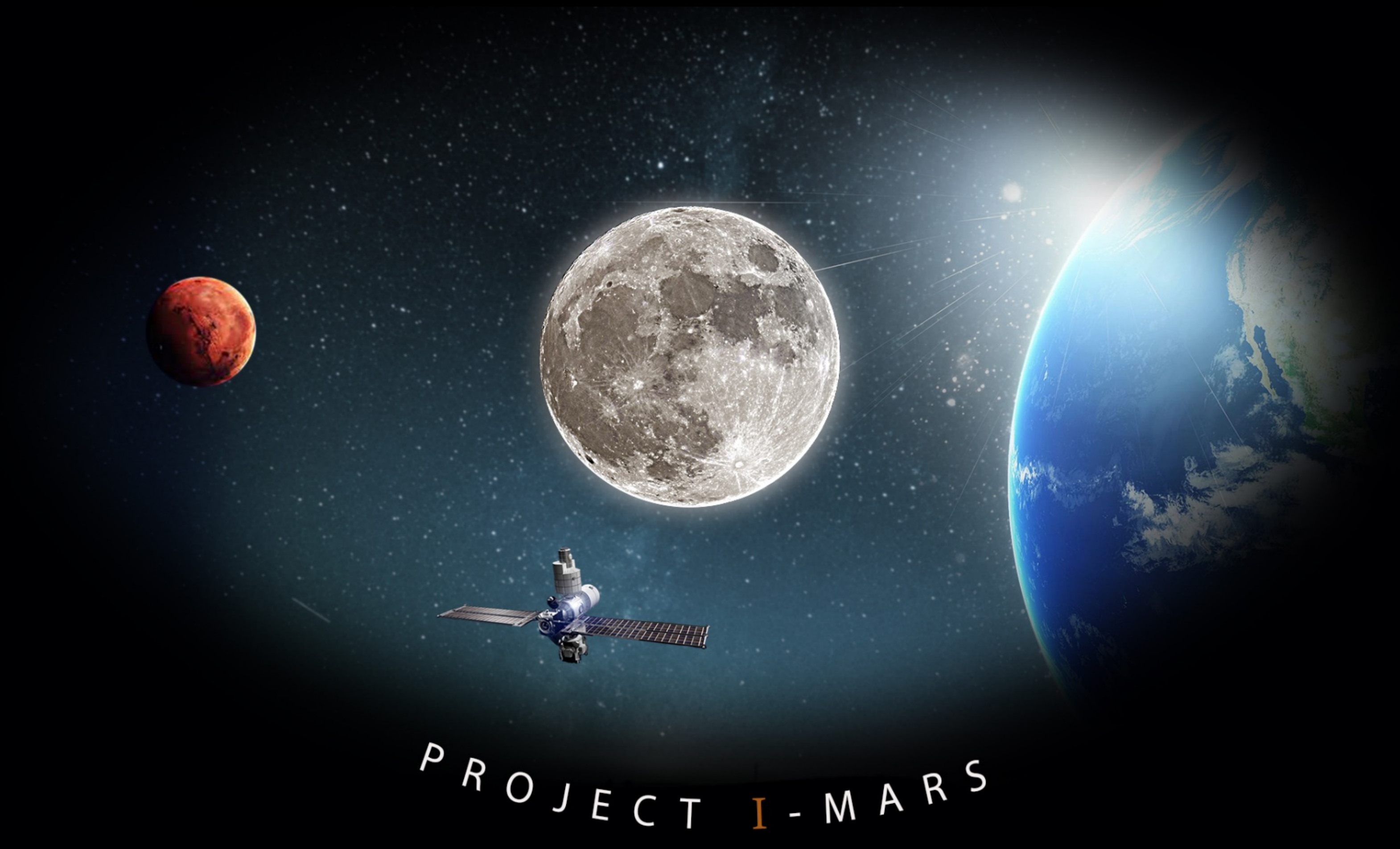
ShadX-New Horizon Presents

Project I-MARS:

Intelligent Multi-surface Access Reusable Spacecraft



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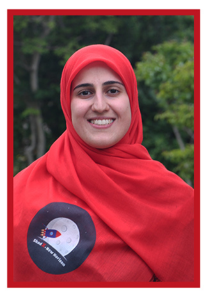

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Acronyms

ADCS Attitude Determination and Control System

AIAA American Institute of Aeronautics and Astronautics

ATA Annular Tank Arrangement

AMPS Ariana Main Propulsion System

BEAM Bigelow Expanding Activity Module

BFR Big Falcon Rocket

BOL Beginning of Life

bps bits per second

BWG Beam waveguide antenna

CAD Computer Aided Design

CDH Command and Data Handling

CFRC Carbon Fiber Reinforced Carbon

CM Crew Member

CMG Control Moment Gyro

ConOps Concept of Operation

DRA DSG Robotic Arm

DSG Deep Space Gateway

DSN Deep Space Network

ECLSS Environmental Control and Life Support System

EIRP Equivalent Isotropically Radiated Power

EOL End of Life

EPS Electrical Power System

ESA European Space Agency

EVA Extra Vehicular Activity

FEM Finite Element Method

FoV Field of View

GNC Guidance, Navigation and Control

HGA High Gain Antenna

HSRV Hercules Single-Stage Reusable Vehicle

IMU Inertial Mesearment Unit

I-MARS Intellingent Multi surface Access Reusable Surface

ISS International Space Station

JAXA Japan Aerospace Exploration Agancy

LaU Loading and Unloading

LC Life Cycle

LEO Low Earth Orbit

LGA Low Gain Antenna

LiDAR Light Detection And Ranging

LLO Low Lunar Orbit

LOC Loss of Crew

LOLA Lunar Orbiter Laser Altemeter

LOX Liquid Oxygen

MLI Multi-Layer Insulation

MMRTG Multi Mission Radioisotope Thermoelectric Generator

MOD Meteoroid and Orbital Debris

NRHO Near Rectilinear Halo Orbit

OBC Onboard Computer

ODCS Orbit Determination and Control System

PPE Power and Propulsion Element

PLF Payload Fairing

PLM payload Module

PM Propulsion Module

PSR Premanently Shaded Region

RFC Regenerative Fuel Cells

RFP Request for Proposal

RTG Radioisotope Thermoelectric Generator

RWA Reaction Wheel Control

SC Spacecraft

SDST Small Deep Space Transponder

SLS Space Launch System

SM Service Moduel

SOP State of Practice

SSPA Solid State Power Amplifier

SSR Solid State Recorder

STA Separated Tank Arrangement

TCS Thermal Control System

TLI Tans-Lunar Injection

TOF Time-of-Flight

TRL Technology Readiness Level

UN United Nation

NASA National Aeronautics and Space Administration

NA Not Available

Acknowledgments

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The Moon is the first milestone on the road to the stars.

–Arthur C.Clarke

Historical Perspective Leading to I-MARS

Deep space investigation has been a long-term vision for the scientists; as we human being do not wish to see the Earth as our permanent home. In this line of thought, NASA[1] has developed an extensive exploration road-map to help construct the infrastructure for missions beyond the Earth's orbits. As the primary footstep, ISS[2] has been playing a significant role in understanding the cislunar space and also help to conduct vital scientific researches of a better life on the planet Earth. ISS also provides a test-bed for robotic plus human missions into the deep space. Such valuable information has now enabled us to propose a new station in the vicinity of Moon as the next footstep while ISS is still operational until 2024.

We are in fact in the process of taking the necessary steps to reach the Moon's surface and building the necessary infrastructure to go beyond the Moon and deep into space. Steps that already have been experimented in small-scales inside the ISS, now find the chance to be manufactured in numbers on the Moon surface with the help of its valuable sources. Similar to that suggested by NASA, we also believe that the lunar surface shall soon serve as a crucial training ground for long human journeys deep into space; in addition to a great technology demonstration test site where it will make us prepared for cost-effective missions to the Mars. We firmly believe that the Mars Exploration would be the next habitable world created by human beings. It is, therefore critical to understand the geologic and climatic behavior of the red planet as precise as possible. We are certain that understanding how Moon and Mars function will eventually help us understand how the Earth must be saved and preserved for future generations.

We recognize that the world has experienced a lot since the first planetary mission by Apollo 11 to the Moon; the initial steps of which has its roots well earlier than 1969. However, we feel that we must make Deep Space Gateway (DSG) and its associated trips much safer and more economical to make sure that there would be no pause in space exploration activities. This is, in fact, a great opportunity for all engineers and space enthusiasts to help establish an international station on the Moon, we refer to as "Global Moon Surface Station" in the Near Rectilinear Halo Orbit (NRHO), and this proposal is our sincere effort at ShadX toward that goal. We also propose the DSG modules to be manufactured and used through international collaborations under UN leadership to include NASA and especially its western partners such as JAXA[3] and ESA. Of course, NASA as a scientific world leader would play a pivotal role with projects similar to Orion and SLS: the objective of which is critical to ensure safety and reliability of Deep Space Exploration Missions. In the current proposal, we have paid special attention to the modules that need to be inserted to NRHO. Finally, we expect Ariana to begin initial missions to the Moon surface, not later than 2028 while DSG continues its service until 2040.

[1] National Aeronautics and Space Administration
[2] International Space Station
[3] Japan Aerospace Exploration Agency

Mission Definition

NASA is willing to build an architecture, which leads to multiple lunar surface transfers to increase the feasibility of deep space exploration. Up to now, space missions have been focused on the probability of access to another planetary body such as Apollos'. After 50 years, a new horizon has been opened up to unveil the feasibility of Mars colonization. The first step to establish this goal is fulfilled by the designation of lunar surface access vehicles. Ariana[1] will be a pioneer of these vehicles by the new design philosophy, Reusability. Reusability not only allows the opportunity to construct a settlement on the Moon for further investigations but also decreases the operational cost remarkably by delivering huge rovers, and instruments to the lunar surface for more extensive scientific research.

2.1 Requirements Definition

For the mission success, primary requirements defined by Request For Proposal (RFP) illustrated in table 2.1 must be met. Also, secondary requirements implied by RFP are pointed out in each specific section.

[1]The name Ariana is also the Latinized form of the name Ariadne : "most holy"

Table 2.1: Requirement Definition

No.	Requirement	Chapter
1	The vehicle must be Reusable	All
2	The vehicle should enable the delivery of payload and/or crew to anywhere on the surface of the Moon from the DSG and back	8 9 7
3	Design of the vehicle should allow for the capability of switching between both crew or cargo modes	7
4	In crew mode, the vehicle should support a crew of 4 astronauts to Moon and return to DSG	8 9 7
5	In crew mode, the vehicle shall support the crew without resupply from surface assets for a minimum of 24 hours on the surface	13 11 12
6	In cargo mode, Ariana should be at least deliver 15 tons from DSG to moon's surface and return 10 tons from the Moon to DSG	14 9 7
7	The mission operations, including orbit transfer, station keeping, and other maneuvers necessary to deliver payload/crew to the lunar surface and back, shall be designed	8 10
8	The propulsive cost and time of flight of a range of landing sites from equatorial to polar shall be determined	9 8
9	It shall be discussed how both cargo and crew modes drive the design of the vehicle	7 8
10	The station keeping, orbital phasing and rendezvous, and proximity operation for the vehicle shall be determined	10 8 15
11	The design shall use technologies that are already demonstrated on previous programs or currently in the NASA technology development portfolio	all
12	Lifetime of each component and vehicle and number of surface trips and any potential replacement strategies that could extend vehicle's lifetime shall be determined	16.3
13	The vehicle's deployment to the DSG shall be described in detail	5 10 8 15
14	The cost for the vehicle shall not exceed $10 Billion US Dollars (in FY17)	16
15	The first operation shouldn't be later than December 31, 2028	8 16.4

Executive Summary

Knowing the fact that human presence on the Moon is a possibility with an accepted level of safety; NASA seems to be determined to pave the way for expanding human activities on other solar systems or celestial bodies. This encouraged us as one of the existing ShadX teams to be a part of this undoubtedly international endeavor. In addition, we understand planning a series of space missions in a sustainable manner calls for numerous activities with different teams with different expertise; nonetheless, we do our best to describe our vision in response to the RFP issued for the current competition.

We firmly believe that the first phase to safely execute space-missions with clear end-game must be started with a series of sound and inclusive reconnaissance missions. Once enough data is collected, we are ready to decide about the level of automation and intelligence that must be incorporated into the subsystems. The second phase would involve missions with volunteer crews ,and finally, once we are certain to have a clear understanding of the overall risks and benefits, we might declare that we as human have developed a gateway to other planets that is affordable for most countries and respects existing budget limitations. Obviously, we must be certain that people of the planet Earth are aware of how space-missions are beneficial to issues we still face on the planet majority of people live on. We understand NASA is working on a reusable cost-efficient vehicle for having access to the lunar surface resources and we have carefully studied information available to see how they could be helpful; especially the announced budget of 10 B$. This work, in fact, presents overall schematics of how I-MARS project could satisfy the NASA stipulations.

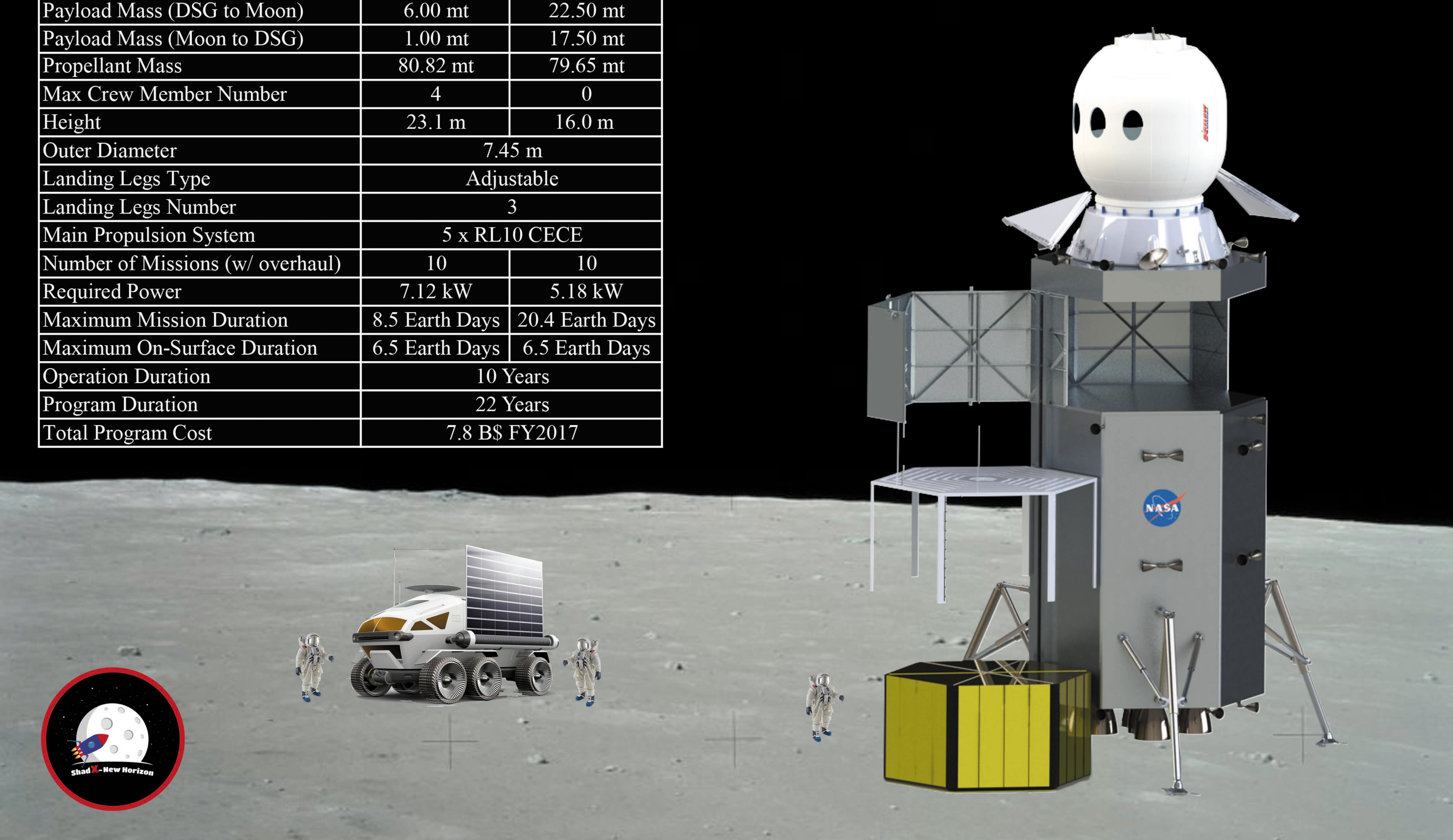
On the other hand, we believe having a Robust Life Support System is the most critical issue for any crew who desires to work on the Moon. Such issues start with biological needs we all enjoy here on the Earth. That is why; we have proposed I-Mars project to first target the areas next to either poles of the Moon; where ice layers have been detected. In this line of thought, missions to any point on the lunar surface begin from an orbital station, we refer to that as "DSG". This station is placed in suitable orbit referred to as NRHO; where propellant consumption for each mission to the Moon's surface must carefully be managed. Aiming for Jan 2025 as a starting date, we have come to the conclusion that it is quite efficient to send necessary subsystems to meet where DSG is. The minimum subsystems that need to be sent to the DSG include propulsive and habitat. Other necessary subsystems, including PLM, are sent with careful planning. Obviously, we need to limit the number of launches from the Earth to only two. This approach calls for very precise planning while assembling process goes on at the DSG. The assembly process leads to the birth of Ariana. With Ariana, we expect to conduct a minimum number of 10 missions to the lunar surface that is expandable to 20, once Ariana's PM is replaced in 2034. The habitat of Ariana is a Modified BEAM[1] that is designed to support comfortable living condition for the crew of 4 for the period of 80 Earth days

[1] Bigelow Expandable Activity Module

through non-regenerative methods. It can also carry and deliver any cargo with a maximum volume of 175 m^3 or mass of 22.5 mt to the Moon's surface. It could also return a total mass up to 17.5 mt from the lunar surface back to the DSG. Ariana enjoys an automated Loading and Unloading (LAU) mechanism with learning capability based on the harshness of the landing site on the Moon. Its propulsion system uses 5 RL10 CECE engines with both LH2 and LOX propellants that provide enough power for critical missions. The foldable ATK's Ultraflex fold-out solar panels together with batteries are especially efficient for missions to the Moon polar areas which are hardly exposed to the sunlight. In Attitude and Orbit Determination and Control Systems, R-40B thrusters are used that will be utilized in assembly and landing. We effectively use all protective measures to protect critical components considering thermal conditions on the lunar surface. Sensitive components are Cryogenic propellant tanks and batteries that are protected through passive and active methods as needed such as cryocoolers. Communication subsystem is designed in such a way to keep communication possible even while Ariana is on the far side of the moon. Communication with Earth is also available through DSG. Here we use both primary and secondary High Gain Antennas (HGA) for robust coverage. Two omnidirectional antennas are also placed on the cargo and the propulsion module as a backup. Ariana's structure is designed for many loading conditions including those during launch, maneuvers and landing on the moon surface. Composite materials and metal alloys both helped to keep the vehicle weight to its minimum with 1.3 factor of safety. And finally, Ariana endows an adjustable-clustered-tri-pod landing mechanism that allows it to safely land on areas with asymmetric slopes up to 13°. This system works in conjunction with a network of Landing-Site Surveillance Cameras (LSSC) that with processing techniques prevent Ariana to land on dangerous areas. As the final statement, we at ShadX firmly believe that mission to the Moon is, in fact, a challenging one with a sgreat many factors. Nonetheless, we have been able to cover all issues requested by the RFP and beyond in the best possible way.

	Crew Mode	Cargo Mode
Dry Mass	24.20 mt	12.39 mt
Initial Mass on DSG	105.52 mt	103.95 mt
Mass on Surface (w/o payload)	49.02 mt	37.88 mt
Final Mass on DSG	27.39 mt	15.72 mt
Payload Mass (DSG to Moon)	6.00 mt	22.50 mt
Payload Mass (Moon to DSG)	1.00 mt	17.50 mt
Propellant Mass	80.82 mt	79.65 mt
Max Crew Member Number	4	0
Height	23.1 m	16.0 m
Outer Diameter	7.45 m	
Landing Legs Type	Adjustable	
Landing Legs Number	3	
Main Propulsion System	5 x RL10 CECE	
Number of Missions (w/ overhaul)	10	10
Required Power	7.12 kW	5.18 kW
Maximum Mission Duration	8.5 Earth Days	20.4 Earth Days
Maximum On-Surface Duration	6.5 Earth Days	6.5 Earth Days
Operation Duration	10 Years	
Program Duration	22 Years	
Total Program Cost	7.8 B$ FY2017	

First Launch Date	Jan 2025
Number of Launches	2
Launch Vehicle	SLS block 1B Cargo
First Launch Mass	15.48 mt
Second Launch Mass	17.08 mt



Mission Overview

5.1 Mission Operational Architecture

Based on the mission criteria, overall configuration, modular design significance deployment ,and integration trade studies, are conducted as key design parameters discussed in the following.

5.1.1 Overall Configuration

According to Ariana's initial architecture investigation, two configurations for Ariana is considered including a single stage vehicle and a multi-stage, consisting of a lander, orbiter, and habitat [1]. The following list includes the crucial parameters affecting configuration trade study in table 5.1 .

- ΔV budget: DSG to Moon transfer maneuvers are the same in both configurations nonetheless in the multi-stage configuration, additional plane change is required as a cause of orbiter rendezvous phase with the lander.
- Reliability: Due to Crew Members' (CMs) safety provision, it is highly desirable to reduce the operational complexity of the vehicle [2].
- Communication: The presence of orbiter module in Low Lunar Orbit (LLO) improves the lander/DSG link availability.
- Landing: Regarding less mass and moment of inertia of the lander module, less torque is required to keep the vehicle on the desired powered ascent/descent trajectory.
- Control: Additional set of control systems is required for the orbiter station keeping.
- Inspection: More challenging and longer inspections and maintenance is needed in a multi-stage configuration.

Table 5.1: Overall Configuration Trade Study

Overall Configuration	ΔV	Reliability	Communication	Landing	Control	Inspection
Single-Stage	low	high	bad	hard	easy	easy
Multi-Stage (Orbiter + Lander)	high	low	good	easy	hard	hard

To accomplish the mission requirements demonstrated in section 2.1, the single-stage configuration is selected as the result of the evaluation carried out in table 5.1.

5.1.2 Modular Design Significance

Modular design is necessary for the single-stage Ariana in order to meet RFP requirements and constraints shown in table 5.2.

Table 5.2: Ariana's Design Requirements and Constraints

No.	Requirement
1	Separate crew and cargo operation.
2	Switching capability between two modes.
3	Payload and/or crew deliverability.

Ariana consists of the following modules: propulsion, cargo, service module, and habitat. As mentioned in section 5.2, vehicle integration accomplished by assembling those modules respectively. Ariana's modular design allows operating in cargo mode with payload, with the absence of habitat and service module while they are docked to DSG. In crew mode, habitat and Service Module (SM) undock from DSG and dock to cargo module to be prepared for manned operation.

5.1.3 Deployment and Integration

Ariana's modular design obliges to determine the vehicle component integration strategy. Due to the ΔV budget for mission operation, and the amount of payload delivered to the lunar surface, Ariana is defined as a heavy lunar lander. According to SLS payload fairing development, it is not feasible to deploy Ariana to DSG with a single launch on December 31, 2028. Likewise, three strategies for the integration of vehicle are considered.

- LEO assembly: In this strategy, the PM is launched to LEO and waits for the second launch to assemble and start the journey to the Moon. The habitat, SM and PLM (Payload module) are inside the second Payload Fairing (PLF) respectively.
- DSG assembly: : In this scenario, PM and SM provide required ΔV for capture to NRHO, phasing maneuver and rendezvous for docking in order to assemble on DSG.
- Orion CO-Manifested Payload (CMP): Another deploy strategy is utilizing Orion's payload delivery ability. According to the Mission Planners Guide for SLS [3], the Orion CMP deploy is not feasible due to the maximum CMP mass and PLF dimension.

In table 5.3, a trade study of the deploy strategy selection is thoroughly discussed.

Table 5.3: LEO & DSG Assembly Strategy Trade Study

LEO Assembly		DSG Assembly	
Pros	Cons	Pros	Cons
Less mission risk	Different situation in comparison with lunar environment	Less ΔV budget for rendezvous and phasing maneuver	More complexity in docking # to assemble
Easier maintenance in case of failure	Station keeping in LEO increase in debris impact probability	Less wasted upmass in terms of launch vehicles.	Required subsystems duplication for both PM and PLM with habitat
Less docking # to assemble	Decrease the engines lifetime	Test-bed provision for future deep space assembly	-

Table 5.4: ΔV Budget for LEO to NRHO Transfer

Insertion Point	ΔV(km/s) at LEO	ΔV(km/s) at NRHO	Phasing	Rendezvous	Total ΔV(km/s)
Apolune	2.641	0.804	0.250	0.010	3.705

Based on the reusability philosophy in Ariana's design, it is highly desirable to increase the number of operations. In the case of the LEO assembly, engine lifetime decreases according to the high amount of ΔV budget for LEO to NRHO transfer as shown in table 5.4. Therefore, ShadX decided to assemble the vehicle on DSG. However, the main challenge of this strategy is a number of necessary docking sequences for the final integration of the vehicle.

The docking sequence is a precise procedure thus, more docking number necessary, the more complex the system is. Ariana is capable of controlling its orientation by the Attitude and Orbit Determination and Control System (A/ODCS) and equipped with a standard docking interface in order to fulfill the docking procedure reliability. Meanwhile, the mechanical arm berths and monitors the modules attachment to DSG radial docking ports.

As a conclusion, above trade studies, conducted Ariana to be a single-stage modular lunar lander, assembled on DSG. Hence, prerequisites for Concept of Operation (ConOps) clarification are defined.

5.2 Concept of Operation

To develop a sound ConOps is very critical that would help keep the project on a homogenous path all along. We at ShadX studied the RFP carefully and realized different phases as well as stages in the overall mission. Stages of the mission are categorized into three main phases, Initiation or "pre-mission"; "Operation" and finally, "post-mission" phase that allows us to evaluate any achievement and aims to provide enhancements to the current I-MARS level of intelligence. It is noted that any stage before completion and deployment of Ariana to DSG are considered part of the "Pre-mission" phase.

Table 5.5: Stages Classification.

