of the considered risk, its consequence(s) are listed, as well as its possible impact on the mission, the severity of the impact, and what action is proposed with respect to this risk. The possible actions that can be undertaken are divided into four categories: Research, Mitigation, Accept, and Watch (in the case risk is not deemed problematic enough but is expected to increase to dangerous proportions).

Table 7.2: Technical risk list.

#	Risk	Consequence	Prob.	Imp.	Action
1	Technologies used not available or proven by 2040	Mission delay	3	5	Research
2	Re-usability of LEO access vehicle lower than expected	Mission delay or more LEO access vehicles needed	3	3	Watch
3	Earth launch failure (rate: 0-5%)	Loss of payload / launcher / human. Mission delay and additional costs	1	4	Mitigate
4	Availability of launch sites is lower than expected	Mission delay	1	3	Mitigate
5	Atmospheric uncertainties during re-entry	Corrections needed or deviation from landing site	1	2	Accept
6	Docking and undocking failure at LEO/LLO station	Loss of payload / module	3	4	Mitigate
7	Errors or failure in stationkeeping of LEO/LLO station	Orbital changes, possibility of crashing	1	4	Mitigate
8	Space debris collision	Partly damaged or total malfunction of module	2	3	Accept
9	Unexpected solar flux influence on spiral transfer	Additional corrections needed	1	1	Research
10	Unexpected thrust variations of LEO access vehicle, transfer vehicle or lunar descent/ascent vehicle	Deviation from path	1	2	Research
11	Thruster failure	Loss of vehicle and payload	2	3	Mitigate

12	More orbital corrections needed than expected during transfer	Not sufficient fuel reserves	1	3	Research
13	Docking and undocking takes longer than expected	Mission delay	1	3	Mitigate
14	CTTV software malfunction	Payload loss, salvation mission required	2	4	Mitigate
15	Equipment damage due to landing	Unable to perform operations	2	4	Mitigate
16	Unable do deploy lunar base	Mission failure	3	5	Mitigate
17	Lunar miners malfunction or crash	Loss of payload and repair mission needed	3	3	Mitigate
18	Accessibility of the mining region is limited	Lower mining efficiency	4	3	Research
19	Robotic arm malfunction	Human interaction required	2	4	Mitigate
20	Payload re-liquefier malfunction	Loss of payload	1	4	Mitigate
21	Unable to obtain (sufficient) in-situ resources at lunar base	Additional re-supply missions needed	3	4	Research
22	Damage to equipment due to lunar dust	Mission delay, repair mission needed	3	2	Mitigate
23	Unexpected repairs are needed due to automated systems	Additional repair mission needed	3	3	Mitigate
24	Radiation effects on materials and humans	Material degradation and death/sickness of humans	3	3	Research
25	Temperature differences influence materials	Material performance fluctuations	3	2	Research
26	Computer failures	System failure	2	4	Mitigate
27	Geological regolith predictions are wrong/inaccurate	Mining rate increased or mission irrelevant	2	4	Research

7.3.2 Risk Map

In this section, the results of Tab. 7.2 are graphically represented. The risk map portrays the severity of the risk versus the likelihood of the risk. The ultimate goal is to reduce all risks to the lower left corner of the map, if possible. However, practice often shows that most risks, while they can be reduced, can never be completely reduced to an improbable, low-impact occurrence. Fig. 7.3 shows the risk map. The numerical indices refer to the risk indices in Tab. 7.2.

Two risks in Fig. 7.3 are critical: Risk 1 and Risk 16. Risk 1 refers to the technology readiness level of fusion technologies. Indeed, if there is no fusion technology available, it would make no sense to set up an entire space mission and infrastructure to bring back He-3 in large quantities, without any purpose for the He-3. Therefore, either governments or corporations must decide to invest intensively in fusion research and development, in the hope the technology will be proven feasible, or the entire mission must be abandoned.

Risk 16 pertains to the establishment of the lunar base. If anything goes wrong in this phase of the mission, the entire lunar segment may be compromised, resulting in imminent danger for any present humans and inducing additional costs for deploying a second lunar base to the Moon. The lunar base is critical to lunar operations, human protection, and survival. In order to mitigate this risk, the base must be developed as redundant and as resistant to its environment as possible.

Table 7.3: Technical risk map.

Catastrophic			16		
Significant	3,7,20	14,15,19,26,27	6,21		
Moderate	4,12,13	8,11	2,17,23,24	18	1
Low	5	10	22,25		
Negligible	9				
	Improbable	Remote	Occasional	Probable	Frequent

7.4 Life Cycle Cost

The overall cost will be a driving factor in the feasibility of the mission. If the costs are too high or investments too risky, investors will not have confidence in the project and it will not be executed. To address the overall cost first a detailed cost will be provided of both detailed designs. Secondly all the segments are evaluated based on investment cost and annual operating cost. Using the lifetime of the specific segments, an annual investment cost is provided. Finally the overall annual cost is estimated. For the cost per segment the worst and best case are addressed. For the costs the following assumptions are used:

- All the costs are converted to fiscal year 2012 using the cost conversion in Sec. 3.4.
- Interest cost for loans is not taken into account.
- Disposal costs are neglected.
- Precursor costs are treated separately.
- Mission lifetime is assumed to be 30 years.
- Nuclear fusion plants are estimated separately and are not included in the ground segment.

7.4.1 Detailed Cost Estimation

Based on the costs found in Chs. 5 and 6 a detailed cost estimate could be made. In the cost estimate the R&D costs, production costs, and cost for launching the vehicles to LEO and LLO have to be included as well. Detailed cost for the CTTV are presented in Tab. 7.4. In this overview the subsystem costs follow from Ch. 5. The total manufacturing and testing cost follow from the cost estimation relation of 20557 €/kg dry weight [60]. The research and development cost follows from the development costs of the VASIMR-engine [249] and the development cost of a transfer vehicle [171]. The launch cost to LEO are assumed to be 1500 €/kg, based on a highly competitive space market [147]. With these values the cost of the theoretical first unit was found and from applying the learning curve method stated in Sec. 3.4, the average cost per CTTV was found. Based on the lifetime and the O&M costs, the total annual cost per CTTV was computed. For the LSAM the same method was applied, resulting in a detailed cost estimation shown in Tab. 7.5.

Table 7.4: Detailed cost of the CTTV.

Item	**Cost [M€]**
Propulsion	250
GNS	1.2
Thermal Control	0.0
ADCS	2.4
Power Supply	168.0
TT&C	2.9
Structural materials	0.0
C&DH	2.0
Total cost subsystems	426.5
Total R&D	400
Total M&T	210.6
Total launch cost	15.4
First unit cost	1050.7
No. CTTV [-]	*22*
Total cost	14539
Average investment per CTTV	660.9
Lifetime [years]	*30*
Yearly cost per CTTV	22.0
Average operation cost per year	0.1
Average maintenance cost per year	3.0
Total cost per year per CTTV	25.1

Table 7.5: Detailed cost of the LSAM.

Item	**Cost [M€]**
Propulsion	5.4
GNS	1.2
Thermal Control	0.3
ADCS	10.0
Power supply	0.9
TT&C	2.4
Structural materials	4.0
C&DH	2.0
Total cost subsystems	26.2
Total R&D	1000
Total M&T	284.2
Total launch cost	91.9
First unit cost	1402.3
No. LSAM [-]	*9*
Total cost	10703.7
Average investment per LSAM	1189.3
Lifetime [years]	*10*
Yearly cost per LSAM	118.9
Average operation cost per year	0
Average maintenance cost per year	10
Total cost per year per LSAM	128.9

The subsystem costs follow from Ch. 6. The total manufacturing and testing costs use the same relationship as the CTTV [60]. The research and development cost are assumed to be twice the development cost of a small lunar lander [170]. The launch costs to LLO are assumed to be 6650 €/kg [170][171]. After applying the learning curve stated in Sec. 3.4 the average cost per LSAM was computed. Taking O&M into account (operational cost are neglected since it uses propellant available on the Moon), the total annual cost per LSAM is computed.

7.4.2 Overall Annual Cost

For the overall annual cost all the costs for the segments have to be combined. The precursor missions have been treated in Sec. 4.10 and are will be included in the annual cost in Sec. 7.4.3. For every segment the annual investment cost is computed using the lifetime and investment cost. The yearly operational cost (incl. maintenance) was found and the results are documented in Appendix A. For the CTTV and LSAM the values for best and worst case indicated are the values found in Sec. 7.4 and 7.5 with a margin of 10%. Adding the annual cost per segments gives an annual cost of 26.7 - 393 B€/year.