Phase	Stage
Pre-Mission	1, 2, 3
Operation	4, 5, 6, 7, 8, 9
Post-Mission	10

I-MARS consists of three integrated modules of: (1) PM; (2) PLM and (3) Habitat. Modules have been carefully designed to get into DSG with a minimum of two separate launches of SLS. While orbiting in DSG, modules are assembled to get to their final configuration and to start the mission operation phase to the Moon. In contrast, "Mission Operation" includes stages comprising transferring crew and cargo between DSG and the Moon. With table 5.5, the following sections provide a brief description of the stages. It is noted that stages are executed sequentially; with a Stage-Specific-Completion-Signal (SSCS) that is monitored by the I-MARS Command and Control and Health Monitoring System (ICC-HMS). ICC-HMS also has a role in examining whether any individual stage is completed as planned and whether the next stage is ready to be executed. This process is a safe-life operation with enough redundancies to prevent any catastrophic events that jeopardize the whole mission.

- **Stage 1**

Stage 1 starts with the first launch in Jan 2025. This launch is Ariana's first stage of operation. SLS would send the PM to LEO, and then, SLS upper stage would transfer its payload to TLI, and finally, the PM uses its main engines to capture NRHO and then docks to DSG through suitable maneuvers. Once the module is docked, an intelligent mechanism, DSG Robotic Arm (DRA), examines all critical items such as engines, thrusters, pumps, pipes, and tanks to make sure they are all ready for the next mission. Stage 1, once completed, sends its specific signal to ICCS.

- **Stage 2**

Stage 2, in contrast, aims to take the habitat and the cargo modules of Ariana to the TLI. Once completed, the SM utilizes its own engines to capture into NRHO. The PLM would then be docked to the next available docking port of DSG. It is noted that the habitat module is finally placed inside the PLM. As an integrated part of any assembly operation, all critical elements are examined to make sure they function properly. The provided omnidirectional antenna capabilities are essential to complete stages 1 and 2.

- **Stage 3**

Stage 3 has a role to complete Ariana's assembly. In this stage, we use two active modules of (1) the PM and (2) the SM. After both modules successfully dock to DSG ports, PM docks to PLM, while SM separates from PLM and places in the vicinity of DSG. In this phase, separation of the habitat module and its installation are completed with the help of the DSG's DRA. Afterward, SM docks to habitat; then, the final dock completes the sequence.

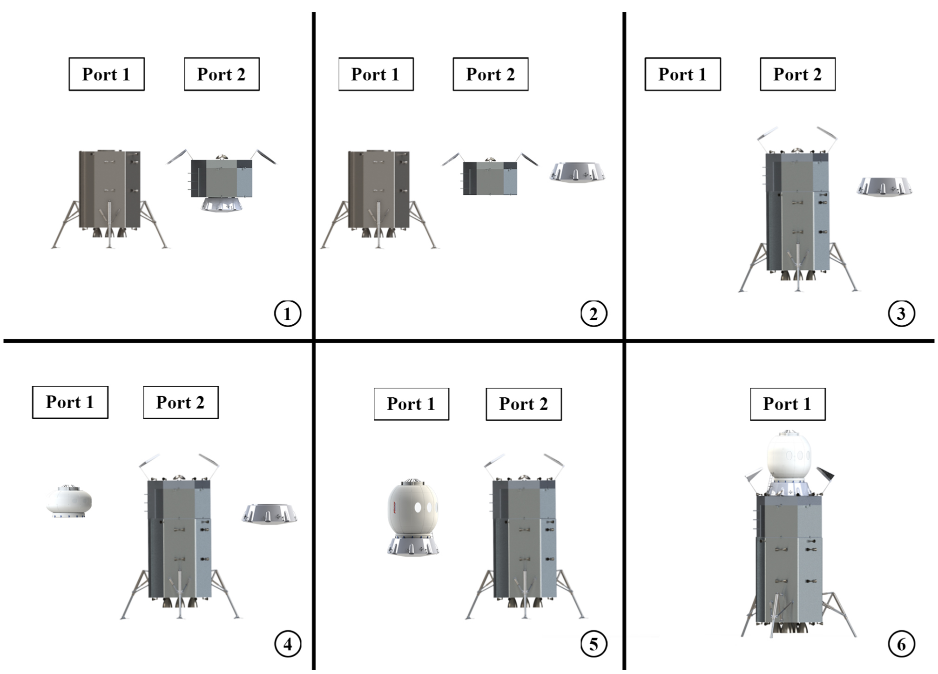


Figure 5.1: Docking Sequence

- **Stage 4**

Stage 4 starts with fueling the tanks when it's docked to DSG. The mission could be executed for either crew or cargo. In the cargo mode, the habitat module remains on DSG while Ariana starts its trajectory to LLO. However, in the crew mode, all modules would be transferred to LLO. Also, this configuration allows Ariana to deliver payload in crew mode.

- **Stage 5**

Stage 5 Ariana performs the capture burn into targeting LLO. After that, a Deorbit insertion maneuver puts Ariana on a low perilune altitude orbit, where the powered descent phase initiates. This phase has been designed to last about 625.00 seconds that lands from 150 km circular orbit on the lunar surface.

- **Stage 6**

Stage 6 begins once Ariana reaches to its stable position on the lunar surface. Such stability is examined by ICC-HMS to make sure all crew members could exit the habitat by available means. During this stage, the crew are able to have two Extravehicular Activities (EVAs) each could last 8 hours. Furthermore, for unloading cargo, we use a specifically designed Cargo-Elevating-System that is also used to lift and position cargo back into the PLM.

- **Stage 7,8**

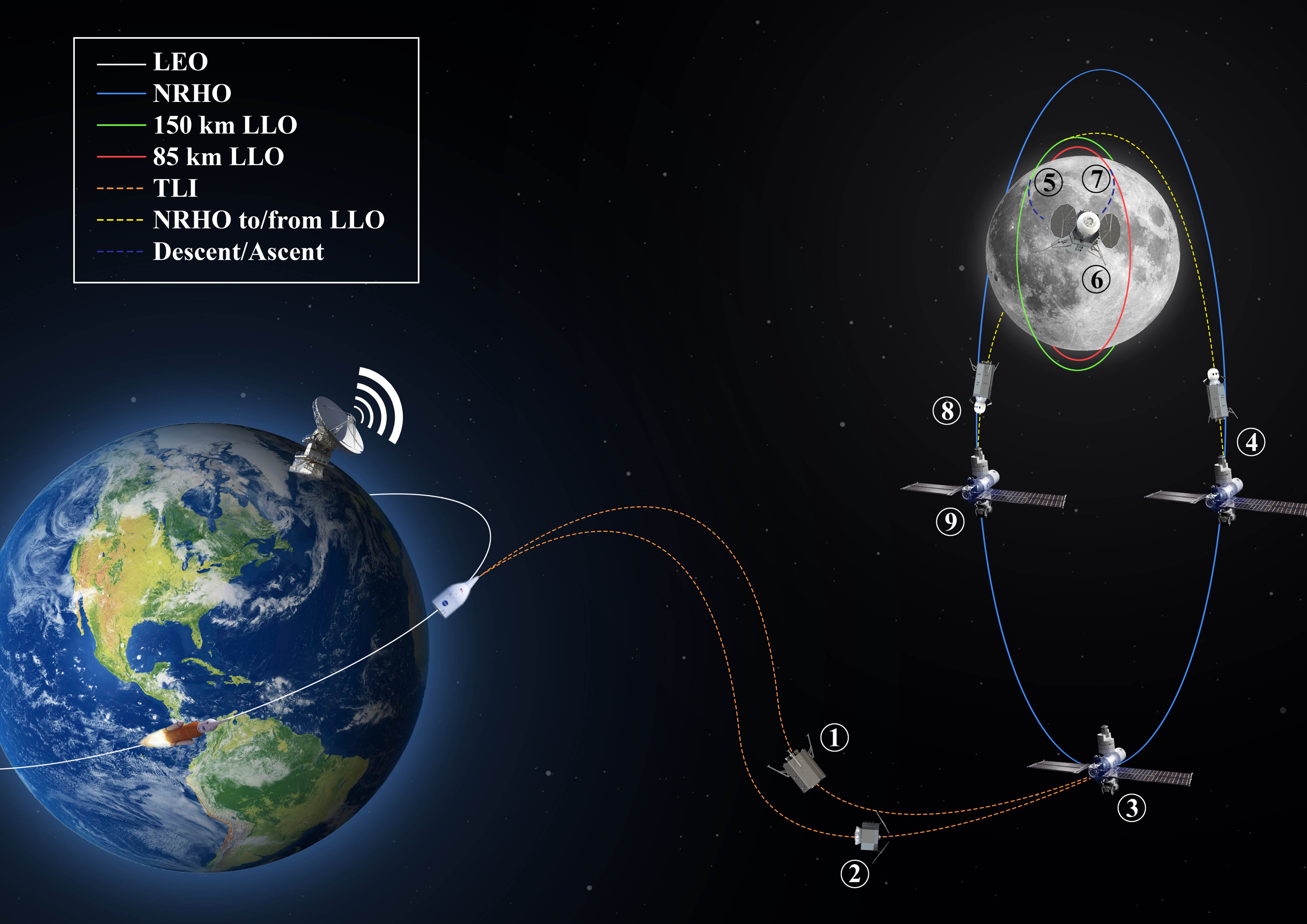
Once operation on the lunar surface is completed, and crew are ready to return to the DSG. During stage 7 Ariana begins the ascent burn. Transfer strategies from the lunar surface to LLO and afterward from LLO to NRHO are similar to stage 4 and 5.

- **Stage 9**

Once Ariana captures into NRHO, she needs to dock back to the DSG; this is designated as Stage 9.

- **Stage 10**

During this stage, as the post mission phase, ICC-HMS examines all data collected during the just completed mission. Proper data-mining allows I-MARS to have a better understanding of its level of intelligence and conduct its upcoming mission with more safety and enhancement. It is further emphasized that during all stages, Ariana keeps its communication either with Earth station or DSG.


LEO
NRHO
150 km LLO
85 km LLO
TLI
NRHO to/from LLO
Descent/Ascent
1
2
3
4
5
6
7
8
9

Science of The Moon

In this chapter, the analysis and data required for the landing phase and the lunar operation are discussed. The important factors such as general physical data, topographical map of the Moon, temperature, slope, hardness, and reflexivity of the surface are analyzed. Also, trade studies for possible and rich points of the Moon are carried out further, in order to locate suitable landing sites with respect to their benefits and reaching difficulties.

6.1 Main Physical Characteristics of the Moon

In table 6.1, the main physical characteristics of the Moon and the Earth are compared. Beside sensible physical properties such as mass and radius, considering the number of molecules in each cm^3 of the Moon atmosphere, it can be easily understood that the moon has no atmosphere at all. This is an important fact for the design process whereas the vehicle will not encounter heat interaction as it enters the atmosphere unlike what happens with re-entry vehicles to the Earth's atmosphere. Another consequence of lacking the atmosphere is the intense alteration of the lunar surface temperature. Moreover, the magnetic field magnitude of the Moon is also negligible with respect to the Earth's.

Table 6.1: Physical Properties of the Moon and Earth [4]

Property	Moon	Earth
Mass	$7.353 \times 10^{22}(kg)$	$5.976 \times 10^{24}(kg)$
Mean Radius	1738 (km)	6371 (km)
Mean gravity	$1.62(m/sec^2)$	$9.81(m/sec^{)}2$
Mean surface temperature	107°C day; -153°C night	22°C
Atmosphere	$10^4(molecules/cm^3)$	$2.5 \times 10^{19}(molecules/cm^3)$
Magnetic field	0 (small paleofield)	24-56 (A/m)
Albedo	0.073(subsolar peak)	0.3

6.2 Environmental Maps Analysis of the Lunar Surface

Further, in this section topographical and thermal map, alongside slope and roughness of the surfaces are analyzed thanks to measurements of the Lunar Orbiter Laser Altimeter (LOLA) [5] . This database is the result of the Lunar Reconnaissance Orbiter (LRO) over 6.3 billion measurements of surface height, with a vertical precision of ∼10 cm and an accuracy of ∼1 m [6]. Figure 6.1 visualizes the data mapped on a lunar surface.

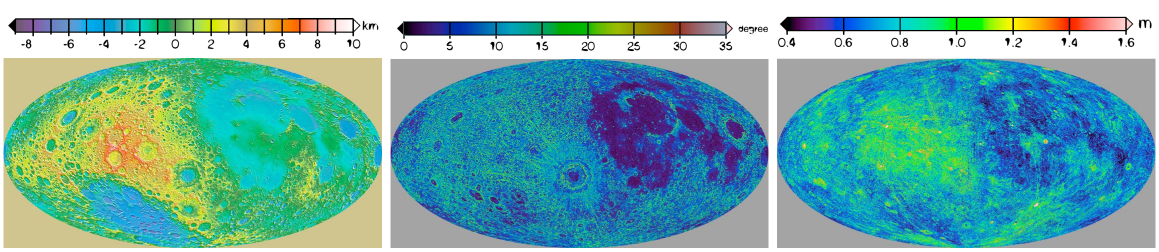


Figure 6.1: Topography, Slope and Roughness of the Lunar Surface [7] [8] [9]

6.2.1 Topography, Slope and Hardness

LOLA provides an accurate global lunar topographic and geodetic map that is the basic foundation in order to understand the surface behavior. This helps future missions for safe landings and enhances mobility on the Moon by providing topographical data. Moreover, slope and roughness analysis are crucial as a cause of Ariana's landing leg performance during the descent phase. These factors will eventually lead to the choice of suitable landing sites for the mission operation.

6.2.2 Thermal Map

As the moon rotates around the Earth in its orbit, different parts of the lunar surface are lightened by the sun, causing the lunar surface, a significant thermal change due to lack of atmosphere in the Moon. Areas affected by the Sun can reach temperatures up to 100 $^{\circ}C$ while areas in shadow can reach hundreds of degrees below zero $^{\circ}C$. Table 6.2 shows the mean estimated lunar temperatures in different regions on the surface. Meanwhile, figure 6.2 plots the thermal contour of the lunar surface versus latitude and longitude.

Table 6.2: Thermal Data of Lunar Surface [10]

	Shadowed Polar Craters	**Other Polar Areas**	**Front Equatorial**	**Back Equatorial**	**Limb Equatorial**	**Typical Mid-Latitudes**
Average Temp.	40 K	220 K	254 K	254 K	256 K	220<T<255 K
Monthly Range	None	±10 K	±140 K	±140 K	±140 K	±110 K

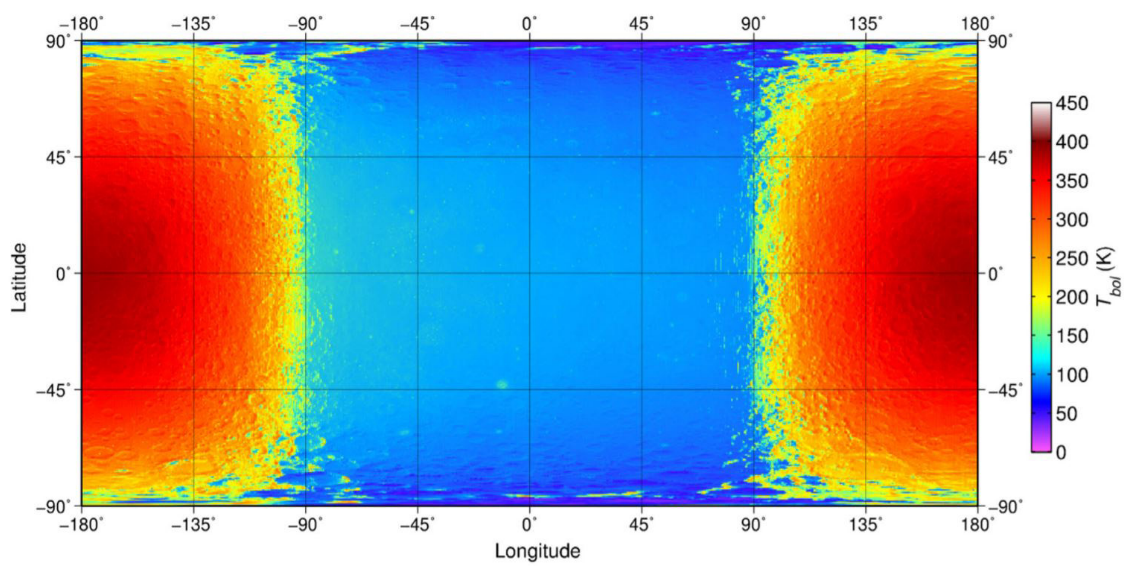


Figure 6.2: Lunar Thermal Map [10]

6.3 Landing Sites Analysis

The National Research Council and Lunar Exploration Analysis Group studies, identified scientific research that should be addressed by future lunar exploration. Furthermore, a series of technology priorities have been identified that would enhance operations at all places on the lunar surface. Some of the main concepts and technology enhancements were defined as follows [11]:

The Lunar Water Cycle: There are three main components of the water cycle which are discovered in the last decade: interior water, surficial water, and polar water. The polar water predominantly resides in some Permanently Shaded Regions (PSRs) on the floors and walls of impact craters.

The Origin of the Moon: Information and details are kept in the lunar rock record. Concentrated sample studies are used to unveil theses unknown origin hypotheses.

Lunar Tectonism and Seismicity: The internal construction, thermal review and, structure of heat loss are linked to the resulting spreading of surface tectonism.

Communication Relay: Currently, by installing a communication relay system, consisting of one or more spacecraft, high priority targets on the lunar farside will be open to exploration. Table 6.3 highlights those surface targets that would be enabled by the ability to communicate with surface assets on the lunar farside.

Surviving the Lunar Night and PSRs: Developing systems that can operate in low temperatures and large temperature variations greatly expands the possible mission zones on lunar surfaces such as lunar poles' PSRs.

Cryogenic Sampling, Transportation, Storage, and Analysis: When analyzing samples, keeping the sample's quality as its original state during the transition phase to analysis is at high demand. Systems capable of fulfilling requirements for sustaining samples under lunar cryogenic conditions are very valuable. Without this protection, sample properties and compositions can alter, ultimately affecting the quality sample.

Automated Hazard Avoidance: Since many possible hazards on the lunar surface are smaller than the highest resolution imagery available, developing and integrating automated hazard avoidance system as part of a spacecraft's guidance, navigation, and control systems are crucial to guarantee safe landings on the Moon. Likewise, Light Detection and Ranging (LiDAR) sensors are utilized to detect impact craters or rocks on the surface. LiDAR generated images clearly indicates the spacecraft orientation during this hovering stage.

Mobility: The lunar rovers are utilized for traveling further distances, collecting a more diverse lunar sample, and gaining a better understanding of the lunar topography.

Dust Mitigation: Reliability progress is necessary for regolith removal from the crew in order to increase the life cycle of the systems and safety enhancements.

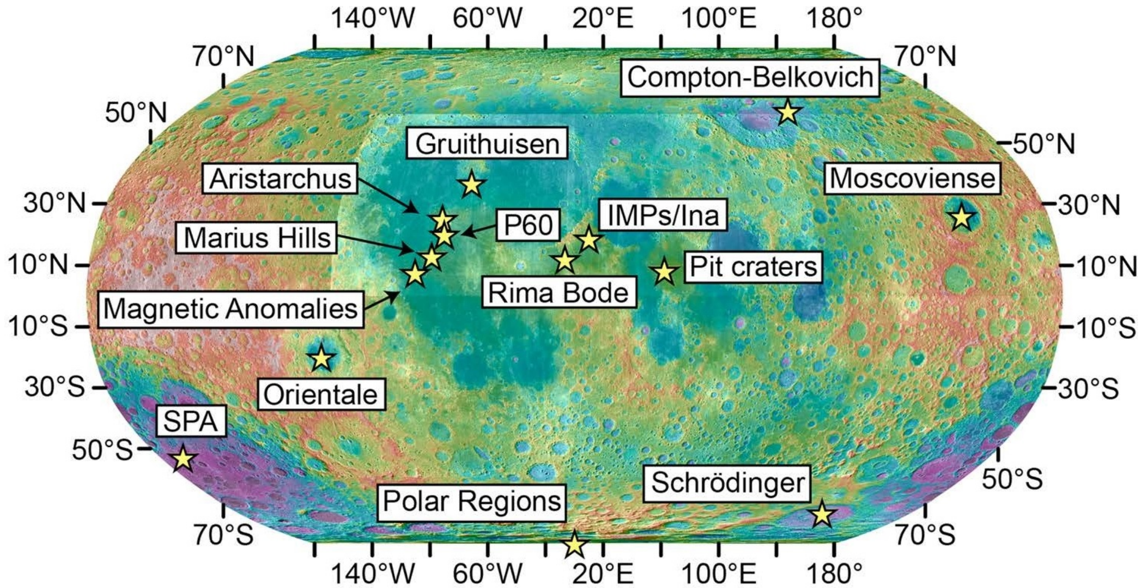


Figure 6.3: High-Priority Landing Sites (with LOLA-shaded topography) [11]

Table 6.3: Potential Landing Sites Analysis [11]

	Overarching Themes										Enhancing Technologies					
Landing Sites	Bombardment	Interior	Crust	Volatile Flux	Volcanism	Impacts	Regolith	Water Cycle	Origins	Tectonics	Communications Relay	Night Survival	Cryogenic Sampling	Automated Hazard Avoidance	Mobility	Dust Mitigation
Aristarchus	×	×	×		×	×		×				×		×	×	×
Compton Belkovich		×	×		×			×			×	×		×	×	×
Gruithuisen Domes		×	×		×			×				×		×	×	×
Ina/IMPs	×	×			×	×		×				×		×	×	×
Magnetic Anomalies		×	×	×		×	×	×			×	×		×	×	×
Marius Hills	×				×			×				×		×	×	×
Moscoviense	×	×	×		×	×	×	×		×	×	×		×	×	×
Orientale	×	×	×		×	×	×	×		×	×	×		×	×	×
P60 Basalt	×				×			×				×		×	×	×
Pit Craters	×				×		×	×			×	×		×	×	
Polar Regions				×				×			×	×	×	×	×	×
Rima Bode		×			×			×				×		×	×	×
Schrödinger	×	×	×		×	×		×		×	×	×		×	×	×
SPA	×	×	×		×	×					×	×		×	×	×
Network of Nodes - Geophysics		×								×	×	×		×		×
Network of Nodes - Exosphere				×			×	×			×	×		×		×
Basin Chronology	×					×					×	×		×	×	

Configuration Study & Development

The configuration design must be involved in detail with every other subsystem of the spacecraft. The configuration design is a compromise among different requirements that some of them are in conflict. In a way that there is not a unique solution for design problems, therefore, the concentration is on finding the best possible solution. In this section, to choose the best possible configuration for Ariana, important requirements have been considered. After contemplating the advantages and disadvantages of each one, the hinged-cargo door configuration is selected for Ariana. Based on the RFP, the effects of the primary requirements on design configuration leads to the final decision is in table 7.1.

Table 7.1: Requirement Effect on Configuration Design

RFP Requirement	Ariana Configuration Design
Landing anywhere	The S/C needs an adjustment system which includes adjustable legs and surface monitoring cameras/sensors
Delivery of payload and/or crew	Which makes modular design essential to be able to switch between these two modes.
The payload capacity of at least $15mt$	Cargo storage dimensions should be size and able to be fully autonomous self-unloading and loading.
Multiple trips to the lunar surface	Vehicle systems should be reusable and the ability to refuel at DSG

Moreover, it is not required to provide cooling, heating, and power for the payload and the payload should be self-dependent. The volume that should be considered for a 22.5 mt payload is assumed to be about 150 m^3, based on a study of the past missions' and RFP keyword about what would be delivered to the Moon's surface or DSG.

Furthermore, the launch vehicle put some constraints on the design of the configuration, including launch weight, available volume, and launch costs. With these system-level requirements and launch vehicle constraints, the process of studying existing concepts and design of the vehicle configuration starts. The design drivers are the mission objectives, e.g., multiple landings, rendezvous, dockings, orbital maneuvers, and also the environmental situation including solar radiation and thermal conditions. So, the vehicle configuration should satisfy all of them. Moreover, there are also other factors such as antennas, sensors' arrangement, payload protection, maintenance requirements, and the matter of reusability.

The study of existing concepts shows the difficulties in the design of a self-unloader vehicle. The Hercules Single-Stage Reusable Vehicle (HSRV) [12] has an aerodynamic shape based on its Mars mission's requirements. For unloading the payloads, it is necessary to have an expendable vehicle to deploy a mobile lift scissor-jack rover. Also, HSRV cannot deliver both cargo and crew to the lunar surface. Finally, the landing legs are not adjustable and cannot land on uneven surfaces.

The Lockheed Martin Lunar Lander [13] has low payload delivery (about 2000lb) to the lunar surface, and it has not a modular design. This concept is not able to return any payload from the Moon's surface, and the location of payload in this concept is near the engines' nozzle which has the risk of penetration and damage.

In this condition, using an orbiter is one of the alternatives. However, as mentioned in section 5.1, the single-stage lander configuration is superior to multi-stage (orbiter+lander).

7.1 Ariana Configuration selection

Briefly, the process of selecting the configuration consists of examination each concept and scoring them in a decision matrix of table 7.2, as well as for choosing the best one. These concepts are External-cargo, Ramp-type LaU, Cargo-mid-propulsion, Slide-out cargo, and Hinged cargo door.

Table 7.2: Configuration Trade Study Chart

Parameter	**Impact Factor**	**External-cargo**	**Ramp-type LaU**	**Cargo-mid-propulsion**	**Slide-out Cargo**	**Hinged Cargo Door**
Single Payload Pallet	0.1	0	1	1	1	1
Protect Payload	0.15	0	2	1	4	4
Stability on Surface	0.1	1	4	5	3	3
Reliability	0.25	3	2	2	1	3
Weight	0.15	3	3	4	2	3
Saftey	0.25	1	2	1	4	4
Total	-	**1.55**	**2.25**	**2.1**	**2.55**	**3.2**

The scoring on table 7.2 is from 0 to 5 (higher better); and for the parameter "single payload pallet"[1], is 0 or 1. The reliability is based on the number of subsystems and the number of actions that needs to complete the LaU process.

[1]It means the vehicle could carry a single piece 22.5 mt payload instead of multi-piece payload

External-cargo: In this concept, figure 7.1, the cargo pallets are outside of the vehicle. The symmetry in the LaU process and low height helps the stability of the vehicle on the moon surface. Cargo is not protected against any impact of meteors; cargo sections should be in 2 to 4 separate parts with approximately similar mass. It is also necessary to have more than one (2 to 4) LaU mechanisms, which are the disadvantages of this concept.

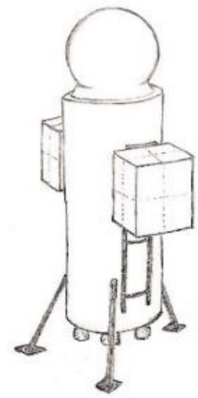


Figure 7.1: External-cargo

Ramp-type LaU: As it is shown in figure 7.2, the Ramp should be consisted of at least two rails that connect the PLM to the Moon's surface. Due to the calculation of the propellant amount needed for the mission (chapter 9) and the constraint of the launch vehicle on the maximum outer diameter of the vehicle (section 8.3.1), the height of the PM will be at least about 10 m. Hence, to have a smooth LaU, it is necessary to have a maximum slope of 30° (based on the steepest slope that Spirit and Opportunity Mars Exploration Rovers (MER) [14] and Chang'e-3 Moon Lander [15] designed to). Accordingly, it needs approximately 20 m rails.

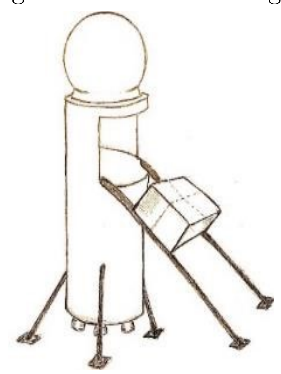


Figure 7.2: Ramp-type LaU

Cargo-mid-propulsion: In this concept, figure 7.3, the cargo is inside of the vehicle between propellant tanks. This configuration causes a reduction in the height of the vehicle and makes the stability better. Furthermore, it has a simple LaU process. Besides having its advantages, this concept is big on the diameter, and the engines are far from each other that makes the propulsion not to be a fail-safe system.

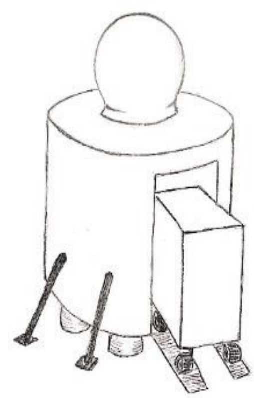


Figure 7.3: Cargo-mid-propulsion

Slide-out cargo: Slide-out cargo: This tandem concept consists of the PM, PLM, and habitat module which continuously mounted after each other. The cargo door opens, the payload will be got outside by two telescopic cylinder actuators. Then the door closes and in the meantime, the excrescence (the green piece in figure 7.4) will be captured by the rail holes that are on the edge of the doors (the red pieces in figure 7.4) and also this integrated rail would be aligned with the rail that is on the outer surface of the PM.

After that, the payload would be brought down through this rail by two electromotors at the end of the telescopic-cylinder actuator. The whole process of unloading is shown in figure 7.5.

The matter of moving payload outside of the vehicle will move the location of Center of Gravity (CG) and will affect the vehicle's stability on the surface. Moreover, the high number of actions required to complete the LaU process causes the system to have more complexity.

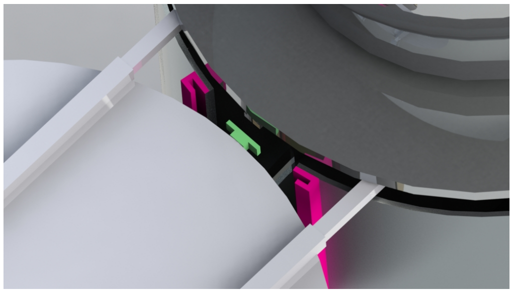


Figure 7.4: LaU Rail Mechanism

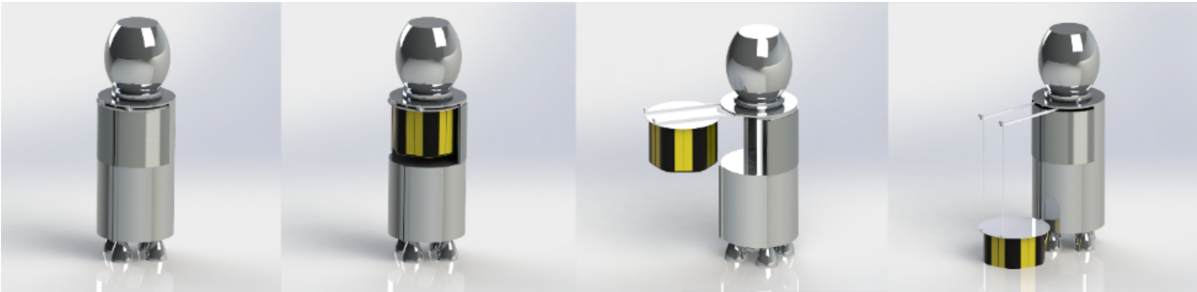


Figure 7.5: LaU Strategy of Slide-out Configuration

Hinged Cargo Door: The overall configuration is analogous to Slide-out cargo, but the strategy of LaU is different. In this concept, there is a cargo door that rotates about a hinge by rotational actuators. After that, the door is opened; the two rails one inside the cargo door (the red one in the figure 7.10) and one outside the PM will be aligned, and the cargo plate can move vertically through this integrated rail by two electromotors that are connected to the plate by cables. The whole process of unloading is shown in figure 7.7. The effect of stability on the surface is like the previous configuration, and also the hinged door causes the structure to be heavier which can tolerate the tensions.

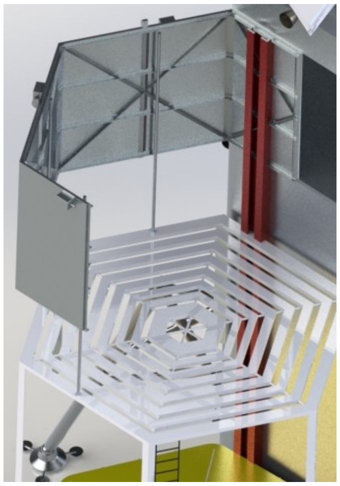


Figure 7.6: Aligned rails of LaU System

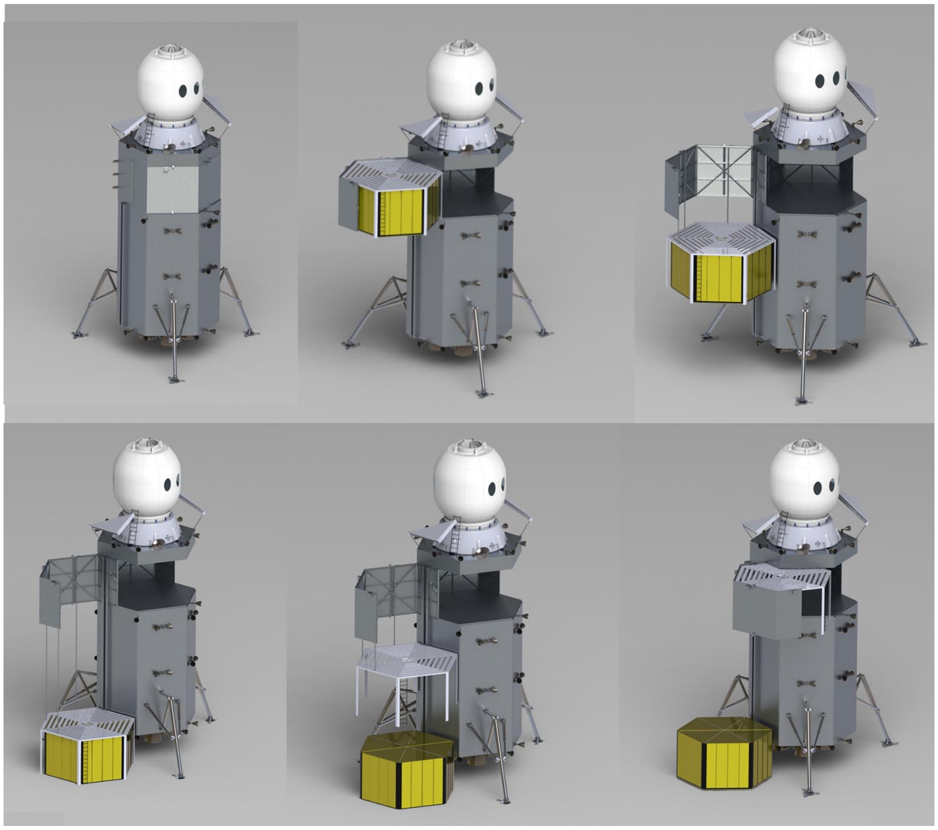


Figure 7.8: LaU Procedure

7.1.1 Conclusion

To select an adequate configuration for the lunar lander mission, ShadX study about all feasible configuration and compare them in a decision matrix in table 7.2, and select the "Hinged cargo" configuration for the vehicle, based on the matter of safety, reliability and cargo protection.

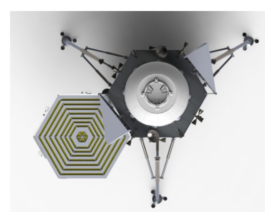


Figure 7.7: LaU Rail Mechanism

7.2 Detail Design

7.2.1 Landing

The enormous challenge of landing on the Moon's surface is to avoid high slope locations and obstacles. As mentioned in (chapter 6), the local data of the lunar surface with a maximum resolution of 50 meters is available. Accordingly, in the landing site selection process, the matter of landing feasibility and risks should be considered.

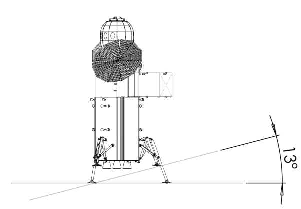


Figure 7.9: View of the Vehicle on the Sloped Surface

The issue of landing on sloped surfaces can be handled by scanning the surface with cameras and control the landing legs to be on their position. On the sloped surfaces, the vehicle will be landed in an orientation opposite to the direction of the opened cargo door; this will help the stability of the vehicle on the moon surface.

7.2.2 Bearings

The conventional bearings used in the Earth cannot be applicable in a vacuum because of the evaporation of the lubricants in high temperature which are designed to work at low temperature. Therefore, the best way is to utilize self-lubricating and low friction materials such as Teflon which is able to operate over an extremely wide temperature range ($-330°$F to $360°$F continuous) and in about zero pressure [16].

7.2.3 Electric Actuators

For running the mechanisms, the vehicle needs actuators consisting of electromotors and rotary actuators. Electromotors should have high reliability and low weight with high torque and moderate rpm. These properties lead us to select the brushless D.C. electromotors specially designed for space applications. Further, one option for rpm reduction is a worm gear mechanism that is utilized in High Gear Ratio Rotary Incremental Actuator [17], This actuator is made up of two key elements including a motor and a multi-stage speed reducer with an output step angle of $0.0030°$ as well as the reduction ratio of 6400:1.

7.2.4 Docking and Connections

The interferences between each module are permanent, semi-permanent, and temporary docking ports. The docking ports are identical with the existing concepts that are described in the international docking system standard [18]. The interference between each module is in table 7.3

Table 7.3: Docking between Modules

	Habitat-DSG	**Habitat-SM**	**SM-Cargo Module**	**Cargo Module-DSG**	**Cargo Module-Propulsion Module**
Docking Type	Temporary	Permanent	Temporary	Temporary	Semi-Permanent

Also, to utilize the DRA of DSG, four number of attaching points are prepared on the outer surface of the vehicle.

7.2.5 Loading and Unloading

Based on the massive amount of payload needed to be transferred to the surface in order to be able to safely unload it and also be able to load at least 10 mt to the vehicle; the matter of LaU plays a crucial role in configuration design. As was described in the cargo needs to be unloaded from about 15.5 m height to the moon surface. The LaU mechanism is based on a rail and two electromotors (and another one as redundancy) that are connected to a reinforced plate, and the cargo pallet is connected to this plate. The sequence of LaU is shown in figure 7.10.

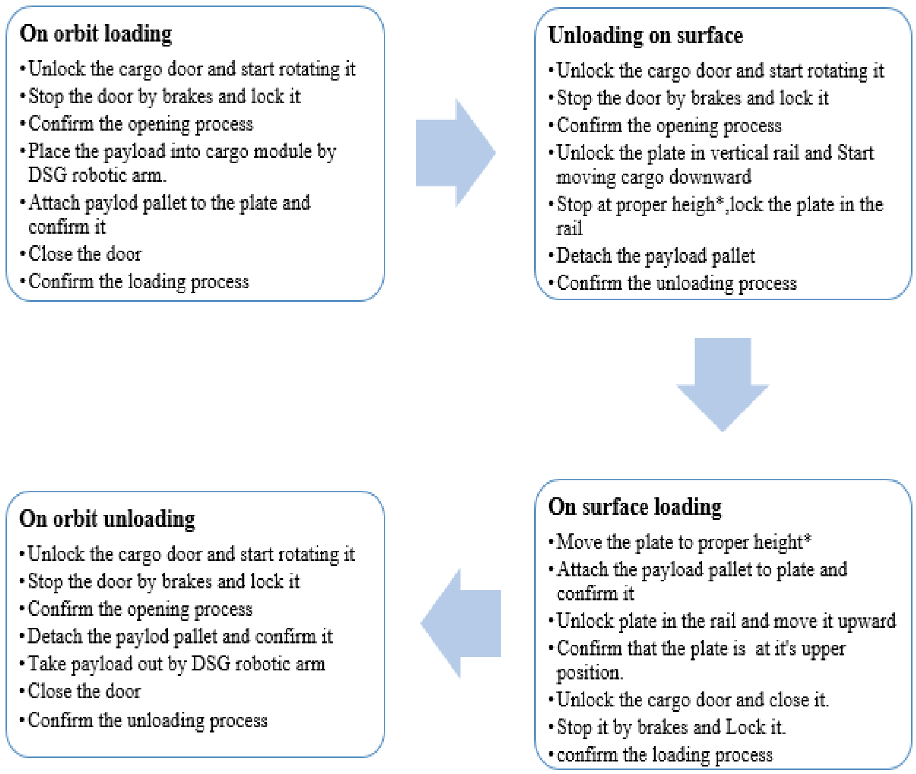


Figure 7.10: Loading and Unloading Mechanisms

∗ The proper height is calculated by determining the surface slope, landing legs position, and the cargo pallet length. Besides, at this height, the payload pallet will be touching the surface.

There are some micro-mechanisms in the LaU system that is introduced:

1. Locks: A lock is needed in each part of the sequence that needs to be adjusted in an exact position.

2. Brakes: Brakes are needed in each part of the sequence that needs to be stopped from the previous motion.

3. Sensors: At the end of every motion and attachment process, sensors like SSCS will confirm the process completeness.

4. DRA: The on-orbit LaU process is dependent on this arm.

5. Payload Pallet: This is inspired by the unit load devices in the aviation world. These pallets can be made from metal sheets or polyester rope nets to keep the cargo fixed to the upper plate of the PLM.

6. The vertical rail is mounted on the inside of the PLM door and also on the external surface of the propulsion module. When the door is opened, these two vertical rails will be aligned together and make the LaU process possible.

The drivers of the mechanisms are electric actuators. For rotating the cargo door, there are two rotational electric actuators (The opening process can be completed even with one of those) (section 7.2.3). Moreover, for vertical movement of cargo, there are three electromotors (one for redundancy). The LaU process is shown in figure 7.10.

Alternatives

In the case of main mechanisms failure, or the door hinge is locked, these two alternative mechanisms will be activated to open/close the door.

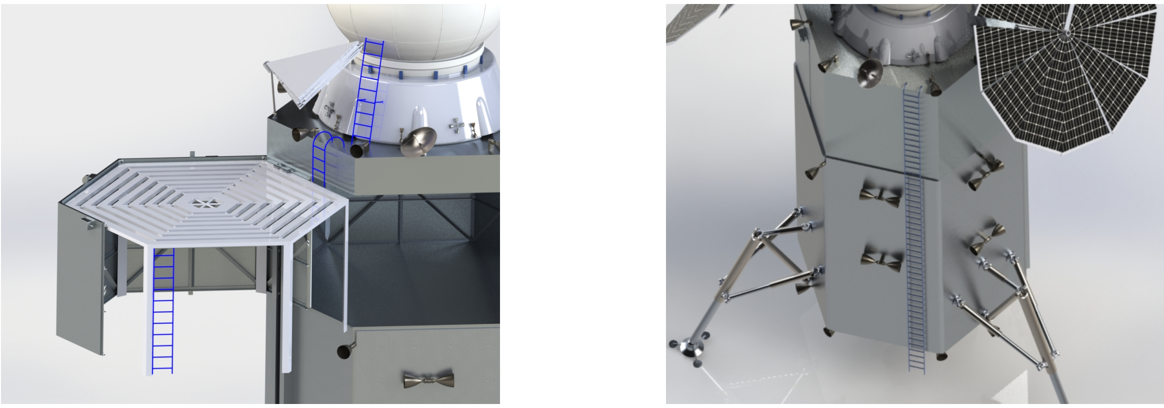
1. Thruster: There is a cold gas thruster on the external surface of the cargo door in the opposite side of the hinge, and if something goes wrong, this thruster opens the door as shown in figure 7.11.

2. Pre compressed springs: Two pre compressed springs will release and make a shock on the cargo door to make it rotate.

Figure 7.11: Alternative Thrusters

CM Get Off/On

In order to take each CM to the surface, the cargo unloader plate is being utilized. Three space ladders are on the vehicle, and one of them is under the habitat airlock to take them on top of the PLM. Another one is on the side of the vehicle to take them on the cargo plate, and the last one is on the side of the cargo pallet to get off the vehicle to the Moon's surface. Also, there is another long union ladder from the top of the PLM to the Moon's surface for using in an emergency with safety tethers.



(a) CM Ladders

(b) Emergency Ladder

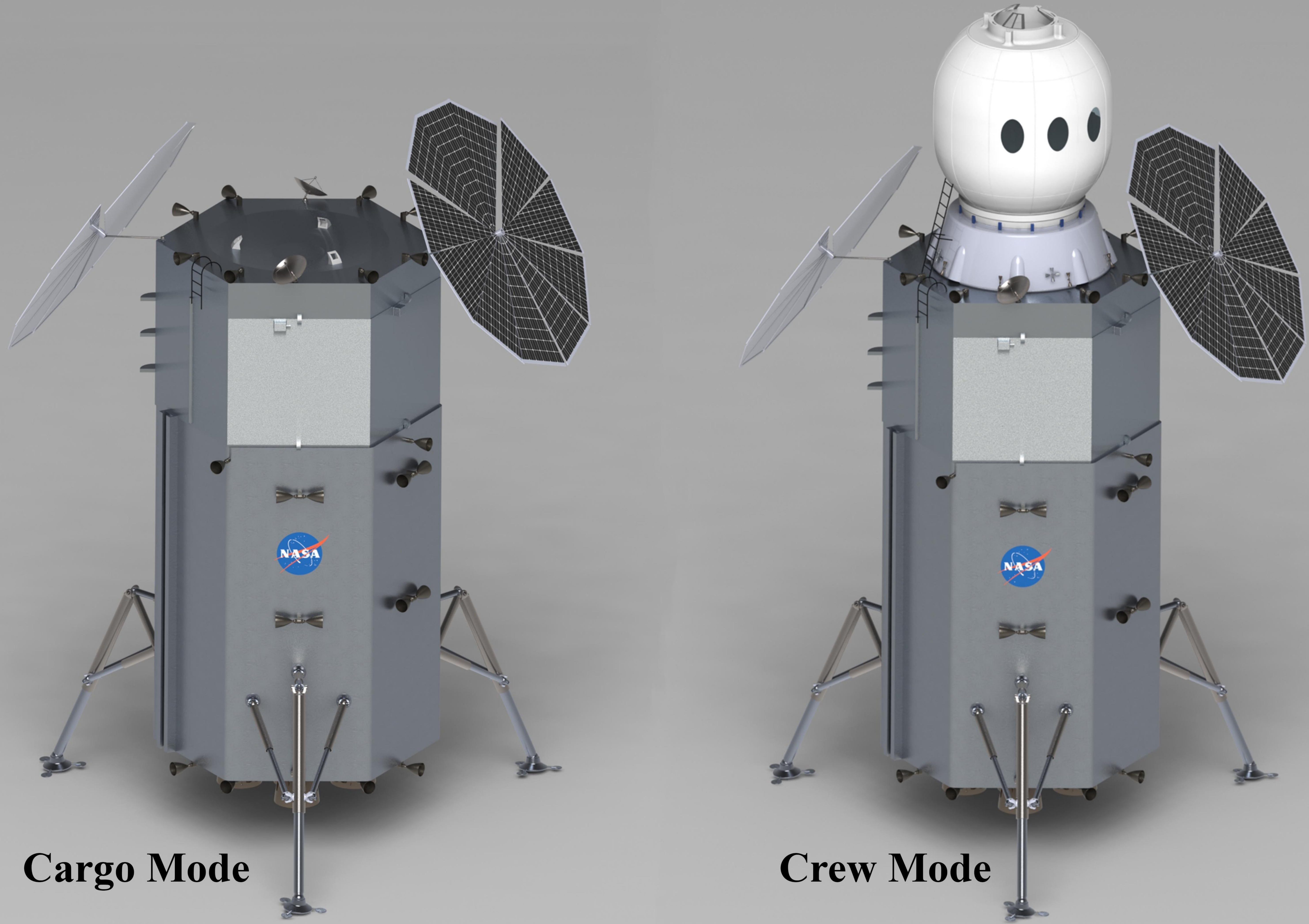
Figure 7.12: Ariana's Ladder

7.2.6 Landing Legs and Adjustment mechanism

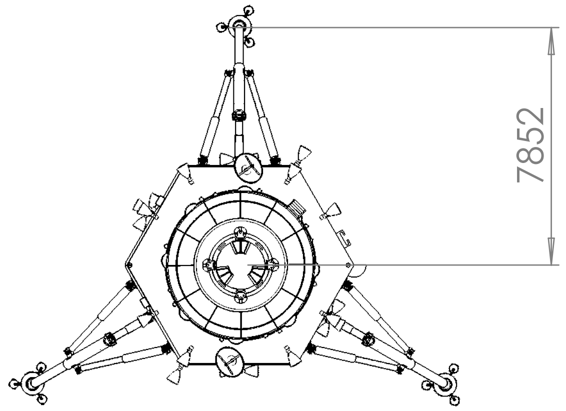
Three aluminum alloy landing legs which are attached to the main structure will support the vehicle in the landing phase. Each of them has three linear cluster-actuators and four attachment point to the main supporting structure. During touchdown, the spacecraft will have both vertical and horizontal velocities. The vertical impact will be handled by a helium gas shock absorber, and the horizontal impact will be tolerated by lateral actuators. Also, this landing leg configuration makes it fully adjustable for different landing situations and enables it for landing in maximum 13° slopes, as shown in figure 7.9, the landing legs are retractable to minimize ground clearance during LaU.

7.2.7 Refueling

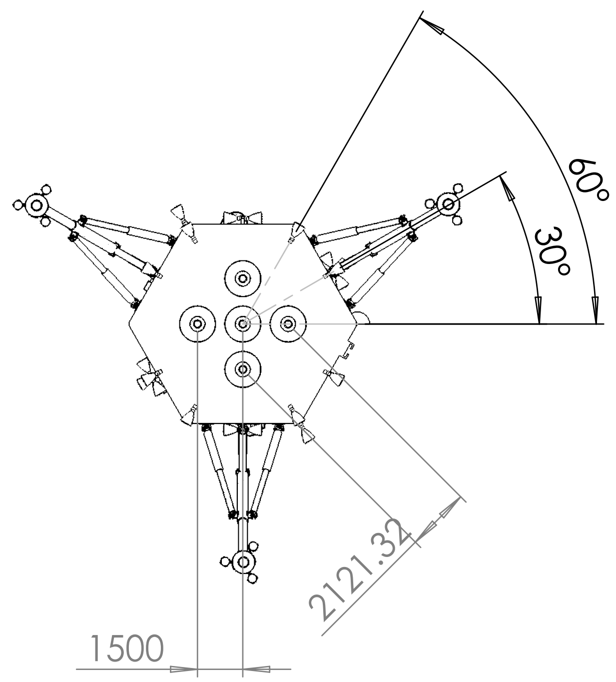
One of the challenges of reusability is the refueling in orbit. Based on the NASA study on refueling of cryogenic fluid mechanism in space under the project of Robotic Refueling Mission (RRM) [19], the process of refueling is being done by the Fluid Transfer Module on the refueling missions.