7.4.3 Additional Costs

Before the mission can start operating, several precursor missions will be required to establish for example the lunar base and the Space Dock. In Sec. 4.10 the costs are estimated on approximately 1.5 T€ total, or 50B€ per year based on 30 years. The crew missions of approximately 1 B€ are included in the annual costs.The precursor costs for 0.1 and 1% are 46.2 and 177 B€ respectively. The nuclear power plants on Earth will need a large budget of 5 - 11.5 B€ per power plant [149]. The nuclear power plants needed are estimated at 2100 - 2350 power plants to provide the 10% energy demand (1100 MW per plant). A fusion plant is estimated to cost 5 - 11.5 B€ [149]. This will yield a cost of 350 - 904 B€ on an annual basis for 10% energy demand, including operations and maintenance [149]. A learning curve is not applied for the fusion plants, since cost developments for different power plants have proven to be unpredictable due to imposed regulations over the previous decades [250]. Adding all the costs yields a total annual cost of 427 - 1347 B€ .

7.4.4 Scalability

Since the mission requires a complex infrastructure, it might be an economic possibility to start the mission on a lower scale. Using the information stated in the previous chapters and the same method for the cost estimation of the different segments the total costs were found and summarised in Tab. 7.6 (including nuclear power plant costs). The cost per MWh is calculated using the energy demand stated in Sec. 2.5. Note that this cost does not account for salaries and other expenditures of a commercial company. Thus, the actual price of He-3 fusion energy will be slightly higher.

Table 7.6: Annual Cost for 0.1, 1 and 10% of the global energy demand.

Scale	**Total Overall annual cost [B€]**	**Cost [€/MWh]**
0.1 %	7.7 - 20.5	34.2 - 100.0
1 %	45.6 - 140.3	20.3 - 68.4
10%	427 - 1347	18.9 - 65.7

7.5 Return on Investments

An important factor when designing a mission is the return on investment. Investors have to be confident the mission will be profitable. This means the He-3 transported back to Earth is sold at a market price with a sufficient profit margin. To establish the return on investment, the market price, market volume and achievable market share have to be determined for the He-3 fusion. The annual cost of the mission was determined in Sec. 7.4.

Market Price: The market cost price of energy in 2040 is uncertain: fossil fuels might run out and renewable energy sources might increase their market share due to improved technology. In this market He-3 fusion will try to get a market share, which means the cost price should be at least comparable to that of the other energy resources. The price range for renewable energy sources in 2040 was found to be approximately 80 to 340 €/MWh [251][14][6]. To make He-3 fusion a feasible option it should be able to compete with energy prices from fossil fuels as well. Exxon predicts an electricity cost price of 46 to 115 €/MWh [5], while as Enerstrat predicts a price in the range of 30.4 to 68.4 €/MWh [25]. The electricity produced with He-3 fusion should be at the low end of those price ranges in 2040 to be competitive in the market. The estimated minimum market price of energy, 30.4 €/MWh will be used in the evaluation of the return on investment. Note that the prices are based on production costs and mainly on U.S. priced energy predictions, which are generally lower than European prices. However, the energy has to be provided globally, resulting in the low-end of the competitive price range for this mission.

Market Volume: The energy demand in 2040 needs to addressed to determine the market volume for the He-3 used for nuclear fusion. The total energy demand in 2040 was estimated to be approximately between $2.05 \cdot 10^{11}$ and $2.25 \cdot 10^{11}$ MWh [5][6]. The market volume for energy is expected to increase even after 2040, which will put pressure on the energy market price. The goal of the mission is to supply 10% of the energy demand stated.

Achievable Marketshare: The goal of the mission is to provide 10% of the global energy demand in 2040. The fusion technology might not be fully developed at that time and with the increasing energy demand the final market share will most likely be less than the 10% aimed for. However, ethical issues may arise or issues concerning the reliability of the mission. If a failure occurs the whole energy production might get stalled, which is not acceptable. However if reliability and technology increase, nuclear fusion using He-3 from the Moon might achieve a large market share [252][253]. Even more if regulations like carbon emission taxes for fossil fuels are introduced. Nevertheless, the achievable market share is difficult to predict because it depends on the development of its own technology, policies and development of other energy sources.

Annual Expected Profit: Using all the cost estimations from Sec. 7.4 and Appendix A the annual cost per year for the mission was estimated. The expected revenue follows from the expected market volume and the market price. For the market price 30.4 €/MWh will be used and for the volume $2.05 \cdot 10^{10}$ and $2.25 \cdot 10^{10}$ MWh. The annual expected profit is estimated in Tab. 7.7.

Table 7.7: Expected annual profit.

	Best Case	Worst Case
Total Expected Annual Revenue [B€]	687	623
Total Annual Costs [B€]	427	1347
Total Annual Expected Profit [B€]	260	-724

The best case, when looking at pure profit, would be the high energy demand with lowest cost. It is assumed that the 200 tons of He-3 can supply the worst and best case of energy demand (i.e. mission efficiency increases). Based on the estimations, the mission can be profitable in the best case. However, investors have to be found before the mission can be operational.

7.5.1 Scalability

For scalability the results are summarised in Tab. 7.8. For the 0.1 and 1% scales the manned missions are taken into account. Under current conditions, the 0.1% mission is estimated to be non-feasible. The other cases might be feasible under optimal conditions. The economic feasibility will increase when energy prices rise, since for these estimates a worst case energy price is used.

Table 7.8: Expected profit for 0.1, 1, and 10% of the global energy demand.

Scale [%]	**0.1**	**1**	**10**
Annual Overall Cost [B€]	7.7 to 20.5	45.6 to 140.3	427 to 1347
Annual Expected Revenue [B€]	6.2 to 6.9	62.3 to 68.7	623.2 to 687.0
Annual Expected Profit [B€]	-0.8 to -14.3	-78 to 23.1	-724 to 260

7.6 Sensitivity Analysis

In this section, a sensitivity analysis on the mission will be performed. Usually a sensitivity analysis yields the analysis of the output of a system due to uncertain input parameters. However, due to the various segments involved in this mission and their interrelation, it is impossible to perform such a sensitivity analysis for the entire mission. It is therefore analysed how sensitive the mission is in terms of scalability, risk, cost and sustainability when mission driving factors like fusion technology, He-3 demand and component availability change. The analysed aspects are based on the trade-off criteria as applied for for the Midterm Report [4].

7.6.1 Scalability

The scalability of the mission is mainly dependent on the state of available fusion technology, corresponding He-3 supply and component availability. It is expected that fusion technology will start to be commercially available in the year 2040. However, it is likely that the technology is still applied on small scale, yielding a small He-3 demand. If the He-3 demand turns out to be even smaller than the 0.1% energy demand, redundant components (too much CTTVs and LSAMs), a potentially too large Space Dock and a too large lunar base might be present. If that would occur, its main influence would be on the return of investment of the mission. That would take longer, or worse, it would not be economically viable at all if the He-3 demand does not increase. If the mission would have to be up-scaled, more (or larger) He-3 canisters would need to be transported. However, that will only be possible if for example the lunar base and Space Dock are big enough, if there are sufficient miners, LSAMS, CTTVs and Skylons or, if all those components are able to handle bigger He-3 canisters. Therefore it is very dependent on the availability of the components. As the availability of each of the mission parts is at least medium (but mostly high), as explained in Sec. 7.7, the mission feasibility is probably not very sensitive while up scaling the desired He-3 transport.

7.6.2 Risk

For the risk sensitivity of the mission, fusion technology availability and defect components will be considered. There are obviously more aspects which can influence the risk of the mission, but the mission is considered to be most sensitive to those two aspects. Another risk is the assets required before the mission is operational, if investments stop the whole mission is endangered. The risk that fusion technology might not be available in 2040 and potentially even after 2040, is the most hazardous threat. There is a risk of unavailable or not fully operational segments of the mission in 2040. It is dependent which segment is not available to determine the impact on the mission. If for example Skylons are not fully operational yet, they can be replaced by conventional launchers. Also CTTVs could be replaced by conventional transfer vehicles, if they are adapted such that they can carry the He-3 canisters. On the other hand, if a Space Dock or lunar base might not be available on the required scale, it might be impossible to proceed with the mission. Alternative solutions for these segments should therefore be considered in further studies on lunar He-3 mining.

7.6.3 Costs

The sensitivity of costs of the mission, goes hand in hand with the scalability of the mission and has overlap with risk. As already mentioned in the scalability section above, the return of investments could be disastrous if all investments are already done while the He-3 demand becomes lower than expected. It might happen that investments needed to develop the new technologies become higher than expected. However, as in the feasibility study it is expected that the designed mission will not be very sensitive for an increase in cost.

7.6.4 Sustainability

The sustainability of the mission is probably the most sensitive aspect of the mission with respect to unexpected changes. If for example the Skylon would not be available, one would have to switch to unsustainable conventional rockets. However, especially on the Moon sustainability might easy be undermined. If for example in-situ resources are not sufficient retractable. From the lunar regolith, propellant needs to be transported all the way from Earth (or alternative propulsion techniques need to be considered). Next to that, considering the number of He-3 miners needed, it might be unsustainable if the He-3 demand increases. Other, relatively more sustainable options should then be considered for further studies.