Cargo Mode
Crew Mode

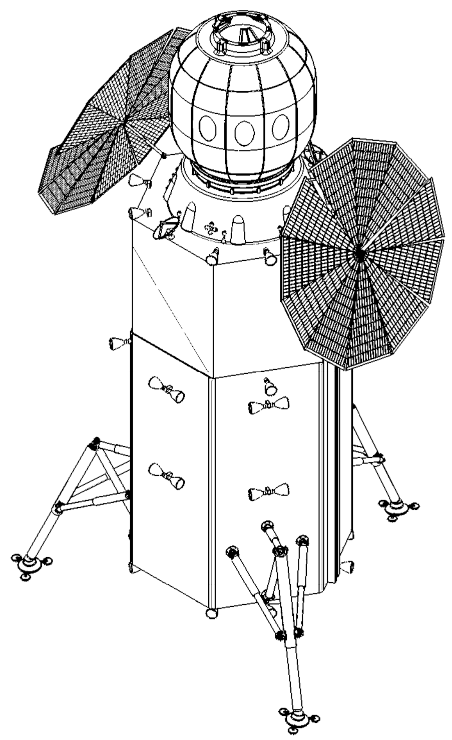
Scale: 1:250
All Units in mm



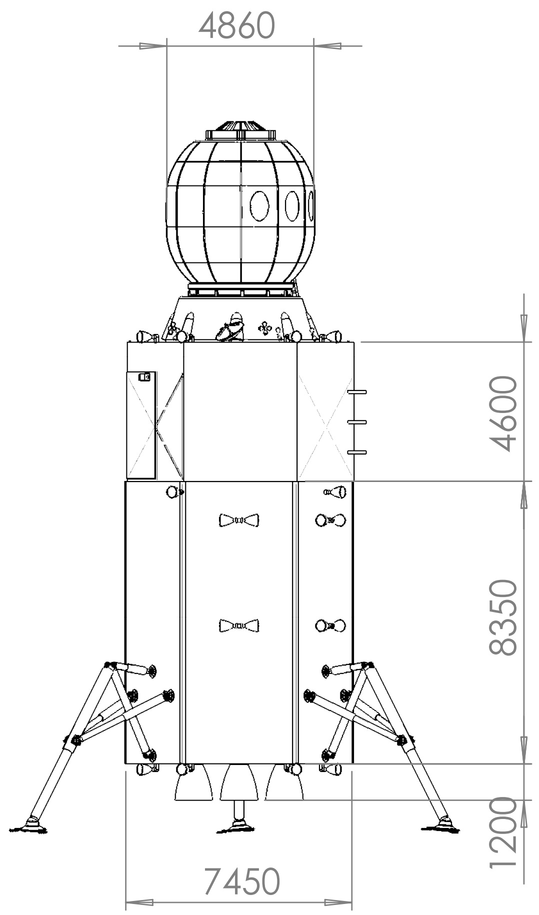
Top View



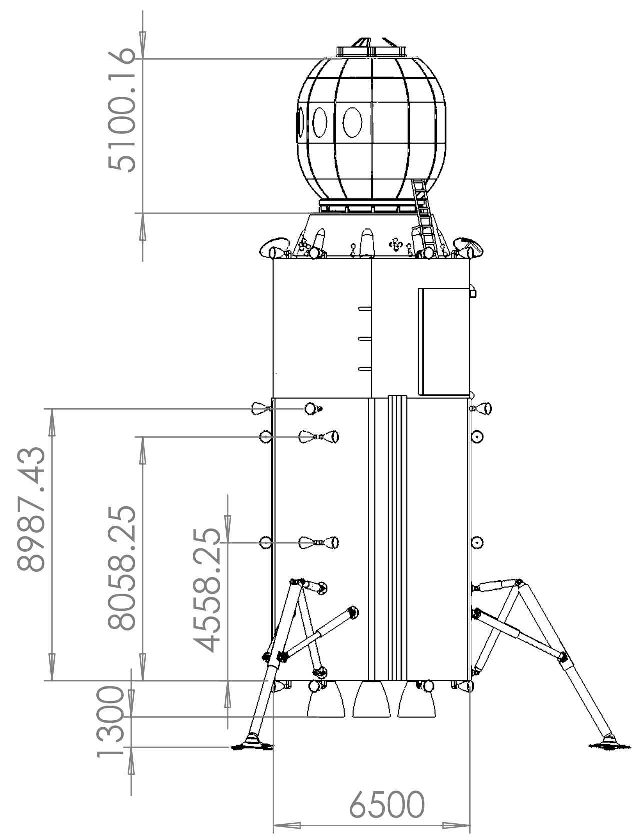
Bottom View



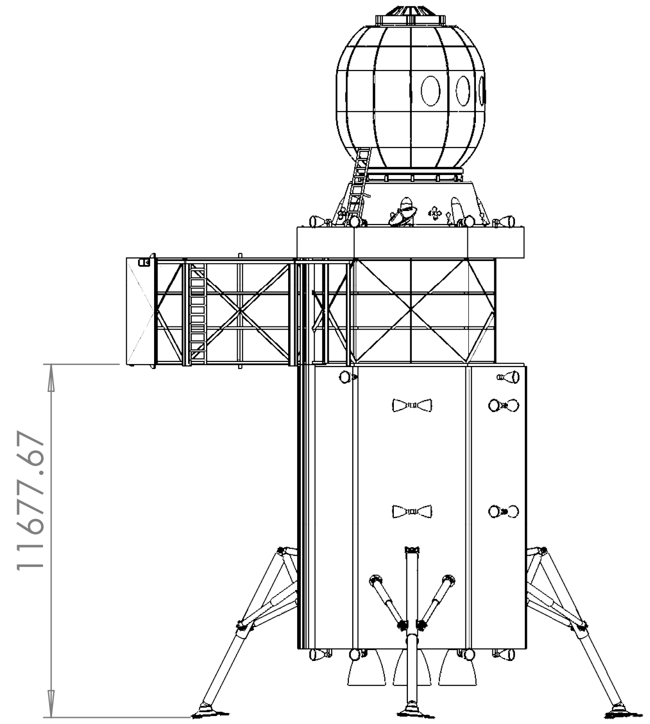
Isometric View



Front View



Side View



Side View
(Cargo Door Open)

Trajectory

In this chapter, all Ariana's orbital maneuvers and transfer strategies including the launch system selection, docking scenario and transfer from DSG to the lunar surface and back is discussed.

8.1 Trajectory Requirements

The operational requirements of RFP for transfer trajectories have been listed in table 8.1.

Table 8.1: The Operational Requirements of RFP

No.	Requirement
1	Ariana shall operate its first mission before December 31, 2028.
2	Ariana shall be able to dock DSG while DSG is the passive vehicle in all docking sequence.
3	Ariana shall be able to land anywhere on the lunar surface.

8.2 Near Rectilinear Halo Orbit of DSG

DSG is going to be a crew tended spaceport in its staging orbit, the NRHO, for journeys to the lunar surface and deep space. All the trajectory simulations in this chapter are based on an orbit with characteristics close to NASA's presentation of a southern L2 NRHO in "Cislunar and Gateway Overview" [20]. Features of the simulated NRHO are demonstrated in table 8.2.

Table 8.2: Main Parameters of Simulated NRHO

Parameter	Value
Perilune Radius	3250 km
Period	6.5 days
Northern/Southern	Southern
L1/L2	L2

8.3 Assembly

8.3.1 Launch Vehicle Selection

The vehicle's configuration and the trajectory are dominant characteristics in choosing the appropriate launch vehicle. Further requirements should also be considered, as shown in table 8.3.

Table 8.3: Launch Vehicle Selection Requirements

No.	Launch Vehicle Selection Requirements
1	Upper-stage shall be able to inject the payload into the orbit with the specified mass and trajectory energy level
2	Launch vehicle's development must happen before the date of the vehicle's first assembly module deployment at DSG

Due to the deploy strategy, the main objective in the launch vehicle selection for Ariana's mission is the ability to inject the payload into the translunar orbit (section 5.1.3). Also, the spacecraft's volume and mass play the principal roles. Current heavy-lift rockets with the ability to put the payload into TLI are listed in table 8.4.

Table 8.4: Launch Vehicles with The Ability of Translunar Injection

Launch Vehicle	**Payload to Moon** (kg)	**Payload Fairing Diameter** (m)	**Launch Time**	**Feasibility**
Yenisei [21]	27,000	-	2030	No
SLS Block 1B 8.4m USA PLF	26000	8.4	2020	No
SLS Block 1B 8.4m PLF Short	40,000	8.4	2025	Yes
SLS Block 2	45,000	10	2029	No
Long March 9	50,000	10	2028	No
BFR	>100,00	9	2022	Yes

SLS presents a variety of payload mass, volume, and departure energy levels. It is going to unveil three major upgrades in the 2020s, comprising Blocks 1, 1B, and 2. Concerning the configuration, Block 1 will not be able to house the vehicle into its PLF.

The Big Falcon Rocket (BFR) is under development by SpaceX to operate in 2022. It is privately funded and the launch cost is over 5 billion dollars.

SLS block 1B benefits from Exploration Upper Stage (EUS) which is powered by four RL10 LH2/LOX engines. Besides, the electrical power is granted by onboard batteries [3]. "Block 1B 8.4m PLF Short" has been chosen to launch each vehicle's assembly module to a 185 km LEO and inject them into the designed optimal transfer trajectory (section 5.1.3). Figure 8.1 illustrates how each module fits into the chosen launch vehicle's fairing.

(a) Propulsion Module

(b) Cargo and Habitat Modules

Figure 8.1: Ariana's Modules Housing into SLS Block 1B

8.3.2 Docking

Ariana's PM and PLM are supposed to capture into NRHO separately. The current section is focused on all the required maneuvers from the time of Ariana's capture until it is completely docked to DSG.

• Capture

After SLS's upper stage places the PM in TLI, the main engines will complete a burn to capture into NRHO. As indicated previously in section 5.1.3, this maneuver takes place at NRHO's apolune with an 804 m/s engine burn.

• Phasing Maneuver

It is presumable that Ariana would be captured into a different phase of NRHO from where the DSG is located. Thus, a phasing maneuver is required to put Ariana in the proximity of DSG for further procedures. Phasing is designed to happen at perilune, where an impulsive burn takes Ariana on an orbit with a different period. Phasing maneuver ends in apolune where Ariana meets DSG, and following maneuvers comprising rendezvous and docking occurs. In table 8.5, required ΔVs have been listed as a function of epoch shift and the number of revolutions.

Table 8.5: Required ΔVs For Phasing Maneuvers

Time Shift (hour)	**1.5 Rev ΔV (m/s)**	**2.5 Rev ΔV (m/s)**	**Time Shift (hour)**	**1.5 Rev ΔV (m/s)**	**2.5 Rev ΔV (m/s)**
2	15.1	4.2	-2	14.6	3.9
4	31.1	8.7	-4	27.2	7.9
6	47.7	14.1	-6	39.3	12.6
8	66.9	19.3	-8	53.9	17.8
10	85.9	24.9	-10	67.1	22.3
18	224.4	59.8	-18	113.1	55.9

The worst case in phasing maneuver occurs when Ariana locates exactly half a period away from DSG, which requires 250 m/s engine burn and 112 days (17.5 periods) to locate Ariana in the vicinity of DSG.

• Rendezvous

Transfer trajectory is followed by a rendezvous maneuver in nearly 100 km distance from DSG. Figure 8.2 illustrates the maneuver cost with respect to change in Time of Flight (TOF) and distance from DSG.

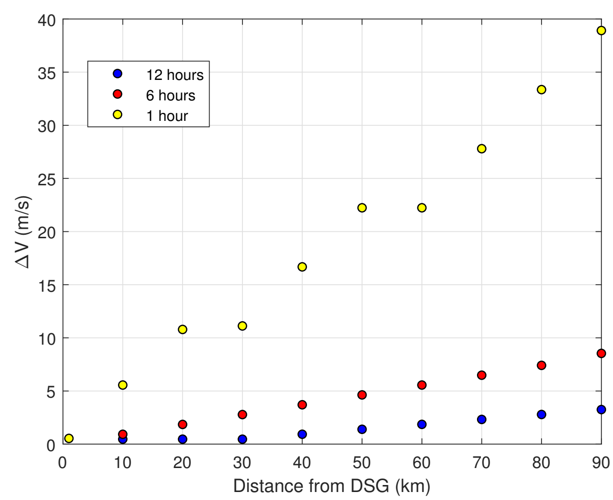


Figure 8.2: Rendezvous ΔV

In order to locate the vehicle in proper position for docking, a two-burn maneuver sets Ariana to the distance of 10-50 meter from DSG. Due to two main errors including 5 km distance of simulated approaching transfer orbit from DSG and the margin of 15 km for the operational uncertainties, a budget of 10 m/s is assumed for rendezvous maneuver.

Finally, ΔV budget for docking process has been shown in table 8.6.

Table 8.6: Docking ΔV Budget

Maneuvers	ΔV (m/s)
Capture	804
Phasing	250
Rendezvous	10
Total	**1054**

8.4 DSG To The Lunar Surface Transfer

The operations to the lunar sites would be started after Ariana is docked and on-orbit tests carried out successfully. In this section, transfer strategies from DSG to the surface and back for both crew and cargo modes are presented.

8.4.1 Design Objectives

As Ariana is subjected to work in both crew and cargo modes, different design objectives for the lunar surface transfers have been defined (Table 8.7).

Table 8.7: Crew/Cargo Mode Objectives

No.	Objective	Mode
1	Since the mission is manned, transfer time shall be minimized. Longer Time TOF increases consumables for keeping humans alive and also expands habitable volume (section 11.2.1).	Crew
2	The transfer can accomplish in more time as there is no crew in the vehicle.	Cargo
3	Ariana must have global lunar surface access.	Crew/Cargo

8.4.2 Transfer Between DSG And LLO

The Low Thrust concept is minimizing the ΔV cost. Due to the low Technology Readiness Level (TRL) of the propulsion system and its inability in ascent/descent phase, this strategy is eliminated.

Trajectory simulations between DSG and landing sites are based on Short TOF transfer. Circular LLOs with different inclinations and Right Ascension of Ascending Nodes (RAAN) as destination orbits, provide global lunar surface access [22]. According to section 8.5, LLO's altitude for an optimal ascent/descent is chosen to be 150 km for DSG to LLO leg and 85 km vice versa. Simulations are produced by solving two-point boundary value problem, fixing the first point on DSG's location and the other on the latitude and longitude of the specified landing site.

• Crew Mode

As mentioned in table 8.7, in crew mode both ΔV and TOF are critical. Therefore, TOF for each leg has to be limited, so, targeted LLOs should pass above the landing site. In this situation, the time that Ariana waits for starting the landing phase will be no more than 2.5 hours (one LLO period). Transfer trajectories from NRHO to the optimal LLO for different latitudes and longitudes and back are simulated via MATLAB. Simulation results indicate an increasing trend in propulsive cost from polar to equatorial regions. Round trip ΔV contour for the entire lunar surface is shown in figure 8.3.

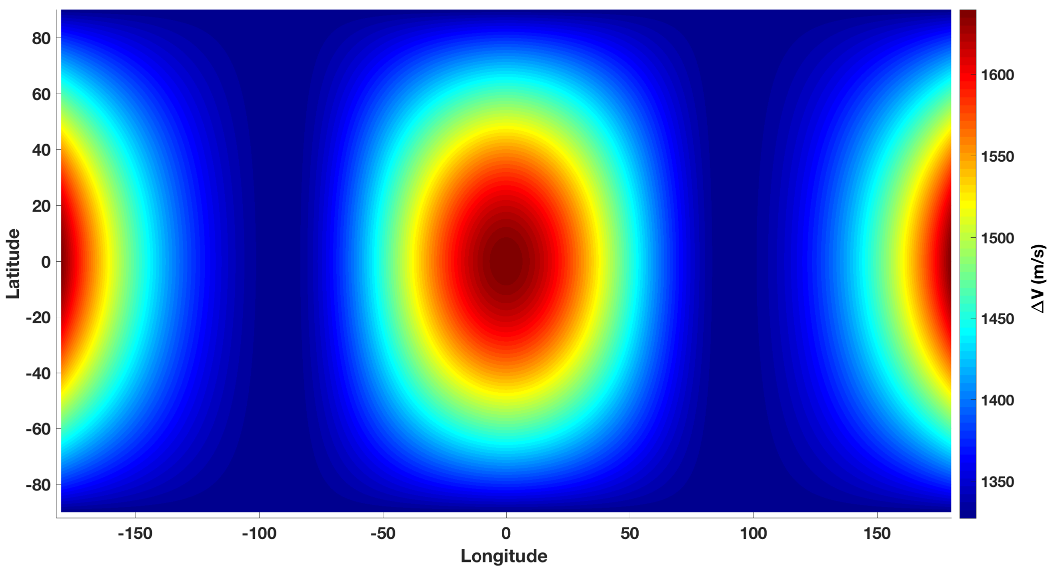


Figure 8.3: Round Trip ΔV Contour For Lunar Surface

Also, figure 8.4 presents a transfer trajectory for the equatorial landing target.

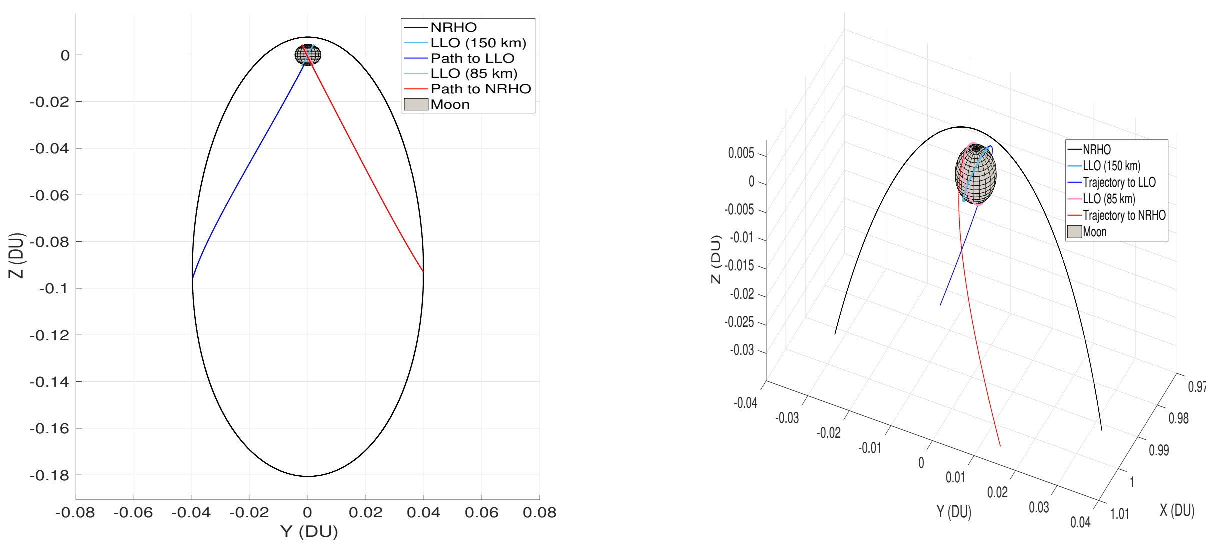


Figure 8.4: Simulated Trajectory For Round Trip to Moon

Scheduling is another determinative parameter in the lunar transfers. DSG is not accessible from LLO in all positions, so a timeline is necessary for every round trip to Moon. In table 8.8, schedule of critical landing points has been listed.

Table 8.8: Crew Mode Transfer Timeline

Latitude	Longitude	DSG To Surface		Stay Time (day)	Surface To DSG		Total Time (day)
		ΔV (m/s)	TOF (day)		ΔV (m/s)	TOF (day)	
0	0,180	802.4	1	6.1	837.1	1	8.1
90	0 to 90	652.6	0.5	6.5	674.6	0.5	7.5
0	45	693.9	1	6.5	709.1	1	8.5

- **Cargo Mode**

Ariana will be capable to deliver more payload if the transfer ΔV decreases. As previously stated, TOF in cargo mode is not a constraint, therefore, transfers with lower ΔV is possible in cost of more TOF. While the rotation period of the Moon is 27.3 days, in the worst case by 13.6 days waiting in a polar LLO, landing sites with different latitudes and longitudes will be accessible for Ariana without any plane change maneuver. Hence, in each cargo delivery operation, Ariana should travel to a polar LLO and wait for the right opportunity for landing and use the same strategy for getting back to DSG. According to figure 8.3, the lowest required ΔV for transferring to a polar LLO is 652.6 m/s and getting back with 674.6 m/s. Thus, round trip ΔVs for the entire lunar surface are the same and equal to 1327.2 m/s. Timeline of these transfers for critical points has been demonstrated in table 8.9.

Table 8.9: Cargo Mode Transfer Timeline

Longitude	Latitude	From DSG To Surface			Stay Time (day)	From Surface To DSG		
		ΔV (m/s)	Δt to LLO (day)	Wait Time in LLO		ΔV (m/s)	Δt to NRHO (day)	Wait Time in LLO
90	0 to 90	652.6	0.5	0	6.5	674.6	0.5	0
45	0 to 90	652.6	0.5	3.4	5.8	674.6	0.5	10.2
0 , 180	0 to 90	652.6	0.5	6.8	5.8	674.6	0.5	6.8
135	0 to 90	652.6	0.5	10.2	5.8	674.6	0.5	3.4

Also, in cases that payloads should be delivered to the Moon quickly, Ariana uses crew transfer scenario for this purpose.

8.5 Descent Phase

The powered descent phase of the Ariana mission is a key challenge in the mission design in terms of flight safety, including the eject considerations and economical fuel criteria. The descent trajectory is thoroughly discussed in the following sections.

8.5.1 Descent Orbit Insertion and Powered Descent Initiation

The descent orbit insertion burn drops Ariana from the 150 km circular orbit to an elliptical orbit which has a perilune of 15.24 km. The perilune of 15.24×150 km orbit[1] is the altitude where the powered descent is initiated with the descent engines of the main propulsion system. Figure 8.5 indicates the LLO and descent trajectory.

[1]This was the same perilune height used in Apollo [23].

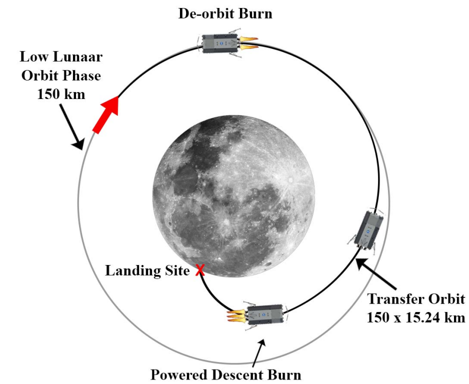


Figure 8.5: The LLO and Descent Trajectory

8.5.2 Powered Descent Trajectory

The powered descent trajectory of the Ariana mission comprises three operational subphases: "Near fuel-optimum", "Landing-approach"(pitch-up maneuver will be performed at the beginning of this phase.), "Final translation and touchdown" [23]. Figure 8.6 and 8.7 depict the Ariana descent trajectory.

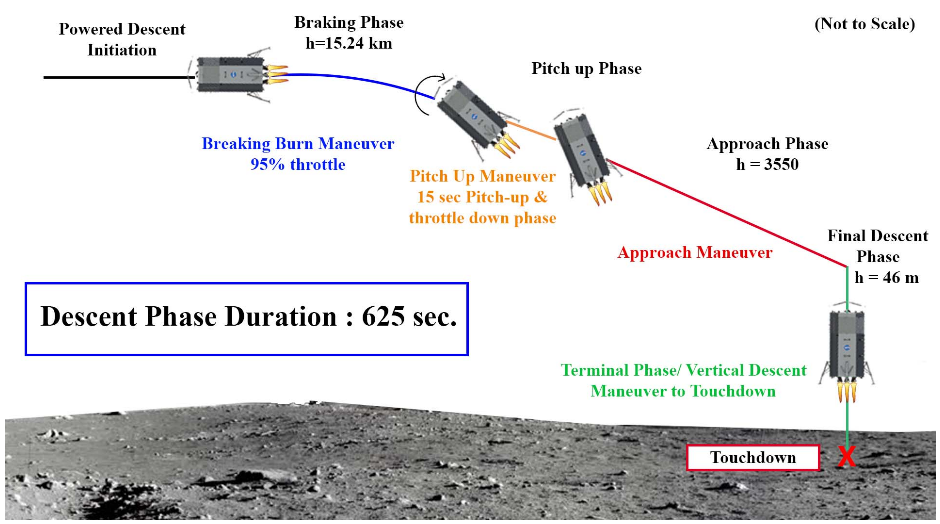


Figure 8.6: Ariana Descent Trajectory Schematic [23], [24], [25]

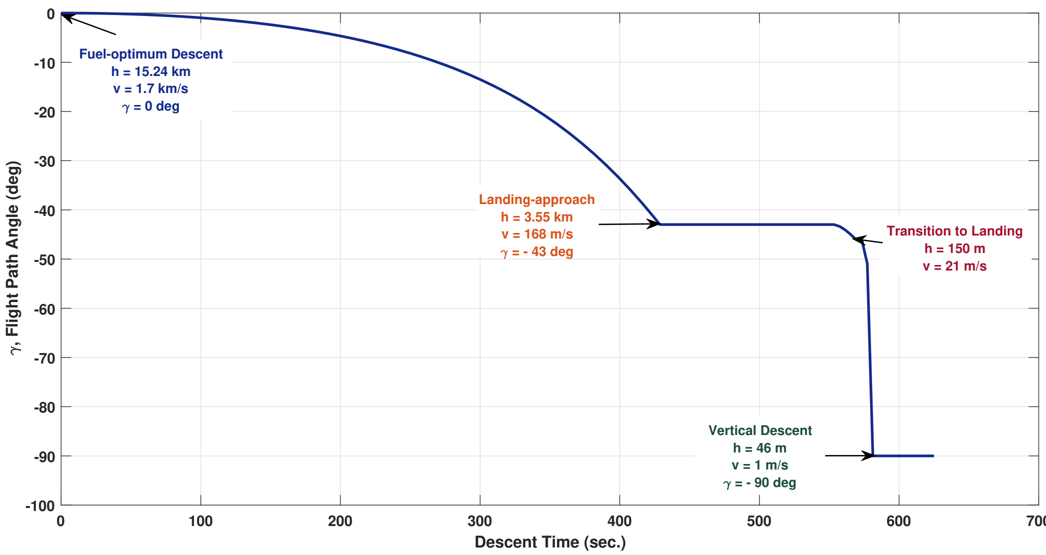


Figure 8.7: Ariana Descent Trajectory [23], [24], [25]

Fuel-optimum Descent: The "Fuel-optimum" phase is initiated at the perilune of 15.24×150 km orbit and terminated at the transition altitude of 3.55 km. It is designed to efficiently reduce the initial descent velocity by the main engines aligned with Ariana's velocity vector.

The initial altitude of this phase is chosen with respect to the following criteria [23]: • It shall be as low as possible from the fuel consumption point of view. • A margin of safety is crucial considering the control and guidance errors and ejection process. • The consideration of the lunar geodetical map shall be made to mitigate any obstacle impacts (section 6) .

A thrust margin[2] of 5-8% is necessary to remove dispersions during this phase of powered descent [25].

Pitch-up Maneuver: At the beginning of the "Landing-approach" phase, the "Pitch-up maneuver" is performed by the attitude determination and control system and main engines.

Landing-approach: The "Approach" subphase is initiated at approximately 3.55 km and ends at the beginning of the "Transition to Landing" phase which is about 150 m. The throttle setting is roughly 60% full main engines thrust.

Final Translation and Touchdown: "Transition to landing" subphase begins at the altitude of 150 m and the velocity of 21 m/s and continues until the altitude of 46 m and velocity of 1 m/s. Main engines provide "Vertical descent" with the rate of 1 m/s, and finally, shut down 1 m above the lunar surface.

[2]Between fuel-optimum descent set throttle and the maximum available engine power.

Thrust to Weight Trade: The trade study shows that the improvement in the initial T/W_{Lunar} from 1.8 to 3.8 does not affect the total descent ΔV significantly. Hence, ShadX team decide to choose T/W_L=2 with the negligible penalty of additional ΔV. Figure 8.8 presents the trade study evaluation.

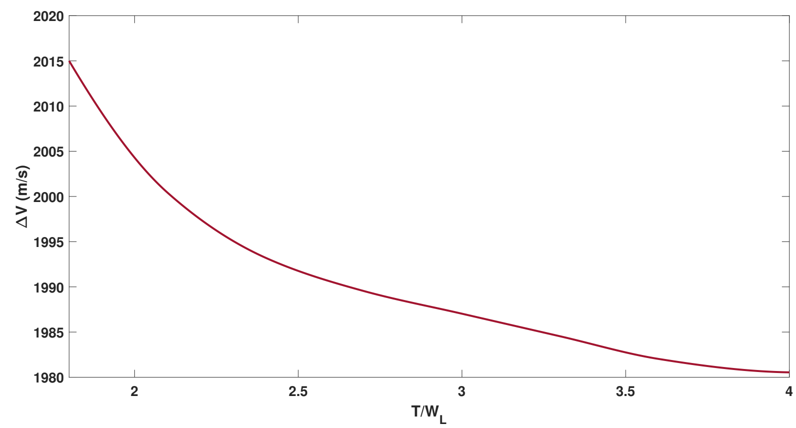


Figure 8.8: T/W_L Trade Study [24]

8.6 Ascent Phase

The ascent phase consists of the "Vertical rise", "Single axis rotation", and "Powered explicit guidance" subphases. They all perform with the ascent engines of the main propulsion system [25]. Figure 8.9 and 8.10 demonstrate the ascent trajectory.

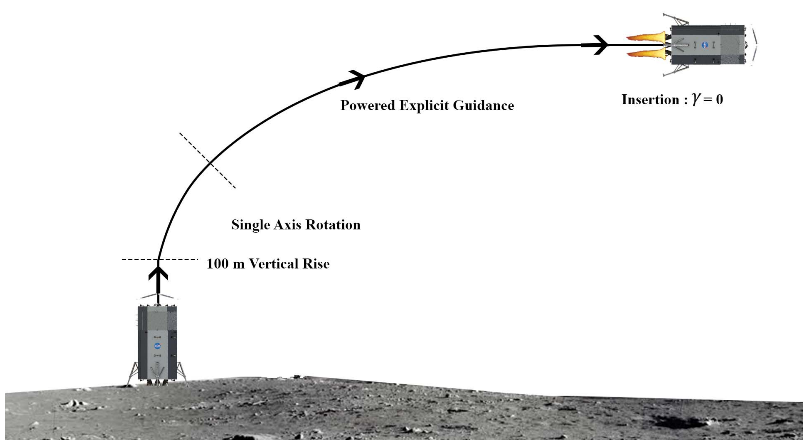


Figure 8.9: Ariana Ascent Trajectory Schematic [25]

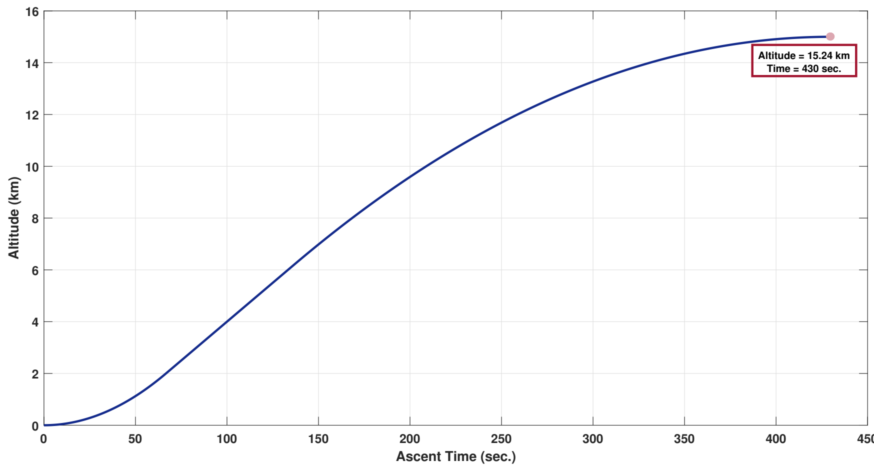


Figure 8.10: Altitude vs. Ascent Time [25]

Vertical Rise: During this phase of ascent, Ariana flies to a height of 100 m to achieve terrain clearance. The time of "Vertical rise" is nearly 10 seconds.

Single Axis Rotation: This maneuver is performed to re-orient the Ariana to the desired attitude for the next subphase.

Powered Explicit Guidance: The final phase of the ascent is focused on minimizing the required propellant. It has a burn time of approximately 380-385 seconds.

8.7 ΔV Budget

Finally, all required ΔVs for a round trip missions to the Moon's surface have been listed in table 8.10.

Table 8.10: ΔV Budget[3]

Transfer Mode	Departure from NRHO	Capture into LLO	Deorbit from LLO	Descent	Ascent	Capture into LLO	Departure from LLO	Capture into NRHO	ΔV Budget (m/s)
Cargo	37.60	615.00	30.00	2000.00	1841.00	16.00	635.30	39.30	**5214.20**
Crew	198.70	603.70	30.00	2000.00	1841.00	16.00	612.30	224.80	**5526.50**

All the trajectories presented in this chapter have been simulated within the context of the Circular Restricted Three-Body Problem. Perturbations impacts such as unsymmetrical gravity, fourth body effect, and solar radiation pressure (SRP) on these simulations are thoroughly explained in section 10.1.

[3]Also, Extra 10 m/s ΔV considered for rendezvous with DSG.

Propulsion

9.1 Propulsion Overview

Ariana Main Propulsion System (AMPS) is comprised of 5 RL10 CECE engines burning cryogenic Liquid Hydrogen (LH2) and Liquid Oxygen (LO2) propellants. In the expander cycle of main engines, fuel is used to cool down the nozzles as well as combustion chambers and power the turbines which drive both the fuel and oxidizer pumps [26]. A cylindrical tank in the center of the PM for storing LO2 and an annular shaped tank for storing LH2 set the propellant tank arrangement for AMPS.

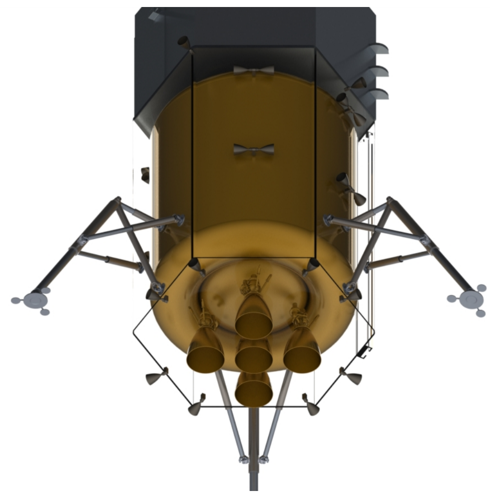


Figure 9.1: Ariana Main Propulsion System

9.2 Propulsion Requirements

The primary objective of AMPS includes the initial capture burn to NRHO, the departure impulse burn from NRHO, the capture burn into the 150 km LLO, the deorbit insertion burn from the 150 km circular orbit, the descent and ascent burn and three final burns to climb back into the 85 km circular orbit and NRHO. Table 9.1 provides a list of AMPS main requirements.

Table 9.1: Propulsion Subsystem Level Requirements

No.	Requirement
1	Propulsion system shall be able to deliver at least 5526.5 m/s of ΔV.
2	Propulsion system shall be restartable.
3	Propulsion system shall be redundant to allow single engine failure without loss of crew life.

9.3 Propulsion System Selection

To satisfy the Mission Requirements, the following choices for propulsion system are considered: cold-gas system, electric and solid option for propulsion, nuclear propulsion, monopropellant and bipropellant system.

Due to the mission requirements, cold gas system, electric and solid option for propulsion are eliminated. Although the cold gas system has the least complexity as well as cost, and can provide multiple restarts and pulses, it has low thrust levels plus specific impulse and is disqualified following that [27]. Solid propellant engines are also not suitable choices for this mission because of restarting disability, limitation of the variability in thrust range , and incapability of turning off. Due to low thrust capabilities, weight ratios, and high power consumption, the electric propulsion system can limit the mission requirements and it was disqualified as well. According to low TRL, nuclear propulsion was ruled out as a candidate for the propulsion system considering the first launch in 2025.

Since the mission design is mass limited and monopropellant systems have lower specific impulse compared to bipropellant options, an additional propellant mass is needed for an equivalent ΔV burns. Consequently, bipropellant systems have tended to be a more appropriate choice for AMPS.

9.4 Propellant Selection

Firstly, the selection procedure of a suitable fuel and oxidizer is coupled with choosing an appropriate engine which cater to the mission thrust requirements. Considering that, at first, a trade study has been conducted to investigate whether to use hypergolic bipropellant or cryogenic bipropellant.

Hypergolic propellants require less thermal protection in comparison with cryogenic propellants, resulting in lower power and mass budget. In addition, they do not need an ignition system, which makes the procedure of engine restarting easier and reduces complexity as well. The most common combinations of this category are N2H4/NTO [1], MMH/NTO[2], and UDMH/NTO[3], among which, the first two combinations have the extensive heritage in use for multiple deep space missions.

Despite all their benefits, some key factors in choosing propellants have made hypergolic propellants inappropriate for AMPS. In the most optimistic case, the specific impulse of propulsion systems based on hypergolic propellants is reached 335-345 seconds, which in turn increases overall wet mass of Ariana significantly. Furthermore, commonly used hypergolic propellants are extremely toxic and corrosive. According to their toxicity and the number of scheduled missions, emitting a large amount of these propellants causes harmful pollution and will be a severe threat to CM health and the lunar environment. Another reason for eliminating these toxic propellants is the difficulties of handling and fuelling operations, which leads to added complexity and cost increment. Emphasizing on reusability requirement, fuel and oxidizer of this category leave a considerable amount of residue on the engines when they are burnt, therefore, it means that they have less reusability compared to cryogenic propellants.

To ensure choosing the most appropriate propellants for the mission, more explicit trade study is conducted for investigating the most common fuels and oxidizer of both categories. In so doing, each of them is rated out of 10 in terms of its density, difference between freezing temperature and boiling point, hazard and toxicity. CH4[4], LH2 and LO2 are cryogenic propellants and N2H4, MMH, UDMH and NTO are hypergolic propellants. Table 9.2 indicates the trade study evaluation.

[1] Hydrazine/Nitrogen Tetroxide
[2] Monomethylhydrazine/Nitrogen Tetroxide
[3] Unsymmetrical dimethylhydrazine/Nitrogen Tetroxide
[4] Liquid Methane

Table 9.2: Propellant Trade Study Evaluation

Propellant	Density (kg/m^3)	Score	Freezing Point ($°C$)	Boiling Point ($°C$)	Temperature Difference	Score	Hazard	Score	Toxicity	Score	Total Score
Weight	**0.261**	**-**	**-**	**-**	**0.153**	**-**	**0.124**	**-**	**0.461**	**-**	**1**
CH4	422.6	4.5	-182.5	-161.9	20.6	2	Moderate	5	Low	10	**6.715**
LH2	70.8	1	-259.3	-252.9	6.4	1	Moderate	5	Low	10	**5.645**
N2H4	1008.5	10	2	114	112	8	High	1.25	Moderately High	3	**5.382**
MMH	874	9	-52	87.5	139.5	10	Moderate	5	High	1	**4.969**
UDMH	783	8	-58	63	121	9	Moderate	6	High	2	**5.139**
LO2	1140	1	-218.8	-183	35.8	2	Low	9	Low	10	**6.294**
NTO	1450	10	-9.3	21.15	30.45	1	Moderately Low	7.5	High	1	**4.164**

Density: Keeping the wet mass of Ariana as low as possible is our priority, but the importance of launcher PLF dimensions limitation cannot be neglected. Therefore, in this trade study, higher density gets a higher score but the requirement of keeping Ariana's wet mass low will be satisfied in section 9.5 considering the specific impulse factor for each engine.

Temperature difference: In this trade study, the wider temperature difference between the freezing point and boiling point leads to less thermal control of the propellant tanks and gets a high score.

Hazard: Indicates the level of flammability and reactivity hazard; lower score means readily capable of detonation, explosive decomposition, and burning.

Toxicity: The most significant factor in this trade study is toxicity due to its great impact on CM health. Some common hypergolic combinations such as N2H4/NTO, MMH/NTO, and UDMH/NTO are highly toxic in comparison to cryogenic propellants.

As table 9.2 indicates, it was determined that CH4, LH2 and LO2 are the wisest choices for fuel and oxidizer. As a result of hypergolic propellants disadvantageous and the trade study evaluation results, they are not fit to the mission requirements. Consequently, our final candidates for fuel are CH4 and LH2. Table 9.3 delineates the advantages and disadvantages of liquid hydrogen and liquid methane.

Table 9.3: Liquid Hydrogen vs. Liquid Methane

Propellant	Advantage	Disadvantage
LH2/LO2	· High Isp (445-465 s) · ECLSS support (Oxygen and Water) · A good source for using in power subsystem · Nontoxic · Extensive heritage	· Low density · High temperature restriction · Losses as a result of vaporization · High cost
CH4/LO2	· Low cost · Readily available · Easy to handle · Wide temperature difference between bp and fp · Nontoxic · Easy to store · Similar storage temperature for the fuel and oxidizer	· Low Isp (350 s) · Required high thermal control · Low heritage

Since the mission design is mass limited, a trade study has been performed between LH2/LO2 and CH4/LO2 to show the amount of reducing propellant mass in case of using LH2/LO2 instead of CH4/LO2 combination for AMPS in the descent and ascent phase, assuming a specific impulse of 445 seconds for LH2/LO2 vs. 350 seconds for CH4/LO2. Figure 9.2 indicates the evaluation result. As can be inferred from figure 9.2, albeit storing liquid methane in fuel tanks requires less thermal control compared to liquid hydrogen, using CH4/LO2 for main propulsion system increased Ariana's wet mass significantly. The thermal analysis indicated that the added dry mass of the thermal subsystem due to more thermal control required for LH2/LO2 is much less than the propellant mass increment by using CH4/LO2 instead of LH2/LO2. Propellant thermal control is thoroughly discussed in section 12.1.2. Moreover, the analyzed data gathered from the past Moon missions shows areas of water ice on the lunar surface with hydrogen and oxygen components, which is a valuable propellant resource for future deep space human missions. For more information about water ice on the lunar surface, refer to 6.3. Eventually, considering all aspects of liquid hydrogen; it is the wisest fuel choice to satisfy the mission requirements.

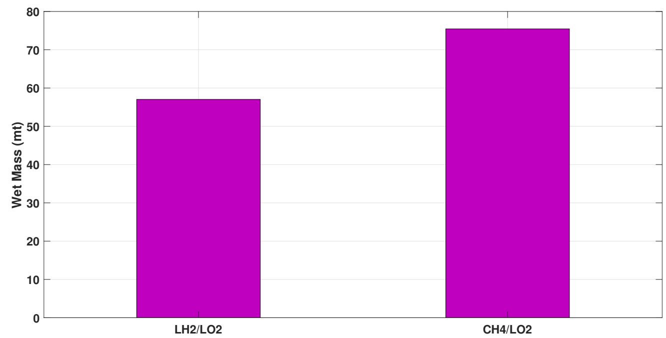


Figure 9.2: Propellant mass needed for the descent and ascent phase using LH2/LO2 vs. CH4/LO2 for AMPS

9.5 Main Engine

The main engine selection process has focused on some significant features, which cater to the mission requirements. Table 9.4 indicates the aforementioned features.

Table 9.4: Main Engine Requirements

No.	Requirement
1	The main engine shall be capable of being deeply throttled.
2	The main engine shall have a high Isp to minimize Ariana wet mass.
3	The main engine shall have a high number of in-space starts and a great lifespan to maximize the number of missions.

Referring to these, it is necessary to consider the significant role of cooling process in satisfying engine reusability requirement. Regenerative cooling of an engine with a pressure-fed system over a wide throttling ratio is not feasible, which eliminated Pressure-fed engines [28]. Ergo, our final candidate is a Pump-fed engine.

In the following, a decision has been made to determine the number of engines. Although an increase in the number of main engines led to less reliability and more expenses, complexity plus mass of propulsion system, emphasizing on CM safety, more than two engines are required to be fully redundant with single engine failure. Thus, one and two engine configurations are not an appropriate choice for the number of engines. Three or four engine configurations are suitable candidates for the number of engines, but five engines have selected to maximize the number of possible missions at negligible penalty of increasing added weight of each engine. Table 9.5 shows 5-engine configuration in detail.

Table 9.5: 5-Engine Configuration Pros and Cons

Configuration	Pros	Cons
Five-engine configuration	. Full redundancy with single engine failure . Thrust vectoring . Higher number of possible missions compared to 1-4 engine configuration	. Lower reliability (0.9975) compared to one engine reliability (0.9995) . High complexity . High mass(5×159 kg)

The main engine requirements and the thrust needed fitting to 5-engine configuration recommended the newest version of RL10 family for AMPS. The RL10 CECE is based on the great heritage of the reliable RL10 engine, which has been used on the "Centaur", "S-IV", and "Delta Cryogenic Second Stage" upper stages. Table 9.6 shows RL10 CECE specification.

Table 9.6: The RL10 CECE Specifications (Base Demonestrator)[29]

Propellant	Thrust	Specific Impulse	Weight/Dimension
Fuel: LH2 Oxidizer: LOX	66.7 kN	Isp: 445 s Throttling: 10 20:1 Reliability: 0.9995 In-Space Starts (total): 50 Service Life (total): 10,000 s	Weight : 158.7 kg Diameter : 1.57 m

At the beginning of the descent phase, the thrust to Earth weight ratio could be varied from 0.3 to 0.5. Considering different fuel consumption ratios during fuel-optimum descent subphase, the maximum throttle ratio of 11:1 is required. A 17.6:1 throttling ratio of RL10 CECE would easily satisfy the main engine deep throttle requirement [23].

9.6 Propellant Mass and Volume

Requiring propellant mass for both cargo and crew modes can be calculated using the Tsiolkovsky rocket equation [27]. A 10% margin is added to propellant mass considering propellant reserve (5%), trapped (3%), loading error (0.5%), outage (1.5%) [30]. Propellant tank sizing is discussed in section 14.5.1. As discussed in section 8.4.2, two scenarios for cargo missions are possible . Table 9.7 includes the total propellant mass and volume and burn time of the mission for both crew and cargo modes.

Table 9.7: Propellant Mass & Volume & Engines' Burn Time Specifications

Transfer Mode	Parameter	Departure Impulse from NRHO	Capture into a 150 km LLO	Deorbit from a 150 km LLO	Descent	Ascent	Insertion into a 85 km LLO	Insertion into transfer orbit (LLO to NRHO)	Capture into NRHO	Total
Cargo (scenario one) $m_{dry} = 12.39\ mt$ $m_{PL} = 19.5\ mt$	ΔV (m/s)	198.70	603.70	30.00	2000.00	1841.00	16.00	612.30	224.80	**5526.50**
	Burn Time (s)	161.22	468.02	20.25	625.00	430.00	4.04	154.01	49.14	**1911.68**
	Propellant Mass Required (mt)	4.63	12.83	0.59	31.58	16.97	0.12	4.22	1.41	**72.34**
	Propellant Mass with Added 10 Margin (mt)	5.09	14.11	0.65	34.74	18.67	0.13	4.64	1.55	**79.58**
	LH2 (mt)	0.74	2.05	0.09	5.05	2.71	0.02	0.67	0.22	**11.57**
	LO2 (mt)	4.35	12.06	0.56	29.69	15.96	0.11	3.97	1.32	**68.01**
	LH2 Volume (m^3)	10.45	28.98	1.34	71.31	38.33	0.27	9.53	3.18	**163.37**
	LO2 Volume (m^3)	3.82	10.58	0.49	26.04	14.00	0.10	3.48	1.16	**59.66**
Cargo (scenario two) $m_{dry} = 12.39\ mt$ $m_{PL} = 22.5\ mt$	ΔV (m/s)	37.60	615.00	30.00	2000.00	1841.00	16.00	635.30	39.30	**5214.20**
	Burn Time (s)	31.41	509.31	21.58	625.00	430.00	4.33	171.32	9.16	**1802.11**
	Propellant Mass Required (mt)	0.92	13.95	0.63	33.65	18.20	0.13	4.68	0.27	**72.41**
	Propellant Mass with Added 10 Margin (mt)	1.01	15.34	0.69	37.01	20.02	0.14	5.15	0.29	**79.65**
	LH2 (mt)	0.15	2.23	0.10	5.38	2.91	0.02	0.75	0.04	**11.58**
	LO2 (mt)	0.86	13.11	0.59	31.63	17.11	0.12	4.40	0.25	**68.08**
	LH2 Volume (m^3)	2.07	31.49	1.43	75.98	41.09	0.29	10.57	0.60	**163.52**
	LO2 Volume (m^3)	0.76	11.50	0.52	27.75	15.01	0.10	3.86	0.22	**59.72**
Crew $m_{dry} = 24.20\ mt$ $m_{PL} = 6\ mt$	ΔV (m/s)	198.70	603.70	30.00	2000.00	1841.00	16.00	612.30	224.80	**5526.50**
	Burn Time (s)	163.65	475.09	20.56	625.00	430.00	4.11	156.58	49.96	**1924.95**
	Propellant Mass Required (mt)	4.70	13.03	0.60	32.06	17.26	0.12	4.29	1.43	**73.47**
	Propellant Mass with Added 10 Margin (mt)	5.17	14.33	0.66	35.26	18.98	0.13	4.72	1.57	**80.82**
	LH2 (mt)	0.75	2.08	0.10	5.13	2.76	0.02	0.69	0.23	**11.75**
	LO2 (mt)	4.42	12.25	0.57	30.14	16.22	0.11	4.03	1.34	**69.07**
	LH2 Volume (m^3)	10.61	29.41	1.36	72.39	38.97	0.27	9.68	3.23	**165.92**
	LO2 Volume (m^3)	3.87	10.74	0.50	26.43	14.23	0.10	3.54	1.18	**60.59**

Figure 9.3 indicates the amount of propellant mass with payload mass variation.

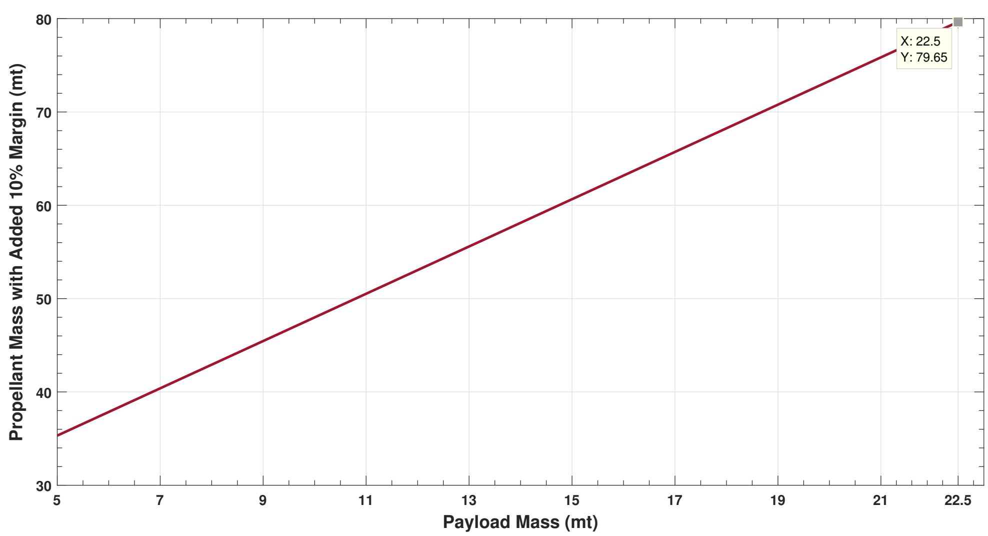


Figure 9.3: Propellant Mass vs. Payload

9.7 On/Off Strategy & Burn Time

As shown in table 9.7 the burn time of all mission stages is calculated. The following on-off strategy is considered to ensure about ten successful missions. The scenario of engines' burn sequences is shown in figure 9.4.

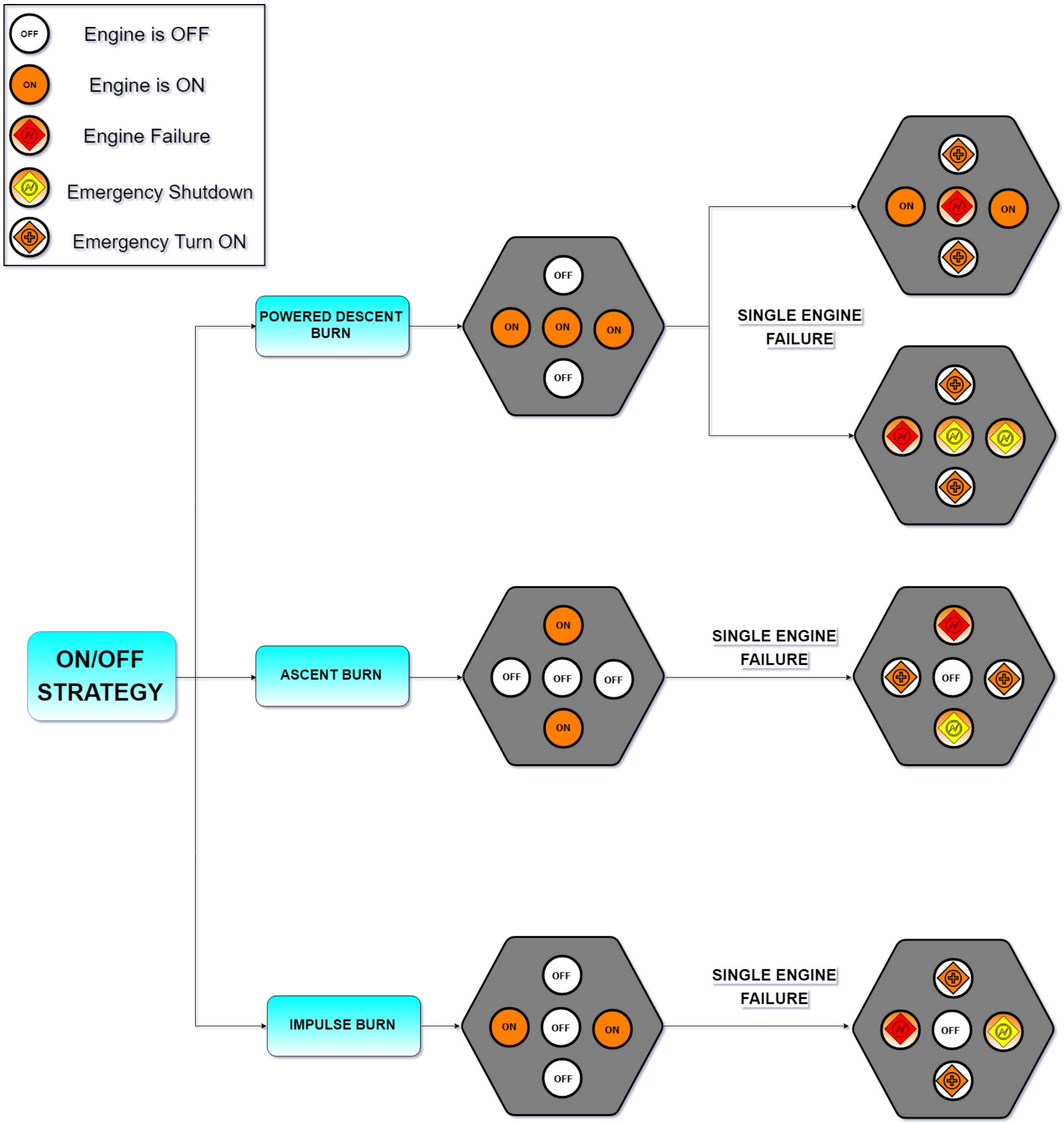


Figure 9.4: On/Off Strategy

Attitude and Orbit Determination and Control Systems

Attitude Determination and Control Systems (ADCS) and Orbit Determination and Control Systems (ODCS) are used in all phases, and each phase has its difficulties which are considered in designing the ADCS and ODCS. Mission requirements are described in table 10.1.