7.7 Reliability, Availability, Maintainability and Safety Evaluation

In this section, a Reliability, Availability, Maintainability and Safety (RAMS) analysis of the entire mission will be performed. To keep an overview, every part of the mission (in order of Ch. 4) will be analysed separately. Each aspect of RAMS will be evaluated separately.

7.7.1 Reliability

For the *payload design*, reliability is high. Although there are not yet He-3 canisters specifically designed for space purposes and although these canisters have not yet the size as needed for the mission, there are proven designs of smaller He-4 canisters [54][55]. It yields the reasonable assumptions that in 2040, these canisters have high reliability. The *Space Dock and docking systems* score high on reliability. Larger and more complex space stations like Mir and ISS have already shown their reliability [94]. The highest risk comes from the propellant depot attached to the dock, but there are already plenty of concepts and small designs [96]. Also the docking and berthing systems are already maturing in development and optimisation [167][168]. The reliability score for the *communications* is also high. Existing ground stations will be used, which already showed its reliability [214]. Moreover, there will be redundant stations available because both ESA and NASA own ground stations used for deep space communications. For the relay satellites, satellites similar to ESA's EDRS will be used. Although they are not available yet, its technology is proven in space industry by means of NASA's TDRSS [216]. Although the *Skylon* is not yet operating, it is already far developed showing confidence in the design [254][83][81]. Next to that, there are already Space Plane companies like Virgin Galactic, which have proved the reliability of Space Planes to a low orbit. Reliability is therefore considered high.

The reliability of the *CTTV* is hard to analyse, as it has not been applied in space yet at the desired scale. However, serious research is going on, again showing confidence in the design [107]. Next to that, it is researched how a CTTV can be applicable in combination with a spiral transfer. Moreover there are sufficient redundant parts in the CTTV available to increase the reliability. Since the mission is scheduled for 2040, it is assumed that reliability will be medium to high. The *LSAM* does better when it comes to reliability and scores high again. It uses proven technology, making it reliable [114][240]. Furthermore there are sufficient redundant parts integrated in the design, such that it keeps working even if some sub-parts break. The reliability score for the *lunar base* is the lowest. A similar project has never been performed, resulting in high uncertainties of building such a base. Risk can not be derived from references, yielding a automated increase of risks in for example mass, power, cost and feasibility of a lunar base. The *(manned) precursor missions* get a high reliability score. The precursor missions will make use of conventional rockets, which make use of fully proven and mature technology. Furthermore these rockets are nowadays used for manned missions to the ISS. In conclusion, reliability scores overall high.

7.7.2 Availability

For the *payload design*, availability of canisters will be high. He-4 canisters already exist on small scale. Next to that, canisters containing other pressurised liquids are already widely available. A supply of 90 canisters per year is therefore expected to be of no problem. The availability for the *Space Dock and docking systems* is also high, derived from existing space stations like ISS which are continuously available [94]. Also the *communications* availability is high. As addressed, existing ground stations will be used which proved to be available at all times. Next to that, there are plenty of redundant ground stations, such that full-time communication can be provided. Also the availability of the relay satellites will be continuous, especially since redundant satellites will be available. For the *Skylon*, availability is dependent on the number of Skylons available. If sufficient funding is available to buy additional and redundant Skylon, availability will not be a problem. However, if limited Skylons will be available then availability might become an issue due to the short equivalent operational life of 1.35 year [255].

Availability of the *CTTV* is dependent on how many CTTVs will be deployed for the mission, but is expected to be high considering the number of CTTVs needed. Next to that, there will be the possibility to have sufficient redundant CTTVs available as back-up yielding a high availability. For the *LSAM*, the availability might be the bottleneck as the LSAM will only operate between LLO and the lunar base. This means that LSAM back-ups would have to be transported all the way from Earth if not enough capsules are available. However, due to its high reliability it is assumed that LSAM will be highly available. For the *lunar base*, the availability aspect scores highest. The size of the base is assumed such, that always a part of the base will be available for the crew. It is derived from the ISS [94], which also provides enough space to house the crew if one of the components of the dock can not be used. The availability analysis of operational equipment on the base is left for further study. The main downside of the *(manned) precursor missions* however, is the limited availability of the conventional rockets. This might be the bottleneck for this part of the mission, as a large number of precursor missions are required due to for example the relay satellites, lunar base and Space Dock. Therefore, the availability of this part of the mission is medium.

7.7.3 Maintainability

For *payload design*, the canisters are expected to require less maintenance , because of the limited number of components. Next to that, it is expected that enough redundant canisters will be available to replace non-working canisters. The maintainability efforts for the *Space Dock and docking systems* are not very high, considering the ISS as reference. Next to that, the crew does not have to perform any scientific missions meaning they have sufficient time to maintain the dock. The maintainability of *communications* is low, as proven by existing ground stations, relay satellites and other communication equipment [214][216]. If maintenance would be needed it would be an expensive operation due to the large distances involved and the high costs of the equipment itself. For the *Skylon*, maintainability will be high. The Skylon is a reusable space plane, but the downside is the maintenance involved after each flight. Relative to the operational lifetime of only 1.35 year, it can be concluded that maintenance efforts for the Skylon will be high.

It is expected that the maintainability of the *CTTV* will be medium to low. As explained, the reliability of the CTTV tends to be high especially since sufficient redundant components are included. Therefore it is estimated that maintainability will be medium to low. The biggest downside is that maintenance is only possible at the Space Dock. Maintenance efforts of the *LSAM* are estimated to be low. Again, due to the mature and proven design it is expected that maintainability is low [114][240]. Next to that, there is a lunar base where maintenance can take place. For the *lunar base*, it is uncertain how much maintenance will be required what again reduces the RAMS score. However, if it is assumed that the maintenance of such a base will be similar as the ISS, then maintainability scores medium. The difference in comparison with the Space Dock comes from the assumption that the crew at the lunar base will have less time available for maintenance compared to the Space Dock crew.

7.7.4 Safety

The safety of the *payload design* scores lowest in comparison with its other RAMS aspects, as the canisters would need to be pressurised to keep the He-3 in liquid state. However, due to the proven design of other pressure vessels it is expected that also the safety of the payload design is high. Safety of the *Space Dock and docking systems* is also high. Again, the safety analysis is based on similar space stations like ISS. The ISS, together with its docking and berthing systems, has already proven to be a safe and mature design over the past decade. It is hard to analyse the safety of the *communications*, as there are no risks that safety of humans will be endangered. Therefore it assumed that the safety of the communications is high. The safety of the *Skylon* is high. As addressed before, there are already existing and proven designs in space industry of planes reaching low orbit. The Skylon does have another 30 years to optimise the design and maturity, concluding that the safety of the Skylon will be high by the year 2040.

A high safety level can be assigned to the *CTTV*. The CTTV will shuttle between the Space Dock and the LSAM were, except from the Space Dock, no humans are involved. Therefore the safety of humans will not be endangered, yielding a high safety score. Also the safety of the *LSAM* is high. Again, in similarity with the CTTV, there are no humans present and endangered in the LSAM yielding a high safety. The *lunar base* has some safety issues. For the ISS it is known that humans are not allowed to be in space for more then six months due to safety concerns. As mentioned, a similar base has never been established before, resulting in a lack of experience and knowledge on for example the presence of a crew on such a base. Furthermore, the crew on the lunar base is far away from any external human support, resulting in a medium to low safety. The safety for the *(manned) precursor missions* is medium to high. As addressed for the precursor missions, the conventional rockets are already a proven and mature technology. Therefore it would be reasonable to set the safety-level high. However, there is always a risk which could potentially endanger human life. Therefore it is concluded that the safety of this part of the mission is medium to high.

7.8 Sustainability Evaluation

In this section, the different aspects related to the sustainability will be evaluated individually on their performance. Again, it must be stressed that, per nature of space missions, it is difficult to systematically employ sources of re-usable materials or energy. Therefore, sustainability is mainly assessed through the most efficient and optimal use of resources, and sustainable usage is employed whenever possible.

7.8.1 Minimisation of Propellant Spent and Number of Flights

The proposed mission consists of three flight elements: the Skylon Space Plane, the CTTV and the LSAM. Skylon produces an exhaust that purely consists of water vapour. While this is an important greenhouse gas, it does not acidify the rain or impact the Ozone layer. The CTTV uses Argon as a propellant, a gas that can be synthesised from the atmosphere, and is abundantly available on Earth. The CTTV additionally uses engines that have a high I_{sp}, which implies very high efficiency of the engines. Lastly, the LSAM uses propellant extracted from the Moon. While using propellant from the Moon increases sustainability in the sense that it does not have to be transported from Earth, the exhaust gases still pollute the Moon atmosphere, and are non-renewable.

The number of flights for all three flight elements were reduced as much as possible by optimising the transports for maximum payload transportation in the least possible number of flights.