Table 10.1: List of Mission Requirements

No.	A/ODCS Requirement
1	Slew maneuvers to provide needed orientation during landing and descent phases for main engine pointing
2	Slew maneuvers required for station and attitude keeping during the transition phase(canceling perturbations effects)
3	Docking phases to DSG
4	Hazard detection and collision avoidance during mission
5	Self-Modification, adaptive and fault tolerant performance to operate in critical situations

10.1 Perturbations

Two of the main requirements of ADCS and ODCS of Ariana are attitude control and station keeping which is mentioned in table 10.1. In order to satisfy those, perturbations effects analysis is accomplished. Generally, perturbations are divided into two main categories; Internal Perturbations and External Perturbations.

10.1.1 External Perturbations

External effects are mentioned in figure 10.1. They are mainly caused by other masses in the solar system's gravitational field such as the Earth and the Sun. Moreover, the Moon oblateness and unsymmetrical shape cause alterations from the point mass gravity field. This variety of gravitational field is known as J effect [31]. Figure 10.1 shows the accelerations applied to the body versus height in order to distinguish the dominant perturbation effect.

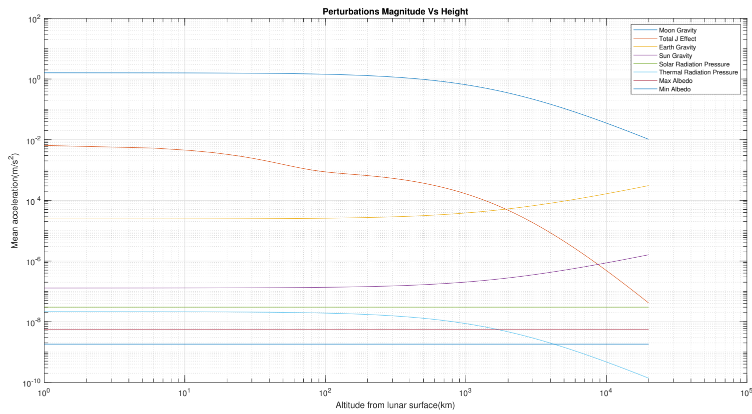


Figure 10.1: Perturbations Magnitude Applied to Ariana vs Height from Lunar Surface

10.1.2 Internal Perturbations

In table 10.2, the internal perturbations and their effects are explained.

Table 10.2: List of Internal Perturbations and Their Effects

Internal Disturbances	Effect on the Vehicle	Typical Values
Center of Gravity (CG) Movements and Moment of Inertia Tensor Alteration	Unwanted torques during translation thrusting Unbalanced torques during firing of coupled thrusters	Separately analyzed in figure 10.2
Thrsut Misalignment	Same as CG uncertainty	0.01 to 0.1 deg
Mismatch of Thrust Outputs	Similar to CG uncertainty	5%
Fuel Sloshing and Propellant Movement	Torques due to fluid motion and Variation in Center-of-Mass (CoM) location	Depends on the spacecraft structure
Dynamics of Flexible Bodies	Oscillatory resonance at bending frequencies Limiting control bandwidth	Depends on the spacecraft structure

CG and Moment of inertia tensor behavior during fuel consumption of vehicle are analyzed separately in section 10.1.3. Thrust misalignments and mismatches don't cause any problem for ADCS at all, as the perturbed moment is far below the moment capacity of the system generated by thrusters. Fuel sloshing and propellant movement in maneuvers are controlled passively as a set of baffles and piston with enough pressure keeps the fuel at the bottom of the tanks.

10.1.3 CG and Moment of Inertia Tensor Sensitivity Analysis

As it is mentioned in table 10.2, two of the main internal perturbations are CG movement and moment of inertia alterations. These effects are mainly caused by fuel consumption of main engines during the mission. Plots of CG place and arguments of the moment of inertia tensor are shown versus current fuel in the tanks in figure 10.2.

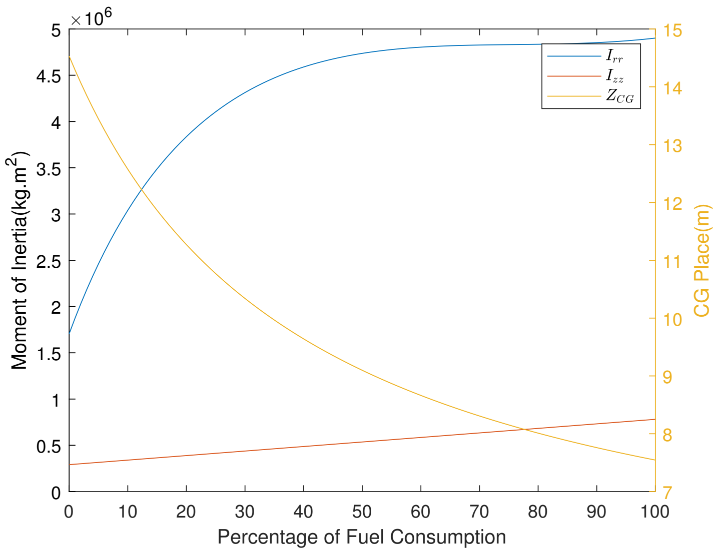


Figure 10.2: CG Movement vs Fuel Consumption

10.2 Attitude Control Methods

In table 10.3 different methods of attitude controlled are clarified.

Table 10.3: Attitude Control Methods and Their Capabilities [26]

Type	**Attitude Maneuverability**	**Typical Accuracy**
Spin Stabilization	Stiff momentum vector	$\pm$ 0.1 to $\pm$ 1 in 2 axes
Zero Momentum (Thrusters)	No constraints, high spin rates possible	$\pm$ 0.1 deg to $\pm$ 5 deg
Zero Momentum (3 reaction wheels)	No constraints	$\pm$ 0.001 deg to $\pm$ 1 deg
Zero Momentum (CMG)	No constraints, high spin rates possible	$\pm$ 0.001 deg to $\pm$ 1 deg

In order to accomplish the mission requirements, four different types of attitude control systems are considered. These types are spin-stabilization, Reaction Wheel Assembly (RWA), 3-axis control thrusters ,and Control Moment Gyros (CMG). According to performance constraints on the spin stabilization method, it can not be used in this mission as it lacks to meet desired pointing accuracy. Thus, a trade study for selecting the control actuators is required to compare the performance of those methods. It is worth to mention that the output of this selection determines further trade study in subsection 10.2.1.

10.2.1 Actuators

Table 10.4: Comparison Between ADCS Actuators [26]

Actuators	Performane Range	Weight (kg)	Power(W)
Thrusters: Hot Gas Cold Gas	.5 to 9000 N <400 N	Variable Variable	Variable Variable
Reaction and Momentum Wheels	.4 to 400 N.m.s for momentum Max torques from .01 to 1 N.m	2 to 20	10 to 110
CMG	25 to 500 N.m of torque	10 to 300	90 to 250

In table 10.4 comparison between performance of the actuators is carried out. The maximum torque available in CMGs and RWAs is about 250 $N.m$; but, based on the spacecraft configuration, a 45-degree slew maneuver in 50 seconds requires at least 200000$N.m$ torque. This amount of torque is far beyond the achievable performance of CMGs and RWAs listed in table 10.4. Hence, the only viable option is thruster which its configuration and sizing are discussed in the next section.

Thrusters Sizing and Configuration

According to a huge amount of torque required for slew maneuvers, the minimum magnitude of thrust for each thruster is at least 4×10^4 N. This constraint will leave us with only two types of hot gas thrusters which are listed in 10.5. The selected thruster is R-40B mainly because of short life span and heaviness of the 11 kN engine.

Table 10.5: Proper Thrusters and Performance Characteristics [32]

Name	Manufacturer	Thrust (N)	Life Span (s)	Life Cycle	ISP (s)	Propellant	MR	Engine Mass (kg)
R-40B	Marquardt	4000	25000	>50000	303	NTO/MMH	1.65	13.6
RS-41	Rocketdyne	11000	2000	-	312	NTO/MMH	1.63	68.95

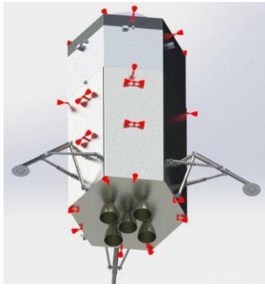
The configuration of the selected thruster is shown in figures 10.3. This configuration is chosen in order to control the Roll, Pitch, and Yaw angle separately. As it is shown in figures 10.3 the positions are selected in places where the exit gas won't damage the body of the spacecraft.

10.2.2 Sensors and Navigation System

In table 10.6 different types of sensors for attitude determination are shown.

Table 10.6: Typical ADCS Sensors

Sensor	Performance Parameters Range	Weight Range (kg)	Power Range (W)
Inertial Measurement Unit (Gyros & Accelerometers)	Drift Rate = .003 deg/hr to 1 deg/hr	1 to 15	10 to 200
Sun Sensors	Accuracy = 0.005 deg to 3 deg	.1 to 2	0 to 3
Star Sensors	Attitude accuracy = 1 arc sec to 1 arc min	2 to 5	5 to 20
Horizon Sensors	Attitude accuracy = .1 deg to 1 deg	.5 to 4	.3 to 10
Magnetometer	Attitude accuracy = .1 deg to 1 deg	.3 to 1.2	<1



(a) Bottom View

(b) Top View

Figure 10.3: Thruster Configuration (Extra four radial thrusters are for assembly phase 10.3.1)

Sensor and navigation system provides exact data about Ariana's state. This system consisting of Inertia Measurement Unit (IMU), star and sun sensors measures rotational and transitional velocity rates used in a closed loop control system in order to generate proper commands for actuators. As a result, this can correct the position and orientation of the vehicle. Ariana's sensor configuration is shown in figure 10.4.

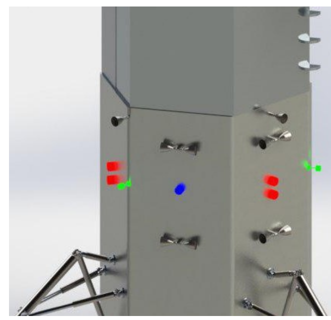


Figure 10.4: Sensor Configuration; Sensor sizes are 10 times more than real size for comfortable viewing; Star Sensors (Blue), Sun Sensors (Red), LiDAR (Green)

Sun Sensors

Table 10.7 shows the performance trade study of the qualified sun sensors. This trade study is done for a large database of sun sensors and many of them were omitted from this list because of the low score of the performance.

Table 10.7: List of Sun Sensors [33–35]

Name	Manufacturer	Weight (g)	Power (mW)	Field of View (°)	Accuracy (°)	Score
NFSS-411	NewSpace	35	130	140*140	0.1	91
FSS	Jena-Optronik	650	330	128*128	0.15	79
FSS	Bradford	375	250	128*128	0.3	75
NCSS-SA05	NewSpace	5	10	114*114	0.5	73

Star Sensors

In table 10.8 the trade study for star sensors selection is carried out. The weight of star sensors varies from 0.1 kg to about 5 kg. Those with more than .5 kg of weight were eliminated at first and are not included in the trade study. The reasoning behind scoring is mostly dependent on the power consumption, weight and accuracy. Thus, PST-3 from TY Space outperforms other star sensors and is chosen as the potential star tracker.

Table 10.8: List of Star Sensors [36–39]

Name	Manufacturer	Weight (kg)	Power (W)	Acquisition (s)	Accuracy ($Arcsecond$)	Score
PST-3	TY Space	0.1	0.6	2	5	91
MAI-SS	Maryland Aerospace	0.17	1.5	0.1	4	89
NST-4	TY Space	0.360	0.6	2	3	84
ST-16RT2	Sinclair Interplanetary	0.235	1	0	5	80

Inertial Measurement Units

To determine the translational and rotational velocity of Ariana during different phases such as landing, and power ascent, IMU are utilized. Table 10.9 compares the performance of probable IMUs that can be used on Ariana. Similarly, the scoring algorithm is based on weight, power consumption and bias of the device. Hence, HG5700 is the selected IMU for Ariana.

Table 10.9: List of Inertial Measurement Units [40–43]

Name	Manufacturer	Weight (kg)	Power (W)	Range (deg/s)	Bias(deg/hr)	Scale factor (ppm)	Score
HG5700	Honeywell	1.36	10	1074	0.01	-	91
LN- 200S	Northrop Grumman	750	12	1000	0.1	300	85
ASTRIX-200	Airbus	12.7	6.5 per on channel	5	0.0005	30	73
CIRUS	L3	15.4	40	30	0.0003	35	65

LiDAR Sensors

Ariana uses LiDARs to scan the Moon's surface with 3D output while planning the landing phase on the different lunar zones. The sensor's result is a 3D map which helps Ariana to distinguish its landing point status. This sensor informs the spacecraft about the distortion, hazards, and depth of the landing point [44]. Also, LiDAR increases the landing precision by informing the control system the exact positions to land with high accuracy [45]. Autonomous Landing and Hazard Avoidance (ALHAT) LiDAR fits the spacecraft needs because it contains all types of suitable LiDARs in it which can be used in different phases of landing. The output of this system will be used in landing legs adjusting system too. Meanwhile, this sensor is used as a docking camera.

10.3 ADCS in Initiation and Assembly Configuration

As the RFP requested, the DSG assembly process requires both separated modules (PM and PLM) to have the attitude control ability separately during its premission phase as it is crucial for docking and transition orbits. The ADCS of both modules is discussed in the following.

10.3.1 PM

Actuator

In order to fulfill the required PM'S ADCS, 4 R-40B, 4 kN radial thrusters are selected and placed on the body. Considering the fact that minimum required moment for this module is much lower than the assembled vehicle, these additional thrusters are coupled with those installed at the bottom of the vehicle to be used just for required maneuvers such as docking and reorientation of the body during the assembly process. This configuration was depicted before in figure 10.3.

Sensors

Ariana's primary sensors are placed on the propulsion module. Likewise, additional sensors are not necessary for this module while it is operating separately.

10.3.2 Habitat and PLM

Actuator

Looking for proper sized actuators with a thrust of about 300 N of force leads to the choice of cold gas thruster for the habitat SM. Table 10.10 indicates the performance characteristics of the suitable thruster.

Table 10.10: Characteristics of the Chosen Thruster for Habitat Module [32]

Name	Manufacturer	Thrust (N)	ISP (s)	Life Cycle	Thruster Mass (kg)	Power (W)
58-126	Moog	266	70	10000	0.181	30

Configuration of the ADCS thrusters set is shown in figure 10.5. It can also be used in the ejection process in order to land the crew safely on the Moon's surface.

Sensors

Similar to the thrusters, a separate set of sensors is needed on the SM in order to measure the navigational state of the system for both the assembly and eject process. The breakdown of this set is listed in table 10.11. Sun sensors are placed in three positions on the SM at every 120 degrees while, the installation angle for the star sensors is 180 degree. Also, LiDARs utilized in both docking and landing (in the eject process) are placed in the top and bottom view.

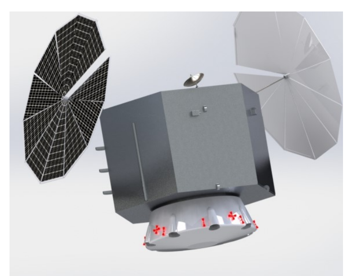


Figure 10.5: Thrusters Configuration

Table 10.11: SM Sensors

Type	Number
Star Sensor	2
Sun Sensor	3
IMU	1
LiDAR	2

10.4 Breakdown

A breakdown of ADCS is provided in table 10.12, 10.13 ,and 10.14.

Table 10.12: Maneuvers Breakdown for Assembled Vehicle

Maneuver	**Duration (s)**	**Mass Usage (kg)]**
Slews	62	179
Descent	70	473
Ascent	94	230
Docking	15	21
Total (15 % Margin)	**287**	**1039**

Table 10.13: Maneuvers Breakdown for SM

Maneuver	**Duration (s)]**	**Mass Usage (kg)**
Slews	20	57
Eject	40	116
Docking	15	11
Total (15 % Margin)	**87**	**212**

Table 10.14: Mass Breakdown of the ADCS Parts

Actuator	**Number**	**Mass (kg)**
R-40B Thruster	36	493.2
Cold Gas Thruster	32	5.8
Sensors	20	20

10.5 Intelligence and Autonomy

The control system of the vehicle is designed to learn, improve its performance, and correct its mistakes by measuring the system's state online. Also, it is designed to keep the desired performance in spite of any damage or misbehavior of any component. This helps Ariana to operate and accomplish its mission almost perfectly even in critical situations. Figure 10.6 explains the performance of the control systems available in typical vehicles. This method consisting of gas thrusters controller and various types of sensors generates force/torque by expelling cold or hot gas under pressure.

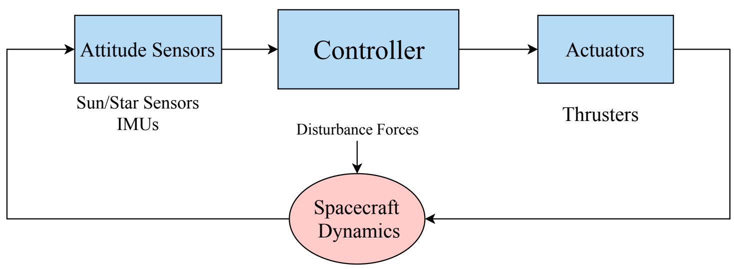


Figure 10.6: Typical ADCS Logic [46]

In many cases, the high level of performance and capabilities demanded from the control system cannot be met by simple traditional control schemes. Also, all subsystems of ADCS have to function continuously and properly for a successful operation of the mission over a long period of time, often more than 10 years. Any anomaly could make the attitude error build up fast, put the spacecraft out of control and discontinue the services performed by the spacecraft. It is, therefore important that anomalies are detected and diagnosed, and crucial actions are taken almost immediately, before the spacecraft goes out of control. Thus, with the integration of artificial intelligence and automatic control techniques , many benefits such as 1. improved performance through learning 2. adaptation, and self-modification under uncertain and changing environments 3. more on-board autonomous spacecraft control operations and 4. fault tolerant performance is viable in order to achieve a high degree of mission success and uninterrupted operation. Hence, an intelligent system is applied to Ariana's control systems to be endowed with the above preferences. Intelligent control systems infer, make decisions under supervision, cope with unexpected problems, and optimize mission resources, under critical real-time constraint. An architecture of intelligent ADCS is shown in Figure 10.7.

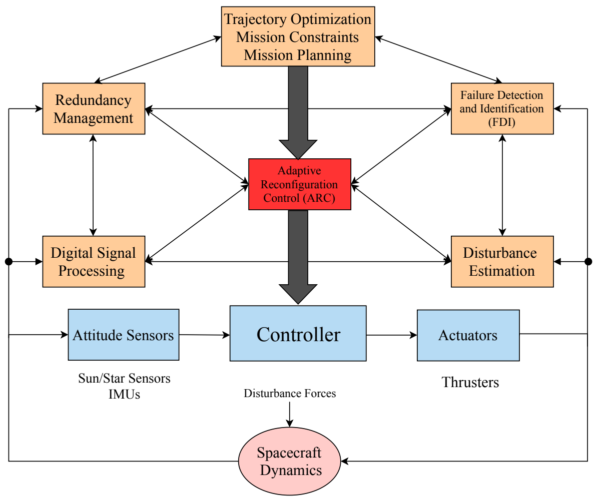


Figure 10.7: Intelligent ADCS Logic [47] [46]

The lower controller provides the routine, low-level closed-loop control, to keep the spacecraft in the desired state despite disturbance forces by generating opposing thrust/torque through the actuators while, on the other hand, the upper controller provides the higher level control by monitoring and analyzing the state of the system measured by the sensors which evaluate the performance of the controller and actuators.

The high-level controller:

1) automatically arrives at appropriate control strategy and finetunes the controller for improved performance.

2) learns unknown, altering features and parameters of the system, and adapts to the current conditions.

3) enables the control system to be fault-tolerant (detects, identifies and isolate failures) in real-time and reconfigures the controller, sensors and actuators system to enable continued operation.

4) copes with and reacts to anomalous, abnormal and unanticipated or rare situations by making appropriate decisions under the human supervision in real-time to modify the controller.

As a conclusion, advantages of a intelligent ADCS over a typical one are listed.

- Self-modification and Adaptability
- Autonomy
- Fault Tolerant Performance
- Self-learning
- Real-time Operation
- Progressive Reasoning

Human Habitation, ECLSS and Safety

Human comfortability and safety is a fundamental issue in crewed space missions. The crew will spend the most critical time of mission in the habitat and make momentous decisions. So, it is essential to guarantee both success and safety. For the Moon crewed missions, the design parameters are consistency, reliability, reusability, and cost reduction. The ECLSS subsystems must be responsible for controlling the air pressure and oxygen level, water supply, and waste management. Moreover, it is necessary to think about the matter of situational awareness of a CM, when they may be on DSG for several days and then want to be on a spacecraft and experience high acceleration although they have been on approximately zero gravity environment. The following section will discuss the process of selecting habitat between already existing concepts. In the next section, the trade study between ECLSS methods will be discussed, and the sizing of ECLSS subsystems will be covered.

11.1 Habitat Selection

Firstly, it is of paramount importance to define the subsystem level requirements that derivate from the RFP requirements (section 2.1).

11.1.1 Subsystem Level Requirements Definition

1. Mission duration: According to the RFP ECLSS and habitat should be sized for at least 24 hours on the Moon surface. Ariana has been sized to support crew for six days on the Moon surface without resupply from Moon resources. In chapter 8 the TOF from DSG to the surface and vice versa decided as 24 hours.

Moreover, the time that needs to be on the Moon's surface to have a proper maneuvering ΔV is at least 6 Earth days. Therefore, the ECLSS and habitat must be able to support CM for an average time of 180 hours or 7.5 Earth days. Besides, it is enough for an emergency and ejection process.

The total duration for all missions based on the Life-Cycle (LC) analysis in chapter 16 is ten missions before the overhaul. Thence, it is appropriate to assume that the total crewed mission is 10 in the vehicle life cycle (with considering extended lifetime after the overhaul).

2. Habitable volume: Based on table in [48] the habitable volume for each CM, that is dependence on mission duration, is between the optimal limit of $6m^3$ and performance limit of about $2m^3$ for this eight-day mission. The habitable volume should be able to keep some equipment like CM seat,food storage and galley, lavatory, controlling sensors and reactors, etc. These expected to be less than $5m^3$. Accordingly, the total adequate habitat volume is $21m^3$ as average.

3. Crew cabin starts to pressurized after assembling and inflating on DSG.

4. Shuttle-like crew cabin atmosphere is $101.35 \pm 1.38kPa$ total pressure.

5. The EVA system is assumed to be two EVAs by 3-person in each mission for about 8 hours.

6. According to the reusability concept, the crew cabin is not depressurized itself. An airlock is placed on the habitat module due to the number of depressurizing constraints.

7. Spacesuits support for 4 CM with oxygen and vent-loop connections.

8. For the matter of manual controlling the vehicle, the habitat has appropriate windows. Also, it is necessary for CM psychological health.

11.1.2 Habitat Trade Study

It is preferred to use the technologies that reached to high maturity or exist in the NASA portfolio. As it is expected, the utilization of inflatable habitats will be prevalent to build deep space basements. Therefore, the trade study has been done between the available habitats, especially which are more appropriate for the mission (table 11.1). The process of trade study has been started by assigning an importance factor to each comparable parameter. Then, each parameter gained a score on a 1 to 4 scale. A higher score represents more desirable characteristics which were multiplied by the importance factor and summed to get the final score.

Table 11.1: Habitat Trade Study

Parameter	Importance Factor	CST-100	Orion	Dragon II	BEAM	Federation
Mass	35%	2	1	3	4	3
Volume	10%	2	2	1	3	4
TRL	15%	3	3	2	4	2
Power	25%	2	1	3	4	2
Cost	15%	2	3	1	1	4
Score		**2.15**	**1.7**	**2.35**	**3.45**	**2.85**

Based on table 11.1 the BEAM habitat selected for the mission (for more information see table 11.3). It is an inflatable habitat that is currently being tested on ISS. However, it needs to be modified for a lunar lander configuration. The expected characteristics for the Modified BEAM (M-BEAM) are as follows:

1. The habitable volume should be $21m^3$; it means 25% is increasing the volume. The expected dimensions for the M-BEAM are presented in table11.2.

Table 11.2: Characteristics of BEAM and Expected M-BEAM

Characteristic	BEAM	M-BEAM (expected)
Mass (mt)	1.4	Less than 2
Length (packed) (m)	2.16	Less than 3
Length (unpacked) (m)	4.01	Less than 5
Diameter (packed) (m)	2.36	Less than 3
Diameter (unpacked) (m)	3.23	Less than 4

2. The BEAM expected lifetime is about 4 years, but the M-BEAM would be able to operate for about 10 years.

3. Proper windows arranged to satisfy the vision requirement of CM.

4. It is able to be connected to an inflatable airlock such as Advanced Inflatable Airlock (AIA), Minimalistic Advanced Soft Hatch (MASH), or Lightweight External Inflatable Airlock (LEIA). All of these are developing to test and utilize in future missions. In addition, it is compatible with testing NASA concepts on lightweight, inflatable, and stowable airlock developing to be staged from small spacecraft and capsules [49].

5. The structure of M-BEAM supports the maximum acceleration of about 1g. It has the proper shielding against radiation and meteors impact.