Lastly, both the CTTV and the entire lunar operations run on electric power derived from nuclear fission. This is in itself a power source that induces risks of meltdown, and raises the question of where to store the nuclear waste. It was decided not to bring back the spent fuel to Earth, because risks during descent through the atmosphere could have disastrous results. Therefore, it was decided that the waste could be stored in the porous mantle of the Moon. However, in the long run, another solution for the disposal should be found.

7.8.2 Re-usability of Spacecraft

The proposed mission excels in this aspect of the sustainable development approach. All vehicles are designed for maximum re-usability. Skylon, the CTTV, the LSAM and even the lunar operations all are fully-reusable for each aspect of the mission they cover.

7.8.3 In-Situ Resource Utilisation

Especially for the detailed design of the LSAM, ISRU was taken into account. Its propellant is lunar Methane and lunar Oxygen, both derived during the extraction of He-3 from the lunar regolith. Additionally, the subsystems of the LSAM were designed to use as much Silicium and other minerals as possible, all available from the lunar regolith.

7.8.4 Disposal of End of Life Objects

Because this study focused mostly on the operations, the end of life aspect of the mission was not extensively covered. For now, it is considered that it is not worth returning any of the used vehicles to Earth. The vehicles are sent to a graveyard orbit, or brought down to the Moon for disassembly and spare parts. The Space Dock will re-enter, for example into the Pacific Ocean, after EOL.

7.8.5 Conclusion on Sustainability

Most space missions are per definition not sustainable. The heavy-lift launch vehicles often function on polluting propellants, use expensive and rare materials, and pollute the Earth environment and orbits, sometimes even space in general. While this mission is not very sustainable either, some important design aspects improve its sustainability performance considerably, ranging from the high specific impulse delivered by the CTTV, to the more efficient use of lunar resources. Again, it is stressed that sustainability for this mission is mostly defined as maximised efficiency, rather than use of renewable materials and/or energy. However, improvements can still be made, with respect to nuclear waste disposal, end of life disposal and the polluting of the Moon and Earth atmosphere. If this feasibility study is ever to become form a very rough basis for a mission, it is strongly advised to further investigate the sustainability of such an endeavour.

7.9 Operations and Logistics

In this section the Operations and Logistic Concept (OLC) description will be given. For this, a block diagram format will be used. The goal of this is to illustrate the use and support of the system; it shows how each element is used and supported in relation to another element. The lay-out which is generated here, already determines system characteristics which must be part of the design. For example, the presence of human personnel on the lunar surface dictates that a protective habitat must be provided (which will be part of the lunar base). Also, coming from that example, the transfer should be split up into two possibilities (one for cargo, and one for crew).

The system is divided into three general segments: the Terrestrial Segment (TS), the Spatial Segment (SS) and the Lunar Segment (LS). These are different than the aforementioned segments used throughout this report. This was done as to get a more correct and comprehensive overview on the nature of the operations. The strict boundaries of these segments are defined as:

- The TS comprises of the activities on Earth including the launch/landing procedures.
- The LS comprises of all activities on the lunar surface, including launch/landing procedures.
- The SS comprises of the operations which lie outside of the above two segments. This, of course, is limited to a region around the Earth-Moon two-body system (cislunar). The main contributor to this segment is the transfer of the desired goods (He-3). Also external communication relays and a Space Dock are part of this segment.

Operations are defined as: *"Operations transform resource or data inputs into desired goods, services, or results, and create and deliver value to the customers. Two or more connected operations constitute a process, and are generally divided into four basic categories: processing, inspection, transport, and storage"* [256].

We acknowledge that the mission which is being designed for includes all four operations processes. Regolith will be processed to retain He-3, He-3 will be transported (next to all needed resources and supplies etc.) and He-3 will also be stored. Inspection is an important process (filling in the supporting role, as part of maintenance) which involves many of the operations.

Logistics are defined as: *"Planning, execution, and control of the procurement, movement, and stationing of personnel, material, and other resources to achieve the objectives of a campaign, plan, project, or strategy. It may be defined as the 'management of inventory in motion and at rest'.* [257]

Operations and Logistics might appear to be rather similar, but it must be kept in mind that operations describe mainly the actions undertaken during processes, while the logistics describe the movement of various elements within the processes and system. In a sense, the blocks in the diagram represent operations, while the the arrows represent the logistics. Even though some arrows do not seem at first hand to be contributing to the overall logistics, they do support those actions. For example, providing communications is not a pure logistic action, but by supplying communications to the lunar segment it enhances the logistics in that segment.

Within the space mission life cycle, "operations and support" is the fourth and last phase (after "concept exploration", "detailed development" and "production and deployment"). It is defined as: *"the day-to-day operation of the space system, its maintenance and support, and finally its de-orbit or recovery at the end of the mission life."* [60]. This also means that in the recurring phase-diagram the deployment will be kept out of the mission lay-out. Therefore, an additional diagram will be provided for the setup phase, as to include the setup operations of the lunar base into the general overall operations.

The centrepiece of the whole mission operation is the Mission Control Centre (MCC). From here, all segments are controlled and supported. Decisions are taken which concern critical characteristics of the mission. To give a quick example: deciding on the plan of action for conducting mining operations is not done by the lunar component, instead it is the responsibility of the MCC to come up with a decision, which will then be carried out by the LS. In contrast, the manoeuvring of a mining vehicle is primarily the concern of the LS. The MCC is coupled to several subordinate components: Communication interface, Tracking & orbit control, Lunar Operations Control System, Spacecraft Operations Control System, and other. They all together form the Terrestrial (Ground) Support.

Terrestrial (Ground) Support comprises of many functions performed at a ground station. Furthermore, this is not a scientific mission according to the postulated requirements. Therefore data is not of primary importance. It is only used in a supporting role, in order to support the vital communication and control functions (tracking, orbit control etc.). In conclusion, the communications interface and lay-out will be kept at a minimum throughout the diagram. Furthermore, autonomous systems (CTTV and LSAM) lower this requirement even more.

The Terrestrial segment is completed with: the storage and (re)distribution system, the launch and landing sites of Skylon, the launch and payload systems and the final product provision to the customers (energy production plants/facilities). The relations are shown accordingly in Fig. 7.1. Keep in mind that full lined arrows are meant for segment interaction, while dashed lines were used for denoting sub-segment process interaction. Maintenance support is an operation which one can find throughout the whole system.

The SS consists of the Space Dock, the communication satellites, and the CTTV. Furthermore, the transfer operations shown in Fig. 7.1 apply for the unmanned He-3 transfers. For safety reasons (mainly radiation), the crew missions will be flown through direct transfers, during 50-110 hours. These were excluded from the diagram, due to the complex lay-out. For the manned mission, the difference with the cargo transfer is that the Spatial Segment would only encompass the communication satellites, while the Terrestrial launch would be linked directly to the lunar landing.

The LS comprises of all activities undertaken on the surface of the Moon (mining, processing, storing, etc.), the launch and landing of the LSAM, the power plant (or network of individual power plants) and the lunar transportation infrastructure.

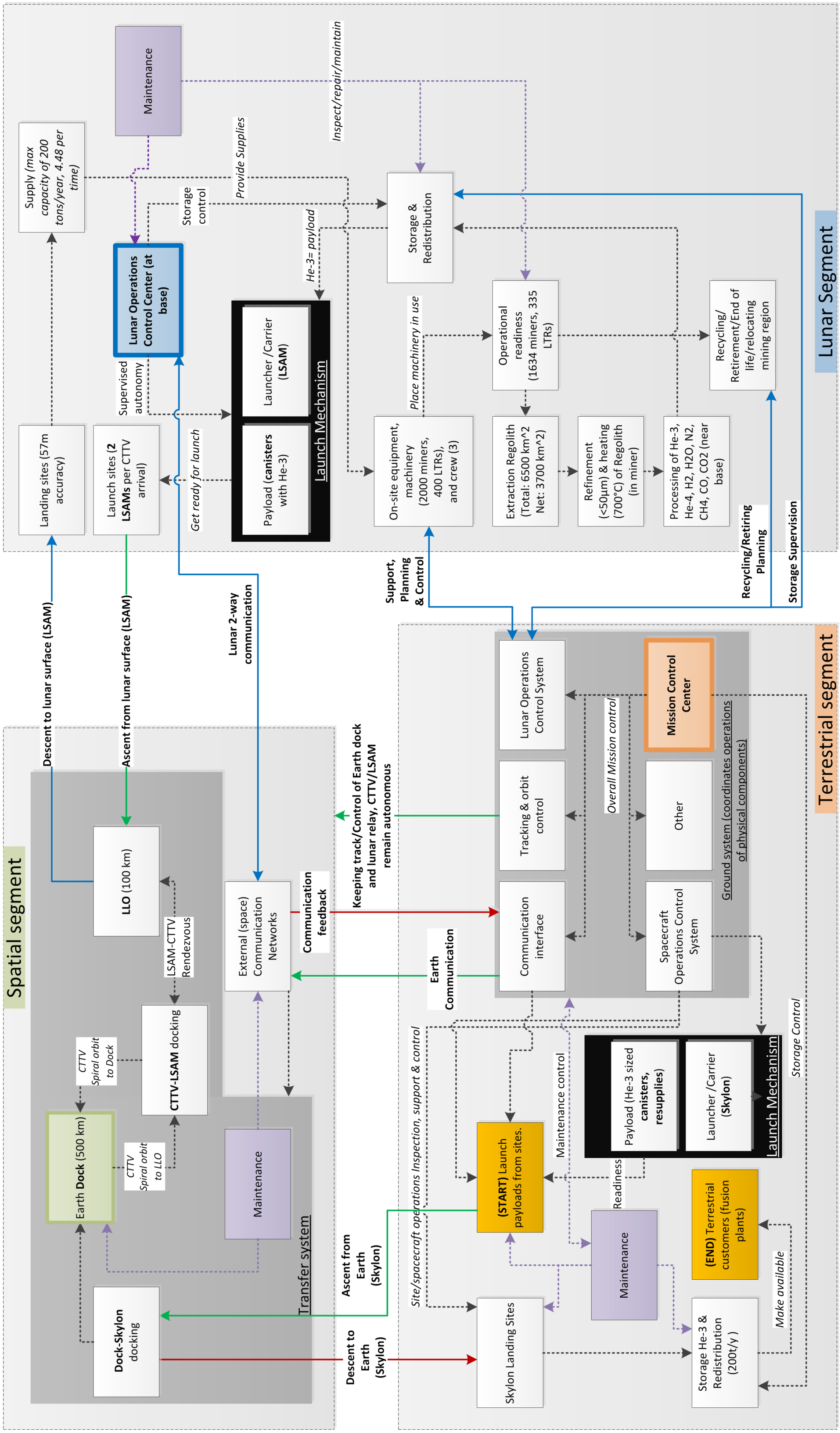


Figure 7.1: The 'Recurring' Operations Flow Block Diagram.

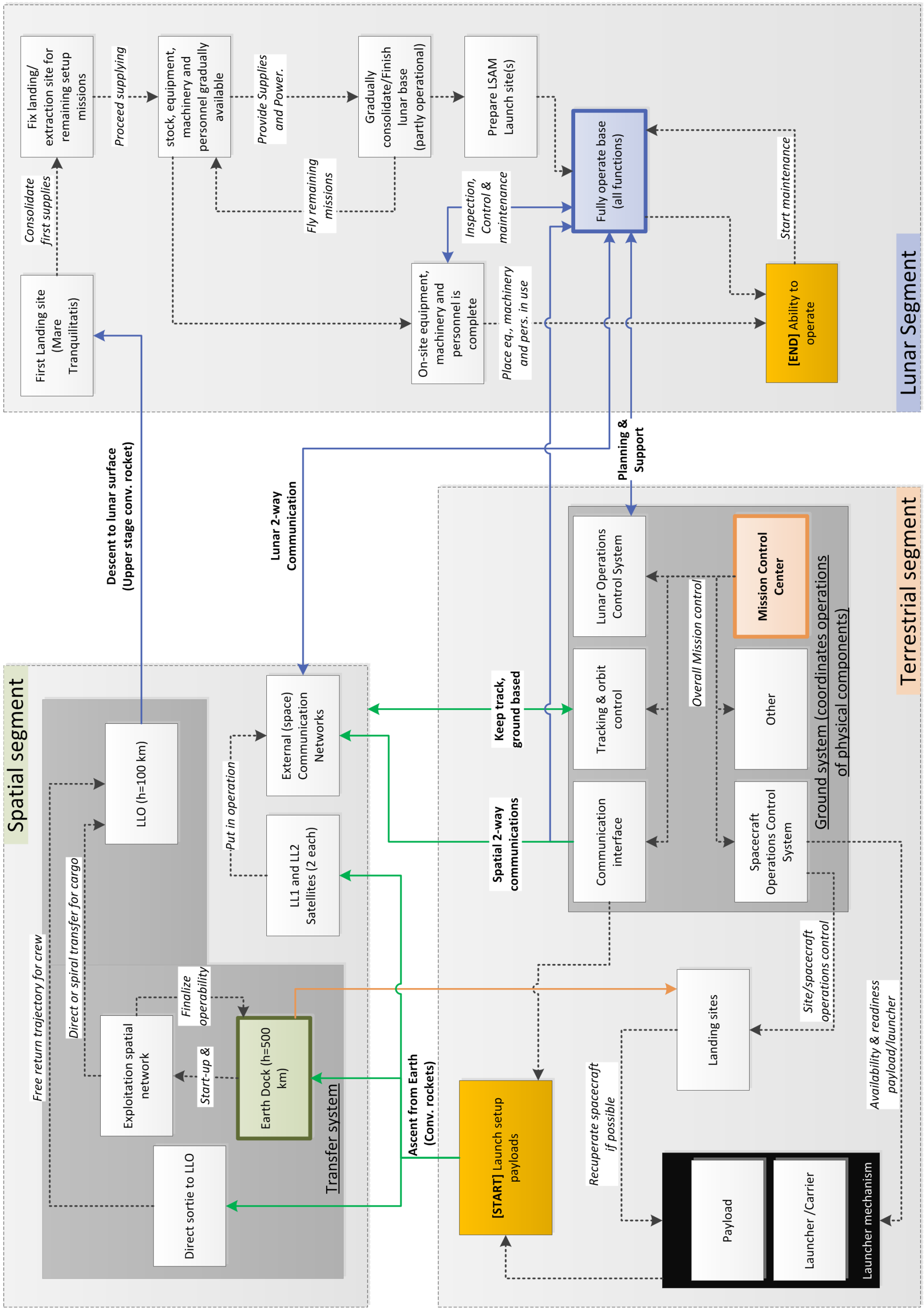


Figure 7.2: The 'Setup' Operations Flow Block Diagram.

Fig. 7.2 shows the lunar system set-up diagram. The same convention with respect to lining is used as in Fig. 7.1. The set-up phase mainly differs from the recurring case in that it has to come up with stable nominal operating conditions for all sub-segments. Some loops are therefore present to account for the building-up of the system. Maintenance support is an operation which is introduced throughout the whole system, once it exits the deployment phase. More on the precursor and manned missions can be found in Secs. 4.10 and 4.11 respectively.

8. Mission Feasibility

To answer the pending question of this mission (whether it is feasible to go to the Moon and extract He-3 for Earth-bound usage), this section will put the feasibility study into perspective. The section is divided into a discussion on political, economical and technical feasibility, after which a conclusion is provided.

Technical Feasibility: There is no doubt that Earth needs a new source of energy. The current dominant energy sources are polluting, and expected to run out in a few more decades. Clean, renewable sources are in steady development, though their current capabilities are not as scalable as originally were hoped to be. Unless scientific breakthroughs occur in the coming years (which is very plausible), a definite gap in the energy market is waiting to be filled. This is a gap for an alternative, new source, which could very well be filled by energy from fusion. However, fusion technology is still in its infancy, which is the first of many hurdles for this mission. International cooperation projects have been set up, as well as other private initiatives, but results have been slow. Successful nuclear fusion today does not even include He-3 as a possible fuel. Even if this technological challenge were to be surpassed, the challenge of going back to the Moon would have to be faced. Mankind has only been able to put foot on the Moon six times in the past half century, often for only 48 hours at a time or less. These missions alone took years of careful planning, training and engineering. This calls for serious questioning of whether any space agency would be able to set up an advanced infrastructure between the Moon and Earth. Additionally, even if man were able to set up a large infrastructure, another question arises once the lunar base is established. To satisfy 10% of the global energy demand, approximately 640 tons of lunar regolith that is richest in He-3 has to be mined per second. While mankind has faced greater challenges, and while such a large scope is comparable to other major energy sources, such a high mining rate requires an intensely complex mission with technology that not only must survive transportation to the Moon, but also continuous operations in the harsh lunar environment.

Economical Feasibility: If all of the above hurdles were to be tackled, other challenges still remain. The mission would require a total of 427 to 1347 B€/year. However, this mission does generate a profit of up to 260 B€in the best case, as established in Sec. 7.5. As can be seen, large amounts of capital will be required, but large amounts will also be returned. There is also a non-monetary aspect to the return on investment of this mission. The total energy invested in the mission on an annual basis is nearly $1.228 \cdot 10^{18}$ J/year, excluding the energy spent in production of the mission elements, as shown in Sec. 7.2. The mission brings back an estimated $1.1 \cdot 10^{20}$ J/year. It can be concluded that while a significant amount of energy is brought back to Earth, a large amount is also invested in the mission.

Political Feasibility: On the political and ethical side of the mission, several questions arise as well. As established in Sec. 3.2, there is no consensus on what entity is granted ownership of the Moon. Therefore, returning materials from the Moon does not directly establish ownership, causing potential legal confusion. Additionally, while the amount of He-3 on the Moon could potentially provide power for thousands of years, it is not a sustainable way of generating energy: the He-3 is deposited at a far slower rate than it is excavated. During this extraction, the lunar atmosphere could be significantly polluted, the lunar surface is altered, and storage of radioactive waste could pose a threat to any future life. These issues and many more must all be addressed before the mission can actually take place.