Table 11.3: Habitats More Information

Habitat	CST-100	Orion	Dragon II	BEAM	Federation
Mass	13	14	9.5	1.4	9.5
Volume	11	11	9.3	16	18
TRL (average)	NA	6	7	9	NA
Power	NA	11.2	4	1	NA
Cost FY17	1.58	0.97	2.69	2.3	0.71
Number of CM	7	4	7	NA	4-6

Figure 11.1: Schematic of M-BEAM

11.2 ECLSS

The basic requirements of ECLSS are presented in subsection 11.1.1. In the following section, the process of sizing the ECLSS is covered, and then the effect of aborting system on habitat configuration is discussed.

11.2.1 ECLSS Sizing

Due to the uncertainties about technologies and requirements of mission in the design process, the high number of iterations is needed to set the design on the baseline. So, In the sizing procedure, the European tool for automated Scaling of Life Support Systems (SCALISS[1] tool-build 1.1.160418) is utilized [50]. In the following table, the input parameters and assumptions to run the tool is described. Moreover, in table 11.4, the summary results of the Advanced Life Support System Evaluator (ALISSE) is demonstrated in table 11.5. (For further information see figure 11.1). As it is shown in table 11.5 the total mass of ECLSS is 5193 kg and required power is about 3.71 kW . According to the previous experience of human presence on the Moon surface, one CM is selected to serve as a command module pilot. Therefore, in first few missions he/she shall be present inside the vehicle even when the other three CMs get out of the vehicle to do an EVA, but after the Moon base completed, EVA condition will be provided for all of them. The spacesuits for lunar surface missions will be Exploration Extravehicular Mobility Unit with Lunar kit (xEMU-L) [51], that equipped with dust protection systems and can able to meet EVA requirements on the Moon surface.

Table 11.4: Input Parameters to SCALISS

Parameter	Value	Unit
Crew Size	4	#
Mission Duration	75	Days
Mission Location	Lunar Surface	-
Modules Number	1	#
Module Volume	21	m^3
Module lLeakage	0.1	%vol/day
Atmospheric Total Pressure	101.3	kPa
Oxygen Partial pressure	21.3	kPa
EVA per Mission	2	#
CM per EVA	3	#
EVA Duration	8	hour
Airlock	Yes	-
Airlock Volume	4	m^3
Airlock Recycle Pump	Yes	-
Shower Included	No	-
Laundry Included	No	-
Regenerative CO2 Removal	Yes	-
O2 Production	No	-
Particle Monitoring	Yes	-
Microbial Monitoring	Yes	-
Crop Production	No	-
Urine Recovery	No	-
Waste Water Overboard	Yes	-

Ejection System

The ejection system should be as light as possible so, some consumables that are mentioned in table 11.5 will be detached. The most massive item includes the 3.2 mt of water on habitat. Accordingly, two water tanks should exist in habitat, the main tank for water which can be detached, and a reserve tank for in case of emergency.

[1]From ESA-box website by “Giorgio Boscheri” permission.

11.3 Service Module Sizing

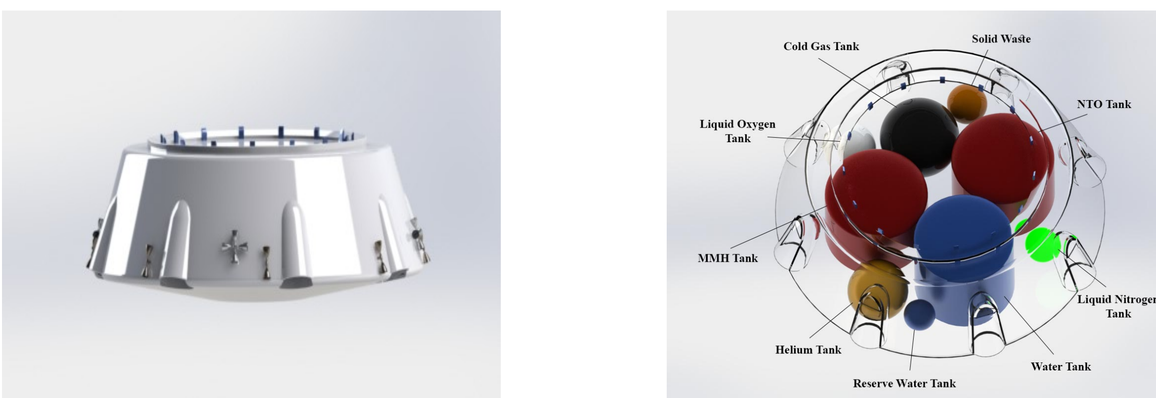
The SM of the habitat will contain subsystems of ECLSS, power, control and propulsion engines and propellant tanks. So, based on table figure 11.1 the volumes that need for O_2, N_2 and water storages, is calculated as 2.634 m^3 (see table 11.6). and volume for other components is shown in table 11.7. So the total required volume is 8.194 m^3, consequently the height of SM is 1.5 m and diameter is 5 m. The SM's components and side view is shown in figure 11.2.

Table 11.5: Input Parameters to ALiSSE

Model	**ALiSSE Criteria**	
	Mass (kg)	**Power (kW)**
Air Management System	711.052	0.971
EVA Support	283.882	2.64
Food Management	164.314	0
Crew Accommodation system	149.455	0
Thermal Management System	186.506	0.005
Waste Management (solid)	120.2	0
Water Management	2208.915	0.092
Total	**3824.324**	**3.708**

Table 11.6: Consumables Mass and Volume

Consumables	**Mass** (kg)	**Volume** (m^3)
N_2	20.5	0.025
O_2	256.7	0.224
Water	2385	2.385
Total	**2662.2**	**2.634**



(a) SM Side View

(b) SM Components

Figure 11.2: SM Sizing

Table 11.7: SM Components Specifications

Propellant Mass and Volume		
Parameter	MMH	NTO
Mass (kg)	1865.6	3078.3
Mass with 10% Margin (kg)	2052.1	3386.0
Volume (m^3)	2.34	2.33
Pressurant Tanks		
Pinitial (Mpa)	34.4	
Pfinal (Mpa)	3.8	
Propellant Tanks		
Pmax (Mpa)	3.3	
Ullage	5%	
V_{MMH} (m^3)	2.46	
V_{NTO} (m^3)	2.45	
Helium		
ρ @ T=323 K (m^3/kg)	44.79	
Mass (kg)	31.9	
Volume (m^3)	0.71	
Cold Gas		
Mass (kg)	150	
Volume (m^3)	0.18	

Thermal

Operating of Ariana on the Moon surface necessitates performing a suitable thermal condition for all sections considering the operational environment. Thermal environment of the Moon has been studied in section 6.2.2 and in this chapter the thermal requirements of Ariana elements have been analyzed. Due to these considerations, adequate thermal control methods have been selected and designed.

12.1 Thermal Control Systems (TCS)

The objective of TCS is to maintain all Ariana elements in their required temperature so that operate sufficiently by using passive and active methods.

12.1.1 Thermal Requirements

Typical thermal ranges requirements for Ariana's components during the mission to survive/operate that are summarized in 12.1 shows that some elements like propellant tanks or batteries have a very narrow temperature range for operation and would be highly sensitive to large temperature variation.

Table 12.1: Typical Thermal Requirements for Components

Subystem	**Component**	**Typical Temperature Range (°C)**	
		Operational	**Survival**
Power	Batteries	10 to 25	5 to 45
	Solar Panels	-150 to 110	-200 to 130
	Power Box Baseplates	-10 to 50	-20 to 60
Control	CMG / IMU	5 to 45	-20 to 50
	Sensors	0 to 40	-20 to 50
Propulsion	Liquid Hydrogen	-259 to -253	-259 to -253
	Liquid Oxygen	-218 to -183	-218 to -183
	Monomethyl hydrazine	-52 to 87	-52 to 87
	Dinitrogen tetroxide	-9 to 21	-9 to 21
Communication	Antennas	-100 to 100	-120 to 120
	Onboard Computer	-20 to 50	-55 to 70
Habitat	Crew	18 to 27	15 to 30

12.1.2 Passive Systems

Due to no power consumption in passive thermal control systems that causes more simplicity, lower cost and lower mass (comparing to active TCS) they are so favorable between candidates in design procedure.

- Multi-Layer-Insulation (MLI): most common passive thermal control method is MLI that covers outer spacecraft surface to minimize heat loss/gain from the desired component it, also increases heat capacity of the spacecraft to be resistant against thermal cycling.

  Structure of MLI that covers the outer surface of the spacecraft is made of 30 layers of kapton/mylar reflectors with Dacron netting as interior with 2.6 kg/m^2 density and with 0.2 W/m^2K thermal conductance in 300 K. coated with aluminized kapton with an absorbance value of α=0.46 and emittance value of ϵ=0.86.[52] propellant and oxidizer tanks also will be covered with a Foam/VD-MLI combination coating that performs several weeks storage condition for cryogenic fuels in the lunar environment. This insulation structure by foam, Double Layer Aluminized Mylar with Dacron net spacers in multiple segments [53]. For our 358 m^2 area of tanks this method of insulation has 520 kg mass.

- Ablative Materials : Carbon fibre reinforced carbon (CFRC) has been used as covering material in case of payload and spacecraft protection from engine exhaust that reaches to 1100 K. (section 12.1.4)
- Radiators: Radiators will attach to structure sides and reject heat from baseplates of electronics and batteries to keep them at operational condition with a specific heat rejection capacity of 55 W/kg

12.1.3 Active systems

Although it is desirable to design systems completely passive, the existence of Crew members in some missions and usage of Cryogenic fuel with narrow temperature range necessitates to utilize active TCS simultaneously beside presented passive systems. That would have remarkable benefits such as reducing risk of thermal damages, rapid response for temperature control and more accuracy than passive systems.

- Selected habitat equipped with embedded active and passive TCS including water cooling pipes, heat exchangers and radiators. Based on section 11.3 an estimation of power and weight of TCS presented in the table 12.2
- Temperature Sensors: to measure the temperature of components for controlling thermal condition ESCC Epoxy coated thermistors have been used.
- Cryocoolers: Cryocooler performance and reliability are continually improving. Consequently, they are frequently implemented in space programs. Joule-Tompson cryocooler utilized to maintain propellant in 20 K with a specific refrigeration power of 300 W/W and also Stirling coolers used for oxidizer cooling in 70 K with specific refrigeration power of 60 W/W. [54]
- Heat Pipes: A closed loop of Aluminum Ammonia pipes (VHCP) [1] over widely varying powers and sink temperatures provides a ±1-2 °C temperature control with limited power consumption and low density [55].
- Electroheaters: Patch heaters organized from electrical-resistance circuit sandwiched between two Kapton layers with a heat generation capacity of 6 W/cm^2.

[1] Variable Conductance Heat Pipes

12.1.4 Blast Protection and Payload Safe-Distance

At the end of unloading procedure it should be checked that if heat and the force exerted on unloaded payload on Moon surface from exhaust jet has any destructive effect on payload stability and thermal condition. From the thermal point of view, in ascending phase, unloaded payload on the surface that is in the vicinity of Engine Exhaust with 2500 W/cm2 heat flux (about 20 seconds) will be covered in ablative materials. In the procedure of landing the vehicle passes through its exhaust, and in low attitudes exhaust of the engine reflects to the spacecraft. That makes it essential to reinforce subjected areas with ablative coatings (section 12.1.2). These materials are resistant to a 5000 W/Cm^2 and could protect the payload from thermal damages.

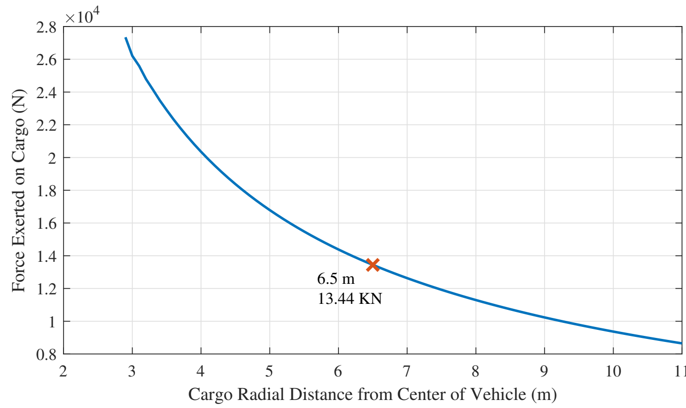
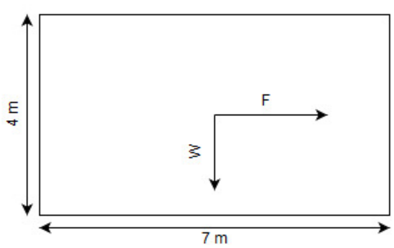


Figure 12.1: Force Exerted on Cargo by Nozzle Jet

Force effect on payload also modeled with [56] and modified for this mission by revising propulsion system properties and configuration. The results of the simulation has been validated by [57].

The figure 12.1 shows that the maximum force that applied on cargo is approximately 13.4 KN. For finding the stability margin of unloaded cargo the following calculations are applied.



$$3.5 \times W = F \times 2 \rightarrow W = 7.65KN$$

$$Mass_{min} = 4.7mt$$

Figure 12.2: Free Diagram of Forces on Cargo

The minimum mass of the cargo that remains stable during ascent with dimensions that fully filled cargo section (worst case) is 4.7 mt. So, if there is a payload lighter than this limit, it should be able to move farther to safe-distance according to figure 12.2 by an existing rover in payload or Moon surface. Otherwise, if there is no capability for the payload to be displaced, the mass limit must be considered.

12.1.5 Summary

Based on previous assumptions and calculations, the estimated needed power and structural weight of TCS has been presented in table 12.2. Considering that weight of MLI coatings of the outer surface of spacecraft also counted in Stuffed Whipple housing of structure (section 14.4)

Table 12.2: Summary of Thermal Control Systems properties

Thermal Control System	Type	Mass (kg)	Power (W)
Habitat TCS	Active & Passive	180	600
MLI	Passive	970	-
Foam/VD-MLI	Passive	520	-
Radiator	Passive	190	-
Cryocoolers	Active	250	310
Heat Pipes	Active	20	90
Electroheaters	Active	10	100
Total	**Active & Passive**	**1960 kg**	**1100 W**

Power

The power system is a vital part of Ariana, since its significant role in the functionality of subsystems and spacecraft's life. The Electrical Power System (EPS) comprising the power source, energy storage, power distribution and control systems to supply required electrical energy for all subsystems and payloads as well as possible.

According to the RFP, the vehicle shall be able to land everywhere on Moon's surface; as a result, it might be in eclipse for a long time. Hence, the vehicle must have a reliable source of power for this condition on the Moon. Besides, the most important objective of the spacecraft is its reusability, so, the power system components must either have a high life cycle or be accessible for maintenance. The power system sized by the total power requirement of subsystems at the End of Life (EOL) with pick power consumption to ensure about Ariana's performance during its lifetime.

13.1 Power Source

The primary power source of the EPS must supply Pick power demand for all loads as shown in table 13.1. On the other hands, the power storage must be fully charged for the most challenging part of the mission which is the lunar surface operation. The most common power generators are listed in table 13.2 by their characteristic data. Ariana must rely on a safe and continues power source due to crew delivery missions. The trade study has been utilized for selecting the best possible choice to meet the design criteria. As demonstrated in table 13.3, solar photovoltaic has been chosen because of the mature technology, low cost, lightweight, high lifetime, and high level of safety. Also, the solar rays generate enough power due to reasonable operational region distance from the sun. The primary fuel cell is in contrast to reusability concept, Also, Multi-Mission Radioisotope Thermoelectric Generator (MMRTG) is expensive, and its hazards for a human cannot be negligible.

Table 13.1: Power Budget

Subsystem	Capture to DSG (kW)	DSG to Moon (kW)	Moon's surface (kW)	Standby (kW)
Payload (Cargo)	0	0	4.00	0
ECLSS (Crew)	3.70	3.70	3.70	3.70
Communication	0.08	0.08	0.08	0.08
Propulsion	0.14	0.14	0.10	0.10
Power	0.10	0.10	0.10	0.10
Thermal	0.80	0.80	0.80	0.80
Attitude and Control	2.01	2.30	0.10	0.10
Total Cargo Mode	**-**	**3.42**	**5.18**	**1.18**
Total Crew Mode	**6.92**	**7.12**	**4.88**	**4.88**

Table 13.2: Power Source Comparision

Design Parameter	Solar photovoltaic	KiloPower	MMRTG	Fuel Cell
Power range (kW)	0.2-300	1-10	0.11	0.2-50
Power Density (W/kg)	25-200	2.5-6.5 [58]	2.5	275
Cost	Low	High	High	Medium
Maneuverability	Low	High	High	High
Dgradation	Medium	Low	Low	Medium
Storage for eclipse	Yes	No	No	No
Safty analisis	Minimal	Extensive	Medium	Medium

Table 13.3: Power Source Trade Study

Design Criteria	Weight (0.1-1)	Score			
		Solar Photovoltaic	KiloPower	MMRTG	Fuel Cell
Power density	0.8	4	2	1	5
Safety	1	5	2	2	4
Solar Independency	0.6	1	5	5	4
Cost (low)	0.7	5	1	2	3
Reuseability	1	4	5	5	1
TRL (high)	0.6	5	1	3	3
Final Score		**19.3**	**12.9**	**14**	**15.3**

13.1.1 Solar Array

First of all, the type of solar cell must be determined for the solar array sizing. The main objective of solar array selection is low mass and high efficiency. To satisfy these objectives, the best current State of Practice (SOP), Triple Junction (TJ) GaAs solar cell with an efficiency of about 30%, has been selected. However, it is obvious that until 2028, the efficiency of advanced solar cells like 4 and 5-junction cells, would be improved, at the condition of high or low temperature, irradiation, and radiation. Recently space missions such as Insight and Phoenix, as well as future ones, have been used ATK UlrtraFlex [59] fold-out solar panel configuration, which replaced with the abolished rigid configuration. It is more efficient in mass and volume than others by stowing in a much smaller footprint on the top of cargo module and is compatible with the chosen solar cells [20]. This panel configuration benefits from stowing in ascent and descent phases to reduce the Dynamic loads, moerover prevent solar arrays being dusty which increases lifetime. Table 13.4 represents solar panel's technical data. The solar panel has been designed to provide max power of 7.12 (kW) for loads with the margin of 30%, besides keeping power storage full-charge.

13.2 Energy Storage

Energy storage supplies auxiliary power during the eclipse and pick-power demand. One of the challenging parts of the mission is the possibility of a long eclipse on the moon's surface. Hence, the power storage is sized [60] to provide sufficient power during 156 hours (table 8.8) of the lunar surface operation, considering the margin for the possibility of the extended eclipse into the flight path. The secondary power choice to meet Ariana's criteria are

Table 13.4: Solar Panel Charactristics

Parameter	Value
Efficiency	30%
Number of Wings	2
Total Mass (kg)	125
ATK's Ultraflex power density (W/kg)	150
Each wing Area (m^2)	25
Each Wing Diameter (m)	8
Thermal Range	-140°C to 150°C

listed in table 13.5.[61]

Table 13.5: Secondary Power Comparision

Design Parameter	NiH_2	$Liion$ (SOP)	Advanced $Liion$	Capacitor
Specific Energy (Wh/kg)	30	100-150	>250	6
Operating Temperature	-20°C to 40°C	-20°C to 40°C	-40°C to 65°C	less than -40°C to 60°C
Cycle Life	>50000	<1000	>1000	>1000000
Cost	Low	Medium	high	Medium

Future space missions require a high energy density, safe, compact, long life cycle, and more extensive temperature range power storage. For instance, Nickel-hydrogen (NiH_2) batteries will no longer be available for the next decadal planetary missions, since their production is being phased out [61]. On the other hands, Regenerative Fuel Cells (RFC) are under development by NASA, as the consistent and reliable power storage to meet the demand of both manned and large-scale robotic missions for future Moon and Mars surface missions, by utilizing from In-Suit Resources (ISRU) as a reactant source for RFC [62]. NASA would be the first user of this new technology, and will likely have to pay all the costs associated with its development [63]. Moreover, its TRL is not high enough to assure it will be available until the lunch date. Currently, the best-proven technology candidate is Li-ion battery due to their high energy density and technology trend progress. SOP Li-ion is massive and bulky for high power supplement missions. Its specific energy storage would be improved by significant investment for next-generation science mission concepts, later than 2025 [61]. Therefore, advanced Li-ion battery with the power density of 300 (kWh/kg) has been selected as Ariana's main power storage.

The supercapacitor is capable of enduring the harsh environments and provide very high power density about 1 to 15 (kW/kg) in a few seconds with high life cycle. It is proper for repetitive, high power demand in a short time. The supercapacitor is utilized [64] for loading and unloading of payload which last about 10 minutes in each cargo mission. Table 13.6 demonstrate energy storage mass beakdown.

Table 13.6: Energy Storage Mass Breakdown

Component	**Mass** (kg)
Propulsion system's Battery	1200
Service madual's battery	2000
Supercapacitor (in SM)	150
Total Mass	3350

13.3 EPS Configuration

Assembly strategy determine the arrengmment of EPS. Hence, it's required to allocate 1.2 m.t of total battery mass in PM. And, rest of the vehicle which contain SM, habitat, and cargo, powers with solar array and battery to complete assembly procedure.

EPS required management systems to distribute power. Also, the control system is utilized to orient solar panels in the best position. Besides, prevent battery overcharging by dissipating excess power.

Structure

14.1 Structure Requirements

The purpose is to satisfy all of the mission requirements related to the structure while minimizing the vehicle's structural mass and volume to decrease the propellant cost. Table 14.1 includes the list of these main requirements.

Table 14.1: Structure Subsystem Level Requirements

No.	Requirement
1	Ariana's structure shall provide connectivity and support all of the other subsystems and protect them in space harsh environment.
2	The vehicle structure shall also be able to withstand launch loads and any forces or torques generated by any of the subsystems, especially the propulsion system, the reaction control system, and the unloading mechanism.
3	In cargo mode, Ariana structure shall house and deliver at least 15 mt from DSG to Moon's surface and return 10 mt from the Moon to DSG

14.2 Material Selection

The selected material has a significant effect on the vehicle's mass and its production cost. Ariana utilizes both metal alloys and composite materials in its structure. Table 14.2 shows characteristics of studied materials.

Table 14.2: Characteristics of The Metal Alloys and Composite Materials [65]

Metal Alloys		Composite Materials	
Advantage	**Disadvantage**	**Advantage**	**Disadvantage**
. Lower cost . High heritage . Ease of production . Weldability	. Increasing structure weight . Sensitive to high temperature gradients	. Modified performance properties suitable for specific application . High specific strength . Stable properties in different environment	. High production cost . Production and assembly difficulties

Mechanical properties for some common metal alloys in space applications, are shown in table 14.3.

Table 14.3: Common Metal Alloys Mechanical Properties [66]

Material	Density (kg/m^3)	Young's Modulus,E (Gpa)	Yield Strength,S_y (Mpa)	Specific Strength (Sy/ρ)	Specific Stiffness (E/ρ)
Aluminum Alloys					
6061-T6	2710	67	276	0.102	0.024
7075-T6	2800	71	503	0.179	0.025
2090-T83	2590	79.3	504	0.195	0.030
Magnesium Alloys					
AZ31B	1770	44	221	0.125	0.025
AZ31B-H24	1770	44	269	0.152	0.025
Titanium Alloys					
Ti – 6AL 4V	4430	110	830	0.198	0.025
Beryllium Alloys					
S 65 A	2000	304	207	0.103	0.152

To decrease structural mass, the material with the highest structure efficiency has been chosen. The mass of the structural element decreases while the specific stiffness increases. Beryllium is eliminated because of its corrosion sensitivity. As can be inferred from table 14.3, steel, titanium, aluminum, and magnesium have approximately the same stiffness ratio, and in the case of withstanding torsion and compression loads, all of them show same behavior. But in the components like beams which their material index is proportional to the square root of Young's Modulus (E) ([65]), lighter materials like aluminum offers excellent weight reduction. Aluminum 2090-T83 will be used in Ariana's main frame structure and supporting beams. This material has high specific stiffness, low cost, and is easily weldable and shapeable. The landing legs are made from Titanium Ti6Al-4V, which have high temperature resistance to tolerate engine blast. Furthermore, Titanium has high stiffness and low weight. Al-alloy honeycomb is used in structures to hold and support instruments.

14.3 Main Frame Design and Analysis

In zero-gravity environment, the external forces exerted on the spacecraft are very small. The maximum loads during Ariana's life cycle are considered to design and size the vehicle structure. Generally, the highest dynamic loads occur during the deploying launch phase. The following table provides an estimation for launching acceleration based on the SLS mission Planner's Guide.

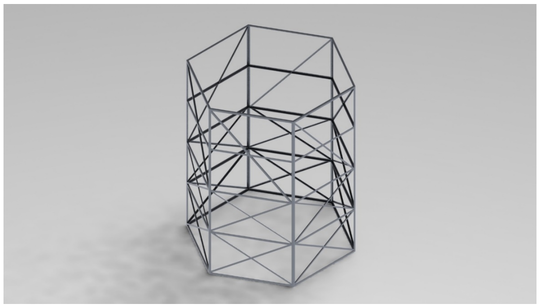


Figure 14.1: Ariana's Main Structure Frame

Table 14.4: Block 1B PPL/CPL Combined Load Factors [3]

Case	Liftoff	Liftoff/Transonic	Max Q	Core Stage Flight	MECO
Axial Acceleration (g)	+1.0	-2.0	-2.3	-3.5	-4.1
Lateral Acceleration (g)	1.5	2.0	2.0	0.5	0.5

Considering that Ariana has five RL10 CECE engines, the
structural tensions during maneuvers in deep space is even greater than SLS dynamic loads. In the most critical case, three engines will start simultaneously which exert about 200 kN force on the main structure.

Aluminum 2090-T83's yield strength is 504 MPa. To ensure that the designed structure can withstand 200kN force, an Finite Element Method (FEM) analysis by "SolidWorks 2018" has been done. Figure 14.2 shows the results of FEM simulation. The maximum stress in the structure by ignoring stress concentrations on the edges is 371.2 MPa. This value corresponds to the safety factor of about 1.36.