8.1 Conclusion on Feasibility

To answer the question of this feasibility study: It is not expected that setting up a space infrastructure, by 2040, to extract enough He-3 to provide 10% of the global energy demand is feasible. While this may seem to be a pretty conclusive statement, some considerations have to be made, as there are some important nuances in the above statement. While it is probably not feasible to perform the entire mission by 2040, it would be possible to 'taper up' the mission. In short, this means that a better alternative would be to start small. Instead of 10% of the global annual demand, it would be a better option to aim for 0.1% of the world energy demand. Dividing the energy requirement by a factor 100, the perspective changes. There are many interesting opportunities by going back to the Moon, not only the extraction of He-3. Other missions could be interested to 'piggyback' on the mission, new technologies would be developed that could have potential uses outside of the space industry, and countless of other advantages. With the experience slowly increasing with lunar operations, the excavations could be scaled up slowly over time, eventually attaining 10% of the world energy demand, or even more. This would be done over the course of more than half a century, possibly even more. Therefore, the final verdict is as follows: while 10% of the global energy demand is not feasible, it would be far more interesting to aim for a smaller number, such as 0.1%, which would be far more feasible, and allow for an abundance of other opportunities.

9. Future Work

The work that has to be performed in order to realise this mission will be discussed in this chapter. Starting with the project design and development logic. Followed by the planning illustrated in the project Gantt chart. At the end the cost breakdown structure will be given.

9.1 Project Design and Development Logic

It is essential that the further continuation of the project is discussed, as the lunar He-3 mining mission had its focus on the feasibility of this rather new and challenging concept. This feasibility study is only one of the many steps that need to be taken to realise the mission.

9.1.1 Mission phases

In space project management terms, the mission is subdivided into seven mission phases [60]. These phases and the corresponding milestones for each phase are shown in Tab. 9.1.

Table 9.1: ESA mission phases and milestones.

Phase	Milestones
0: Mission Analysis and Needs Identification	Mission Definition Review
A: Feasibility	Preliminary Requirements Review
B: Preliminary Definition	System Requirements Review and Preliminary Design Review
C: Detailed Definition	Critical Design Review
D: Production/Ground Qualification Testing	Qualification Review and Acceptance Review
E: Utilisation	Operational Readiness Review and Flight Readiness Review
F: Disposal	-

The project is currently situated in phase 0, although certain elements of phase A might also have been covered. Normally within ESA the Future Missions Preparation Office (SRE-F) is in charge of this assessment phase. During this assessment phase the largest bottlenecks [188], challenges and recommendations were identified. The bottlenecks and challenges that are not elaborated upon in the final assessment remain for further study. Mainly these tasks and activities appear in the future design and development logic.

The future activities to be performed start with an exploration of the concept in the near future, and grow gradually into detailed development as the utilisation phase E is approached within the project. For large, complex space missions typically 10 to 15 years are required for the development and a mission operation time of 5 tot 15 years is not unusual [60]. As the lunar Helium-3 mining mission is by far larger and more complex than past space missions, these numbers will tend to fall be higher. The mission operation time is assumed to last at least 30 years, since the CTTVs were designed for that lifetime. The time available for mission development is 26 years considering the mission needs to be executed at 2040. These development years consists of a roadmap of activities, in which the concept exploration will finally lead to the production and deployment.

9.1.2 Post-DSE activities

In general, the future work which has to be completed for this mission includes: a large amount of research and development for the still unproven concepts, a more elaborate setup of the requirements, and budgeting and planning activities. The overall mission timeline consists of: plannings and development, production, initial launch, constellation build-up (precursor), normal mission operations, replenishment (scaling) and end-of-life disposal. The chronological roadmap of the planned activities to perform the mission is given in Fig. 9.1. The activities in the roadmap are divided into three different categories. This division is made because the owners of these tasks can be split up in three types of stakeholders: the sponsors, developers and operators/users of the mission.

Requirement Generation Activities: The requirements generation activities will be done by the users/operators. The operators control the space and ground assets, and therefore this mainly includes the engineering organisations. The operators should define the requirements on the equipment technically and fiscally. They also should produce and control this equipment during the mission. At the conceptual phase the users state the broad needs, which narrow down to performance objectives and requirements. The operators should ensure that RAMS characteristics are taken into account.

Research and Development Activities: The Research and Development activities are focused on providing a product or capability in time, within the funding constraints. (Sub)contractors, test organisations and the governmental developers are the stakeholders involved in these activities. The developers are held responsible for the acquisition management, from a list of possible options a stable design should be achieved by increasing the technology readiness.

Planning, Programming and Budgeting Activities: The sponsors perform the long-range planning of the project and make estimations of the budgets and funding which are available to meet the needs of the developers, users and operators. To gather investors for this kind of mission is seen as a difficulty; the payback period is rather long.

9.1.3 Post-DSE project planning

The mission timeline is presented in Fig. B.1 in Appendix B. The planning starts with the decision whether to initiate the mission after this feasibility study or not. Then the development and preporatory works are scheduled from 2015 to 2040. And operations are planned from 2040 to 2070. After that it is assumed that mission disposal takes place up to the year 2081.

9.2 Cost Breakdown Structure

After the feasibility study, the costs for executing the mission in the consequent phases have to be identified. Investments have to be made before operations can start. For this a Cost Breakdown Structure (CBS) is created, identifying the major cost elements for the mission (Fig. 9.2).

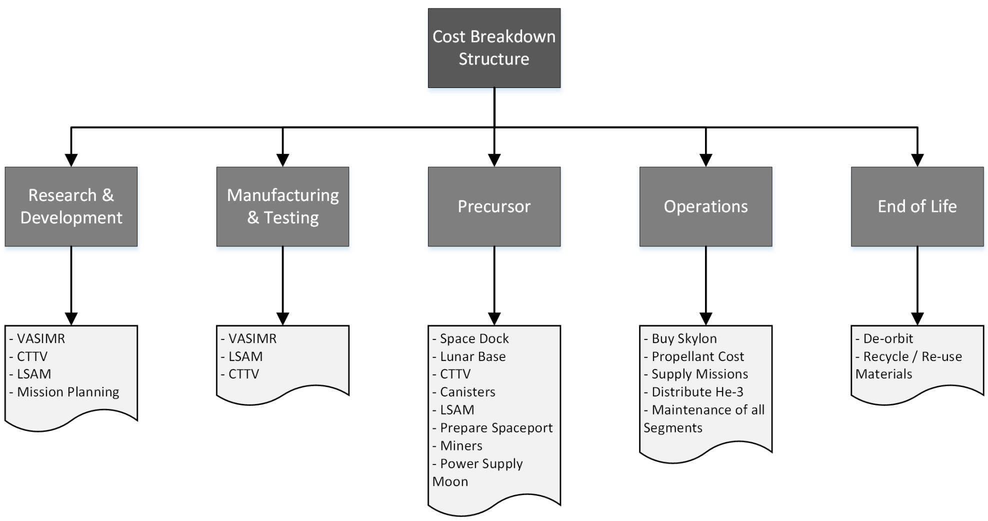


Figure 9.2: Cost Breakdown Structure.

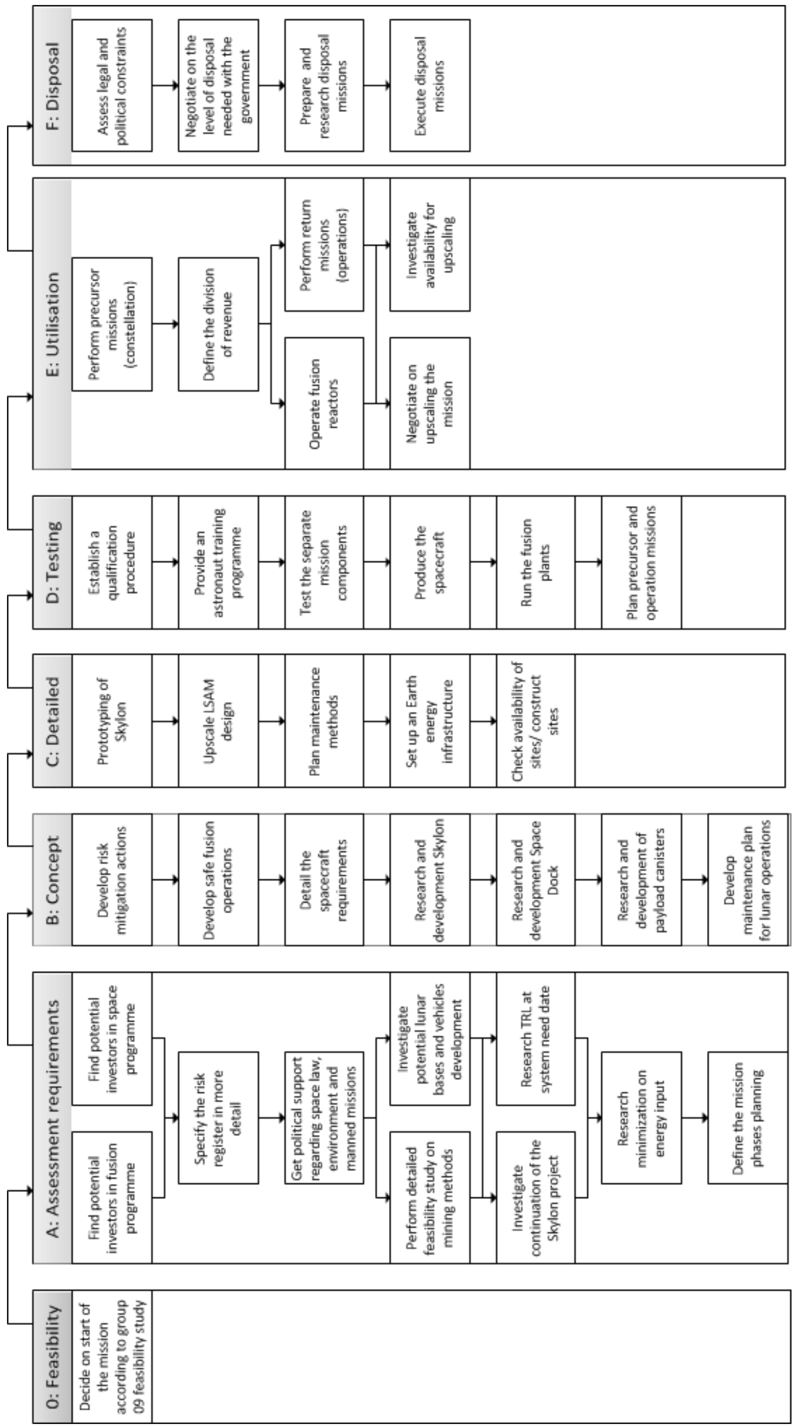


Figure 9.1: Roadmap of the activities to be performed to realise the mission in their corresponding mission phases.

10. Conclusion and Recommendations

A feasibility study on lunar He-3 mining has been conducted for ESA. A large range of aspects of the mission has been taken into account, with a focus on operations. Two of the involved mission elements have been designed in detail: the Continuous-Thrust Transfer Vehicle (CTTV) and the Lunar Surface Access Module (LSAM).

The top-level requirements, summarised, were to design an end-to-end system to mine enough He-3 on the Moon for fusion energy to supply 10% of the global energy demand in 2040. The mission elements had to be designed on a conceptual level and the transport elements on a detailed level. The system shall be both up- and down-scalable.

A systems engineering approach was taken for the design of the mission. After a literature study, a list of possible concept elements for all mission segments was set up. Using TRL scores, unfeasible elements were eliminated. In the following, seven end-to-end concepts were devised, covering all feasible elements. A trade-off was conducted, eliminating concepts and showing strengths and weaknesses of each concept. Subsequently, the well-performing mission elements were combined into three concepts. Another trade-off, evaluation, and re-combination of advantageous mission elements was performed, with a final end-to-end mission as outcome.

The final concept features a fleet of CTTV travelling between Earth and Moon. In LLO, they dock with an LSAM and take He-3 on board. In LEO, the CTTV docks with a Space Dock to exchange payload, refuel, and be serviced. Access from Earth to the Space Dock is provided via a Space Plane (Skylon), which returns the He-3 to Earth, where it is subsequently distributed to fusion plants using existing infrastructure. On the Moon, volatiles are mined from the regolith by mobile miners. A central fixed base processes the volatiles and extracts He-3.

Concerning the energy market in 2040, a literature study was performed. Fusion is expected to enter the energy market around or shortly after 2040. To supply 10% of the global energy demand in 2040, 200 tons of He-3 are required annually. This assumes 2^{nd} generation fusion (He-3 with Deuterium) and an overall efficiency of 67.5%. The Moon can supply this amount of He-3 for several thousand years. Non-lunar sources of He-3 have been considered and discarded due to inaccessibility or too low reserves of the resource.

Additional aspects of the mission have been evaluated. Legal frameworks impose restrictions on exploitation of other celestial bodies. Only a collaborative effort from which all states on Earth can benefit may exploit the Moon. Gaining commercial profit from the exploitation is, however, allowed. Sustainability was addressed by minimising orbital debris from the mission, efficient use of propellant, IRSU, and taking into account exhaustion of gases into the (thin) lunar atmosphere. In preparation for the conceptual design, the radiation environment was considered. Verification and validation procedures of the design process have been addressed.

Every mission segment has been designed on a conceptual level. He-3 will be transported in modular canisters of 10.77 tons each, with 3 meter diameter and 8.55 m length. Each canister will contain 2.28 tons of liquefied He-3. For LEO access, the Skylon Space Plane will be used. It will allow frequent access to the Space Dock, thus decoupling the two segments LLO to LEO and LEO to surface. It can carry a payload of about 12 tons to (supplies) and from (He-3) a LEO orbit of 500 km. Around 150 flights are required per year. The 107.7-ton Space Dock is located in a 500 km, zero inclination orbit. It consists of three modules arranged in a cross pattern. The crew module houses a maximum crew of six. The second module is reserved for maintenance and EVA activities. The third module provides power via solar panels. The Space Dock serves as CTTV-maintenance and refuelling station and as He-3 storage facility. Power, cost, and astrodynamics of the Space Dock have been evaluated on a conceptual level. The first lunar base at the beginning of the mission will be located in Mare Tranquilitatis due to the high He-3 abundance. The base will be the central node for the mining operation. It can support a maximum crew of seven, to conduct maintenance on the lunar miners and the LSAM. Volatile mining will be done by Mark III miners, a concept proposal from the Wisconsin University. With contingency, 2000 miners are required, complemented by about 400 Lunar Transport Rovers. Power for the mining operation will be delivered by nuclear fission reactors and distributed through electromagnetic energy beaming.

In preparation of the mining operation, precursor missions have to bring over 220 000 tons of equipment into LEO, to LL1 and LL2, and onto the lunar surface. Three crew-replacement missions have to be conducted per year, taking into account enhanced radiation levels on the Moon. Over time, increased automatisation would allow for a reduction of crew size and would finally allow to omit physical crew involvement.

CTTV and LSAM were designed in detail. The CTTV concept uses VASIMR engines with continuous thrust to spiral towards the Moon. This transfer requires a total ΔV of 18 km/s for a round trip of 354 days. The wet mass of the CTTV is determined to be 81.54 tons including four He-3 canisters and a total of 27.44 tons of propellant. The refuelling and maintenance is done at the Space Dock. At least 22 CTTVs have to be operational at any time to deliver 10% of the global energy demand by 2040. With an expected lifetime of 30 years the annual costs of a CTTV is between 22.6 and 27.6 M€.

Conceptual design of the LSAM is based on the Apollo Lunar Module and the Altair lander. A detailed design of the subsystems showed that the LSAM will have a dry mass of 13.8 tons. Propellant (71.6 tons), ADCS propellant (1.8 tons), and payload (21.6 tons) add to that. Total launch mass was found to be 108.7 tons. The LSAM was designed to use in-situ propellants. Each LSAM carries two payload canisters. Docking with the CTTV occurs in a

100 km lunar orbit. Each LSAM mission takes 3.5 hours from launch to landing. Annual costs of the three required LSAMs are, over an expected lifetime of ten years, between 116.0 and 141.8 M€.

The end-to-end mission was evaluated. All system elements combined add up to a total mass of 356 000 tons over a mission time of 30 years. The system elements consume a total power of 38.9 GW. A sensitivity analysis was performed for the mission. The main risk of the mission is that critical elements, such as a lunar base, cannot be deployed in 2040. Costs of the mission are significant, but due to the large scope, small to medium increases in one mission element have little effect on the overall costs. A RAMS analysis was performed. Reliability depends on the technology readiness of each mission element and experience with previous similar system (such as a space station). Due to redundant system elements, the availability is good. However, precursor and manned missions present a bottleneck in the design. Most mission elements require considerable maintenance. Nevertheless, maintainability remains good. Safety is rated medium to high, as only the Space Dock and the lunar base are manned and most systems are automated to a high degree. Subsequently, sustainability was evaluated. While the mission is efficient, nuclear waste disposal and (lunar) atmosphere pollution are considerable drawbacks concerning sustainability.

The end-to-end mission was furthermore evaluated with respect to costs. Costs are split into precursor launch costs and annual costs (operational plus equipment cost divided by equipment lifetime). Precursor launch costs are about 1500 B€. Total annual costs are between 427 (best case) and 1347 B€. With a minimum energy price of 30.4 M€/MWh the expected profit is between 260 and -724 B€/year. The profit is only positive if factors such as energy price and fusion power plant costs develop favourably over the years. For 1% of the global energy demand, the mission creates a maximum best-case profit of 23.1 B€. For 0.1% of the global energy demand, the mission only makes losses. Generally, scaling the mining operation up will enlarge profitability.

Taking all aspects and all work done into account, the study reveals that lunar He-3 mining is not feasible (in terms of operations) for providing energy on a large scale. Furthermore, a number of technologies needs to be improved and proven before a large mining operation can be set up. Nevertheless, the mission remains technically feasible. The sheer size of mission renders it politically unfeasible. In particular, the number of miners to extract and process enough He-3 is remarkably high. Legal and environmental aspects add to the difficulties. Economically, however, lunar He-3 mining is possible, if enough initial investment is made. Such a large investment might have substantial negative impact on the global market. Although the net energy outcome of the mission is positive, other terrestrial alternative energy sources may be more favourable to opt for.

10.1 Recommendations

Some recommendations can be made to increase the feasibility of lunar He-3 mining. At the heart of this mission lies the concept of fusion technology. As it is still in a very early stage, it is of fundamental importance to perform more research on the subject. It may be shown that fusion using He-3 is nearly impossible in an efficient manner. Improvements on LEO access vehicles and launchers are critical for the precursor and operational phase of the mission. This could include, but not be limited to, space planes, tethers, space elevators, and Maglev-assisted launches. Additionally, more research should be performed on the subject of space tugs. In particular, low- and medium-thrust electric propulsion should be developed further. The Moon may prove to be an excellent stepping stone and technology testbed for human expansion into the rest of the solar system. However, before the Moon can become a true ”space port” and research centre, an infrastructure must be set up to be able to reliably transfer cargo from Earth orbit to the lunar surface. Lunar He-3 mining can either follow after such an infrastructure is set-up or can be started parallel to expansion to the Moon, creating synergy with other missions. A stand-alone He-3 mining mission cannot be recommended.

Lastly, the lunar segment as a critical element of the mission should be analysed in more detail. While only briefly covered in this report, it is the element on which most of the mission depends during operation. Technology must be developed for automated mining, energy generation, efficient high-rate volatile extraction, and life support on the Moon. Over the years since mankind has reached the Moon, only a few concepts have been proposed for a permanent human presence on the Moon. Volatile extraction will also prove to be crucial for human survival on the lunar surface. Each of the above points involves many other questions and technical challenges that are still left unanswered. Future advances in technology may render lunar He-3 mining feasible. However, this study showed that lunar He-3 mining is unsuitable to provide 10% of the global energy demand in 2040.

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A. Overall Cost per Segment

Overall Costs per segment [Million €]			Best	Worst
1. Ground Segment			**Best**	**Worst**
		Investment Cost	200.0	300.0
		Lifetime [years]	30.0	30.0
	Average Yearly Cost		6.7	10.0
	Operational Yearly Cost		2.6	5.2
	Total Annual Cost		**9.3**	**15.2**
2. Spaceplane			**Best**	**Worst**
		Investment Cost	668.0	856.0
		Lifetime [years]	1.35	1.35
	Average Yearly Cost		494.8	634.1
	Operational Yearly Cost		1169.0	1214.0
	Total Annual Cost		**1663.8**	**1848.1**
3. Space Dock			**Best**	**Worst**
		Investment Cost	9540.0	11660.0
		Lifetime [years]	18.0	18.0
	Average Yearly Cost		530.0	647.8
	Operational Yearly Cost		396.0	484.0
	Total Annual Cost		**926.0**	**1131.8**
4. CTTV			**Best**	**Worst**
		Investment Cost	13085.8	15993.8
		Lifetime [years]	30.0	30.0
	Average Yearly Cost		436.2	533.1
	Operational Yearly Cost		59.4	72.6
	Total Annual Cost		**495.6**	**605.7**
5. LSAM			**Best**	**Worst**
		Investment Cost	9633.6	11773.8
		Lifetime [years]	10.0	10.0
	Average Yearly Cost		963.4	1177.4
	Operational Yearly Cost		81.0	99.0
	Total Annual Cost		**1044.4**	**1276.4**
6. Lunar Operations			**Best**	**Worst**
		Investment Cost	622600.0	11484000.0
		Lifetime [years]	30.0	30.0
	Average Yearly Cost		20753.3	382800.0
	Operational Yearly Cost		39.0	78.0
	Total Annual Cost		**20792.3**	**382878.0**
7. Lunar Base			**Best**	**Worst**
		Investment Cost	8600.0	85500.0
		Lifetime [years]	30.0	30.0
	Average Yearly Cost		286.7	2850.0
	Operational Yearly Cost		1513.0	2270.0
	Total Annual Cost		**1799.7**	**5120.0**
8. Payload			**Best**	**Worst**
		Investment Cost	21.7	27.1
		Lifetime [years]	10.0	10.0
	Average Yearly Cost		2.2	2.7
	Operational Yearly Cost		≈0	≈0
	Total Annual Cost		**2.2**	**2.7**

B. Post-DSE Gantt Chart

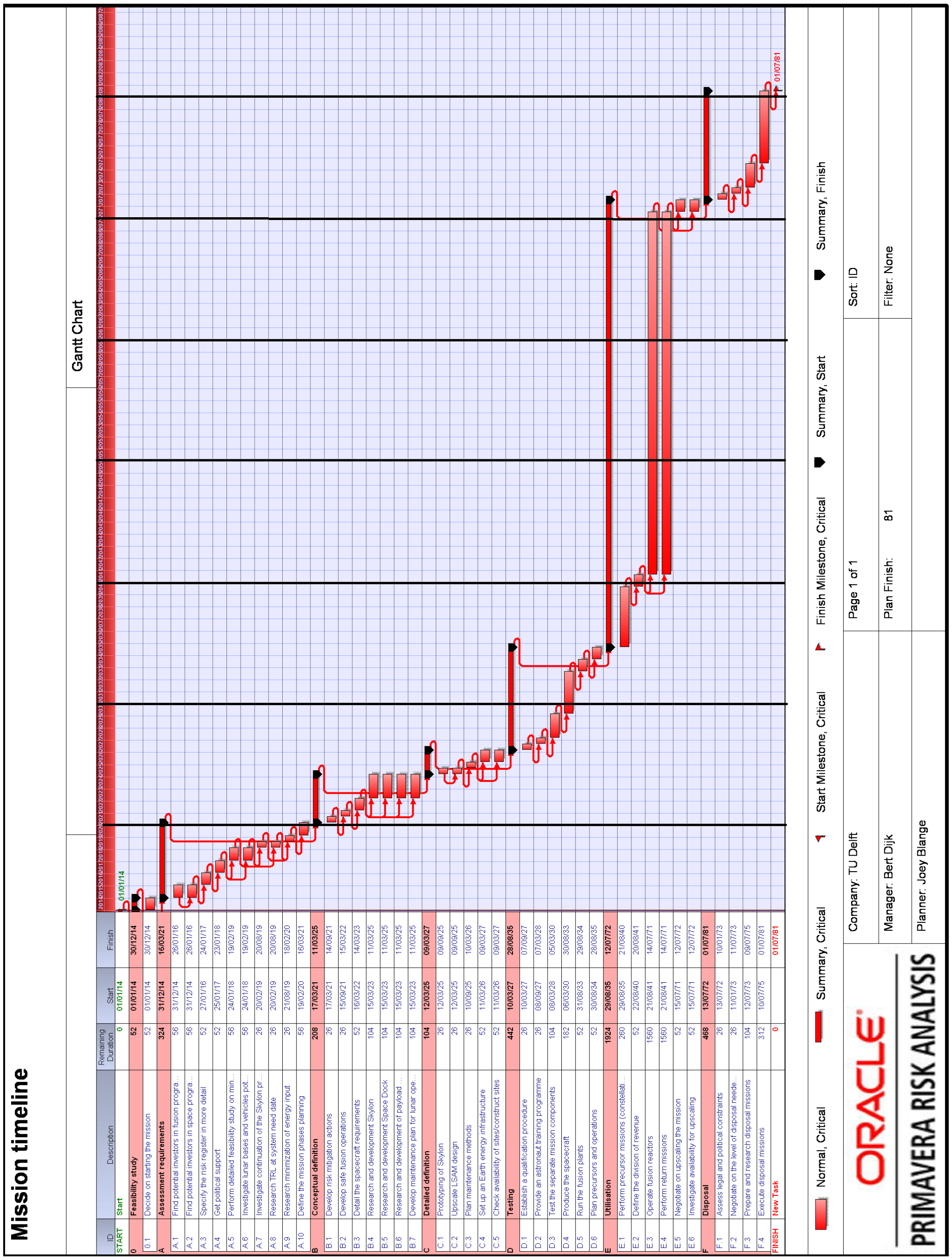

Figure B.1: Post-DSE Gantt Chart.