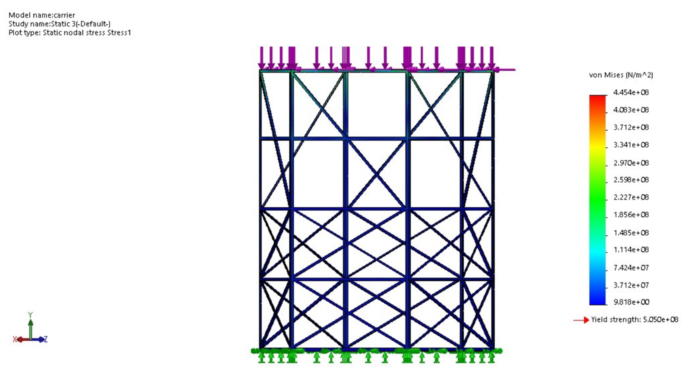


Figure 14.2: Simulation Results for the Loading Scenario

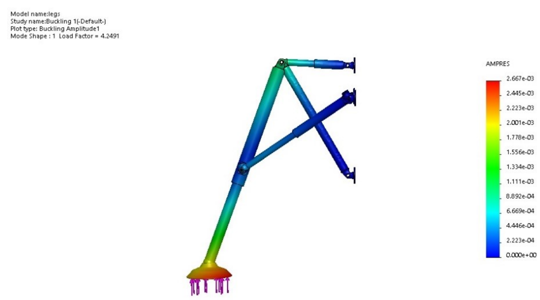
14.4 Housing and Meteoroid Shielding

Space missions require adequate shielding against Meteoroid and Orbital Debris (MOD) for increasing vehicle's survivability. MOD's impact is a function of vehicle's size and operational lifetime. Since, Ariana operates in deep space, the main concern is about meteoroid impact risk. Nextel/Kevlar Stuffed Whipple shielding [67] is used to protect the propulsion module. Also, a lighter Whipple shielding is used to protect cargo module.

14.5 Landing Legs Structural Analysis

In this section, a simple dynamic analysis is conducted to find the maximum force applied to the vehicle's legs while landing on the lunar surface. It is assumed that the lunar surface is flat and vehicle lands vertically on it. The maximum touchdown mass is 56 mt, Moon's gravity constant is assumed to be 1.625. The worst-case scenario is the moment that the vehicle hits the surface just with one leg. It is assumed that the maximum vertical landing velocity is 1 m/s, and it corresponds to an acceleration of 2 $\frac{m}{s^2}$, the impact time is assumed to be 0.5 seconds.

Although the value of 0.5 seconds is a rigorous assumption for landing time, it leads to a sound margin of safety for design. According to aforementioned assumptions, the maximum force is 200 kN. A FEM analysis is performed to ensure that the leg will tolerate the landing impact. The boundary conditions for this analysis is that the connection points of legs to main structure are fixed, and the force is exerted normal to the leg.The maximum stress in the leg components is approximately 360 Mpa by ignoring the stress concentration in the edges. Considering that the material yield strength is 827 Mpa (chapter 14.2), the safety factor would be 2.3. Furthermore, buckling analysis shows the landing leg would not be buckled by a safety factor of 4.25.



(a) Buckling Analysis

(b) Sterss Analysis

Figure 14.3: Landig Legs FEM Analysis

14.5.1 Fuel Tank Design

S/C functionality and fuel type drives the design objectives for propellant tank which include about 45 percent of the overall Ariana's volume. Besides, launch vehicle limited PLF dimensions and low density of the LH2, lead to minimum volume occupied by the fuel tanks[1].

Table 14.5: Maximum Needed Propellant Volume

Propellant	**Volume (m^3)**
Liquid hydrogen	170.9
Liquid oxygen	62.4

Two tank arrangement for the same vehicle's configuration are discussed below:

Separated Tank Arrangement (STA): In STA, there are seven cylindrical tanks with elliptical caps as shown in figure 14.4.

[1] 3 % ullage

Figure 14.4: STA Configuration

Annular Tank Arrangement (ATA): ATA consists of one cylindrical tank in the center for LO2 and an annular shaped tank for LH2 encompassing the central tank. Figure14.5. The annular tank is divided into six separate sections to minimize damage risk and mission failure.

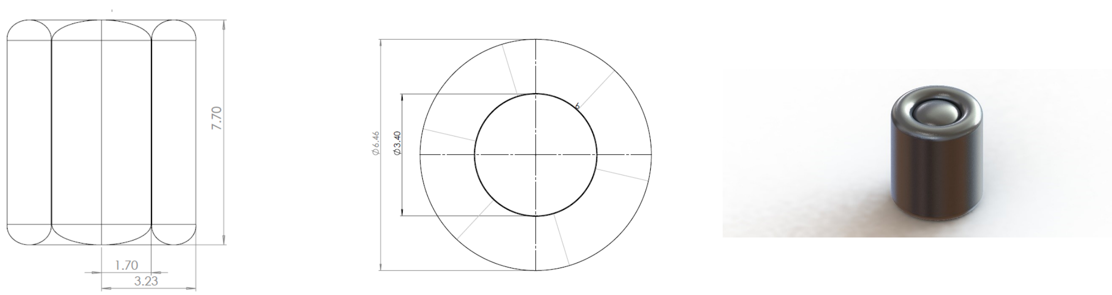


Figure 14.5: ATA Configuration

STA and ATA heights are compared for the same configuration in table 14.6.

Table 14.6: Comparison of ATA and STA Occupied Height

Tank Arrangement	**Height (m)**
STA	10.5
ATA	7.7

ATA offers 2.8 m smaller height because of removing unusable gaps between tanks in STA. Hence, the ATA has been selected. CFRC is being used for cryogenic fuel tanks to reduce mass. IM7, which is manufactured by Hexcel and is used widely in cryogenic tanks in NASA, has been chosen. Table 14.7 demonstrates this material properties.

Table 14.7: Properties of IM7 CFRC [68]

Fiber Type	**Tensile Strength** (Mpa)	**Tensile Modulus** (GPa)	**Strain**	**Density** (g/cm^3)	**Filament Diameter** (μm)
IM7	5300	275	1.8	1.77	5.2

Because of high strength-to-density and high stiffness-to-density ratios, using CFRC will result in a significant weight reduction for cryogenic tanks compared to metal alloys. Table 14.8 compares the propellant tank mass in

ATA for same amount of fuel that is designed for the factor of safety within 1.3 to 1.5.

Table 14.8: Comparison Between CFRC and Al Alloys Tank Mass

Propellant	CFRC (IM7)	AL Alloys
LH2 tank weight (kg) @ 0.3 Mpa	470	1490
LO2 tank weight (kg) @ 0.25 Mpa	156	247
Total Weight (kg)	**626**	**1737**

Accordingly, the vehicle has an ATA propellant tank made from CFRC (IM7). The wall thickness for both inner and outer tanks is 1mm and tanks are fitted and fixed in Hexagonal shape main structure.

14.6 A/ODCS Tank Design

Overall required propellant per mission for A/ODCS is 1039 kg (Table 10.12). The specifications of A/ODCS' propellant is indicated in table 14.9.

Four spherical CFRC tank is used for A/ODCS which are fixed in the bottom of main propellant tanks.

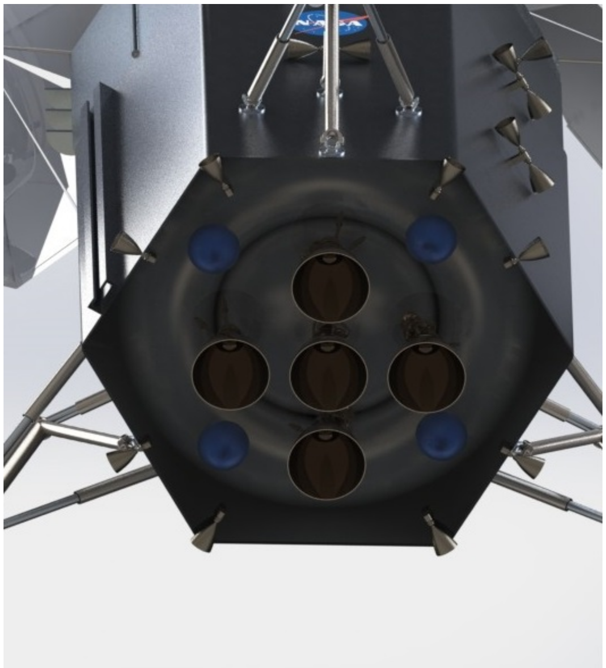


Figure 14.6: A/ODCS Tanks

Table 14.9: Propellant Mass & Volume Specifications

Propellant Mass and Volume		
Parameter	MMH	NTO
Mass (kg)	339.7	560.4
Mass with 10% Margin (kg)	377.4	622.7
Volume (m^3)	0.43	0.43
Pressurant Tanks		
$P_{initial}$ (Mpa)	34.4	
P_{final} (Mpa)	3.8	
Propellant Tanks		
P_{max} (Mpa)	3.3	
Ullage	5%	
V_{MMH} (m^3)	0.45	
V_{NTO} (m^3)	0.45	
Helium		
ρ @ T=323 K (m^3/kg)	44.79	
Mass (kg)	5.9	
V (m^3)	0.13	

Telemetry, Tracking and Command (TT&C)

The communication system design for Ariana aims to provide telemetry, tracking, and command between DSG, Earth and spacecraft wether in crew mode or cargo mode.It is noted that I-MARS is not designed for a scientific mission and high data rate would not be required to downlink. Although the typical data rate for downlink in deep-space mission is 10 kbps [26], downlink data rate for Ariana is determined to be 0.5 Mbps. The reason is the safety factor of the crewed mission, and the fact that essential information about crew must be transmitted.

To cover RFP's statement that vehicle should land anywhere on the lunar surface, especially, landing on the far side of the Moon, the communication system drives requirements and architecture for the mission which will be presented in following sections.

15.1 Driving Requirements

The main communication system requirements are listed in table 15.1:

Table 15.1: TT & C Requirements

No.	Reqiurement
1	The vehicle Shall be able to maintaine linkage between the Ground station and DSG.
2	Compatibility of communication system designed for I-MARS with the ground station and DSG relay station is necessary.
3	A secondary antenna for more coverage and emergency mode should be employed as redundant.

Maintaining stable communication in both modes is of the TT&C paramount importance. Since the I-MARS is the crewed mission, continuous communication for crew health status is required. Therefore, it would be highly required to uplink and downlink continuously to/from the vehicle in both modes.

15.2 TT&C Equipment

Due to subsystem-level requirements, two High Gain parabolic Antenna (HGA) on Ariana are required. One antenna is in the opposite direction of the other one to increase communication coverage. In case, one of the antennae fails, the other one would operate. Ariana will transmit and receive in X-bands because of lower bandwidth cost and higher link availability than Ka-band [69]. Besides, higher data rate would not be required for the mission. Also, X-band antennae are smaller and have better accuracy than S-band. Gimbals and motors for HGA are necessary to point accurately.

Ariana transmits and receives using Small Deep-Space Transponder (SDST) developed by General Dynamic and NASA's Jet Propulsion Laboratory (JPL). Power amplification for the downlink is accomplished by an X-band Solid-State Power Amplifier (SSPA) from General Dynamics. The SSPA is able to provide the required power for communication. There is an additional transponder and power amplifier on board as redundant. Additionally, there are two low-gain omnidirectional antennae at lower data rate operating in X-band, in case of rendezvous and backup for transmitting and receiving about 5 bps and 10 bps in emergency mode. One is placed on the propulsion module, and the other one is placed on cargo module in the opposite side for increasing communication coverage.

The power and mass budget for TT&C subsystem are shown in table 15.2.

Table 15.2: TT&C Components and Their Masses and Power Budget [26],[70]

Component	**Mass (kg)**	Power (W)
High Gain Antenna (x2)	15	0
Low Gain Antenna (x2)	5	0
Diplexer (x2)	1.2	0
Cables	5	0
HGA Gimbals	35	14
SDST (x2)	6.4	15.8
SSPA (x2)	3.5	50
Total	**71.1**	**79.8**

15.3 Ground Station

Deep Space Network (DSN) was selected as ground station due to its better performance in deep space mission and Apollo missions. DSN ground stations are placed around the Earth about 120° apart in California/USA, Madrid/Spain and Canberra/Australia. In 2025, the 70-meter antennae at all three locations will be decommissioned for communications and replaced with the 34-meter Beam Waveguide Antenna (BWG) antennae arrays[71]. Therfore, the new configuration of an antenna will be serviced for ground segment data transmission. The 34 m BWG is capable of uplink and downlink in X-Band [72]. The 34m BWG antenna characteristic is shown in table 15.3a.

15.4 DSG Communication Relay

The fact that communication window access is not available on all points of the lunar surface makes it difficult to meet the RFP stipulation of being able to land on any point on the Moon. The access link between Ariana and DSN is impossible in the far side of the Moon surface. Hence, for increasing possibility of a direct link between the spacecraft and DSN, DSG would operate as a relay station for communication between the spacecraft and DSN. The ESPIRIT and Power and Propulsion Element (PPE) module of DSG provide additional lunar communication[73], and the PPE module will be capable of providing X-Band [74]. The DSG communication system characteristics has been shown in table 15.3b. These are based on studies of the communication system of the DSG that is necessary to maintain a complete communication system link between the DSG and Visiting Vehicles [75].

15.4.1 DSG Moon Coverage

According to the type of NRHO, the presence of DSG would support almost full access window for the Moon surface but not entirely. Southern family NRHO in section 8.2 would not support full coverage of the Moon's surface especially regions close to north pole. As shown in figure 15.1 DSG on southern NRHO, could cover almost the 10% of the Moon surface in worst case.

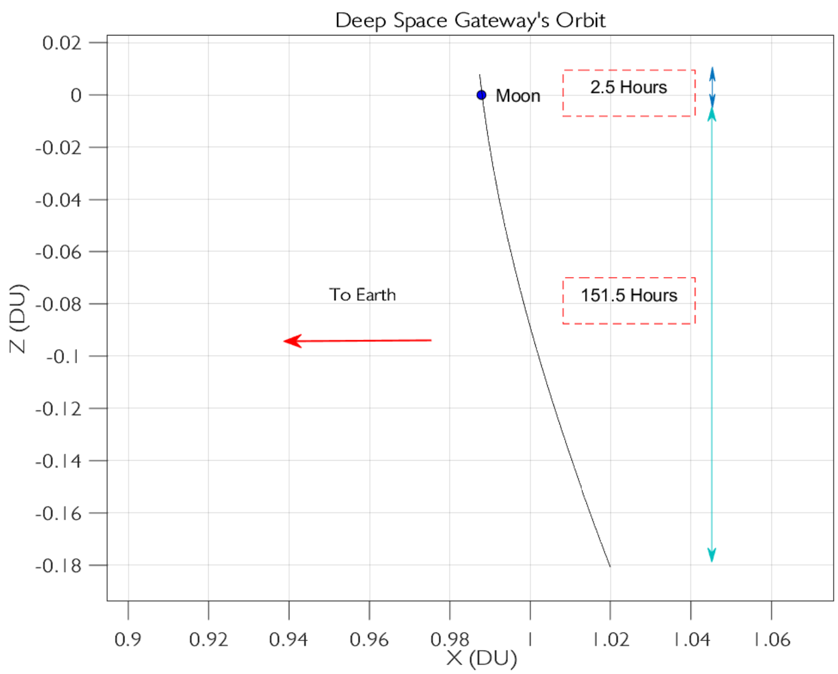


Figure 15.1: Side View of NRHO

15.4.2 DSG Earth coverage

The DSG is always in Earth sight, and there is no occultation for NRHO to communicate with ground stations [76].

Table 15.3: Ground Station and DSG Communication

(a) DSN characteristic [77]

BWG 34 m Characteristics	
Uplink frequency (MHz)	7145-7235
EIRP (dBW)	89.5-109.5
Downlink frequency (MHz)	8400-8500
Power Output (kW)	20
Uplink data rate (kbps)	256
Gain (dBi)	68.2

(b) DSG Characteristic

DSG 2 m Characteristics	
Uplink frequency (MHz)	7190-7235
EIRP (dBW)	39
Downlink frequency (MHz)	8450-8500
Power Output (kW)	0.065
Uplink data rate (kbps)	100
Gain (dBi)	35

15.5 Communication Architecture

Ariana architecture feasibility is demonstrated by the following strategy:

Ariana would communicate with DSN while landing on the near side of the Moon and communicate with DSG as a relay to the ground station while landing on the far side of the Moon. Access window for Ariana and DSG is illustrated during one NRHO peroid in figure 15.2 for the far side of the Moon. The access window gradually decreases from southern of far side to northern.

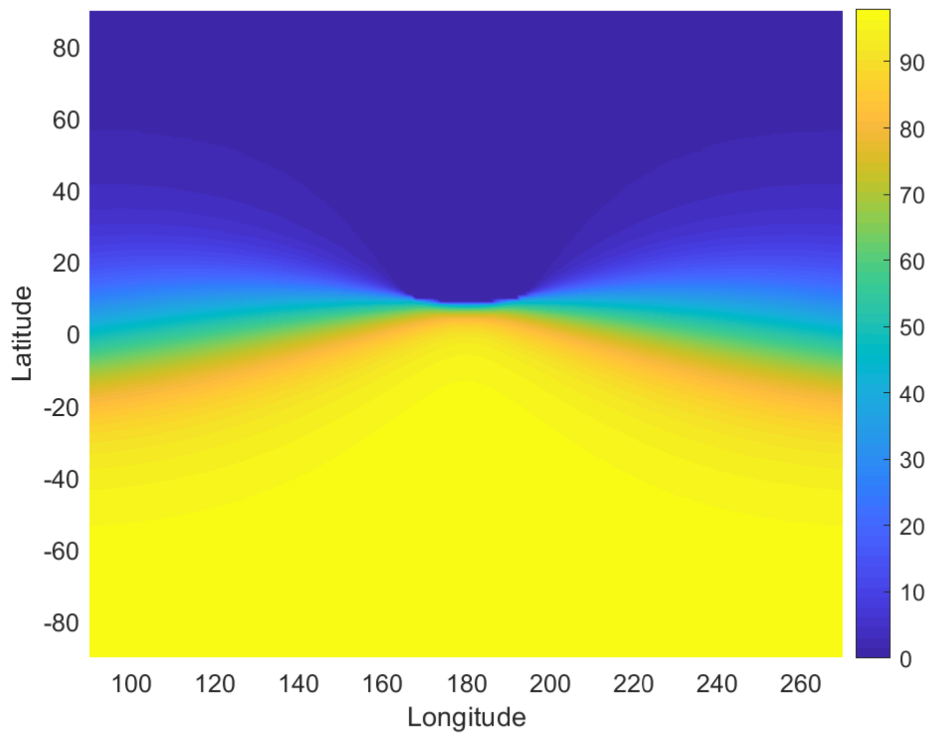


Figure 15.2: Ariana and DSG Access Window in The Far Side

Up to now, most of the lunar landers and lunar explorers are orbiting the Moon in a southern halo orbit, so, their most lunar surface coverage is southern regions. According to NRHO orientation, the link access opportunity time in southern family is up to 6.5 days in NRHO peroid. However, the link access opportunity time in northern regions is less than an hour. So, the relay station is required to achieve linkage in the far side while Ariana is operating.

In 2019, Chandrayaan-2 orbiter will be launched to lunar orbit. Also, the orbiter communication system is compatible with DSN and provide X-band frequency [78]. In the mid-to-late 2020s, NASA will provide lunar relay network for science and robotic exploration missions, especially for polar and far-side locations [79].

15.6 Link Budgets

Ariana's communication system is sized based on the value of the Radio Frequency (RF) transmitter power and antenna diameter. The downlink analysis is shown in table 15.4. In the worst case of downlink analysis, DSG lunar antenna have been sized into 1 diameter. The modulation used, is Qadrature Phase Shift Keying (QPSK) with 10^{-5} Bit Error Rate. Link margin have been set to at least 12 dB for downlink to have robust communication system. Miscellaneous loss include atmospheric loss, pointing loss and receiver line losses.

15.7 Command and Data Handling

The Command and Data Handling Subsystem (C&DH) consists of a processing unit and Solid State Recorder (SSR) for storing data that has been received and must be transmitted by Arina. The BAE RAD750 processor is required for on-board unit.

The power and mass budget for C&DH subsystem are shown in table 15.5.

Table 15.5: C&DH Components and Their Masses and Power Budget

Component	**Mass (kg)**	**Power (W)**
SSR	15	45
BAE RAD750	0.01	5
Total	**15.01**	**50**

Table 15.4: HGA Link Analysis

Parameter	**X-Band Downlink**	
Architecture	HGA-DSN	HGA-DSG
Uplink Frequency (GHz)	7.2	7.2
Downlink Frequency (GHz)	8.45	8.45
Maximum Distance (km)	400000	70000
Data rate (kbps)	500	100
Transmit Antenna Diameter (m)	0.5	0.5
Receive Antenna Diameter (m)	34	1
Transmit Power (dBm)	36.9	36.9
Transmit Antenna Gain (dB)	30.3	30.3
Receive Antenna Gain (dB)	66.9	34.9
ERIP (dBm)	65.2	65.2
Path loss (dB)	-223.0	-207.9
Misc. losses (dB)	-5	-5
System Noise Density dBm/Hz	-179.0	-183.2
Received E_b/N_0 (dB)	25.3	29.4
Required E_b/N_0 (dB)	9.6	9.6
Receiver System Loss (dB)	-1	-1
Link Margin (dB)	**15.7**	**19.8**

Life Cycle & Cost

Cost analysis of I-MARS project and life cycle analysis of Ariana have been discussed in this chapter. All expenses scaled to FY2017 [1] USD[2] by considering the effect of inflation during the time. As shown in figure 16.1, the CPI [3] is estimated for future years by assumption of a stable economic condition, collected from the Bureau of Labor Statistics [80].

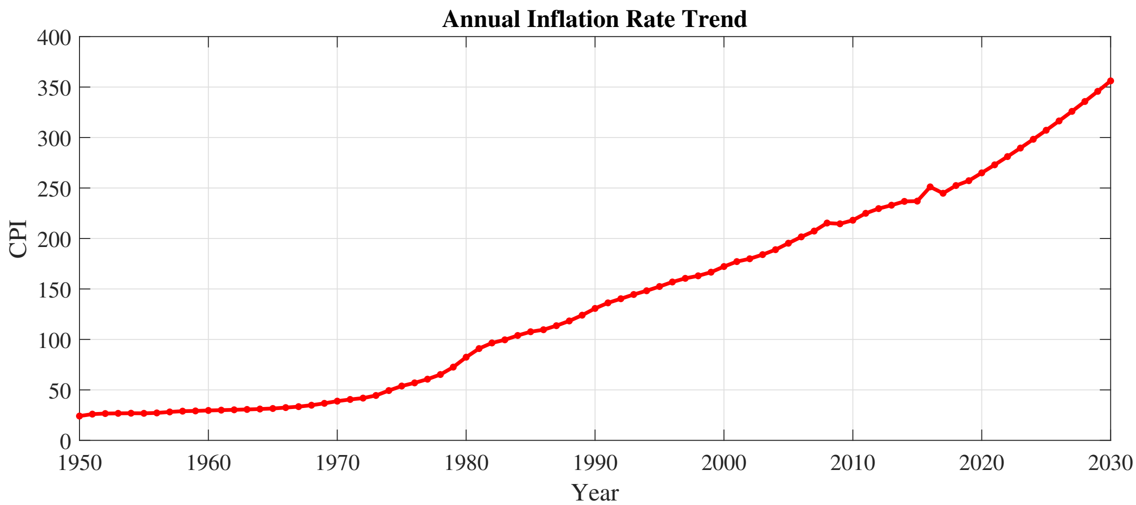


Figure 16.1: Annual US Inflation Rate from 1950 to 2030

Apollo program ended at 1972 with 6 manned landing and numerous unmanned landings on the Moon's surface with an overmuch total program cost of 112 B$. After ending of the program at 1988, cost of a new lunar lander predicted as 3.2 B$ for Design, Development, Test and Evaluation (DDTE), and 1.6 B$ for production [81]. The fact that any manned planetary landing mission after Apollo program has not been reached to production and lunch phase, makes estimation incapable of validation by actual data. However, comparing the estimated total program cost with two cost models approved the validation of present estimation. A list of RFP requirements on cost & life cycle has shown in table 16.1.

Table 16.1: Cost Requirements on RFP

No.	Requirement
1	The cost for the vehicle shall not exceed $10 Billion US Dollars (in FY17).
2	Lifetime of each component and Ariana's overall lifetime shall be estimated.

[1] Fiscal Year 2017
[2] United States Dollars
[3] Consumer Price Index

16.1 Cost Segments

Since cost estimation methods for spacecraft [26] were developed for unmanned, small spacecraft and satellites, a component-based approach has been utilized. In this method, each section's cost is calculated by considering the cost of the major component. Cost segments have been categorized as below [26]:

Table 16.2: Total Cost Breakdown of I-MARS Program

Element	% of total	Cost	Margin	Description
1.Space Vehicle	41%	3220 M$	11%	Including RDTE and production cost of the following elements
1.1.Propulsion and Cargo module	9%	700 M$	10%	
1.1.1.Structure and Mechanisms	5%	400 M$	10%	
1.1.2.Propulsion system	3%	200 M$	10%	5 × RL10 CECE Engines
1.1.3.TT&C	1%	50 M$	5%	
1.1.4.ADCS	0.2%	10 M$	5%	
1.1.5.Power	0.4%	30 M$	3%	
1.1.6.Thermal	0.1%	10 M$	3%	
1.2.Habitat	28%	2200 M$	15%	Modified BEAM habitat
1.3.Payload	3%	220 M$	10%	Scientific equipment such as rovers, Cameras and communication instruments
1.4.Integration/Assembely& Test	1%	100 M$	10%	Acceptance testing and integration of spacecraft
2.Launch & Orbital Operations	41%	3200 M$	20%	Planning and operations related to launch and orbital checkout of the space system
2.1.SLS	38%	3000 M$	20%	2 × SLS launch for Propulsion module and habitat
2.2.Operation	2%	200 M$	20%	
3.Ground Support Equipment	1%	65 M$	5%	
4.Program Level	3%	200 M$	10%	Costs which can not be assigned to an individual hardware or software.
4.1.Program Managment	2%	120 M$	10%	
4.2.System Engineering	1%	80 M$	10%	
5.Mission Operation & Support	4%	325 M$	15%	Estimated for 20 missions during about 10 years
5.1.Maintenance	2%	140 M$	15%	Ground and space Segment Hardware and Software Maintenace
5.2.Operation	1%	105 M$	15%	
5.3.facility	1%	80 M$	15%	
6.Reserve Cost	10%	772 M$	-	
Total Program Cost	**100%**	**7797 M$**		

16.2 Life Cycle Cost

As demonstrated in table 16.2, expenses are estimated based on the cost of the major components. Also NAFCOM[4] and AMCM[5] [82] cost models that are developed by NASA/Air Force performed a rough order estimation for the

[4]NASA/Air Force Cost Model
[5]Advanced Missions Cost Model

development and production phase and compared with our estimation.(figure 16.2)

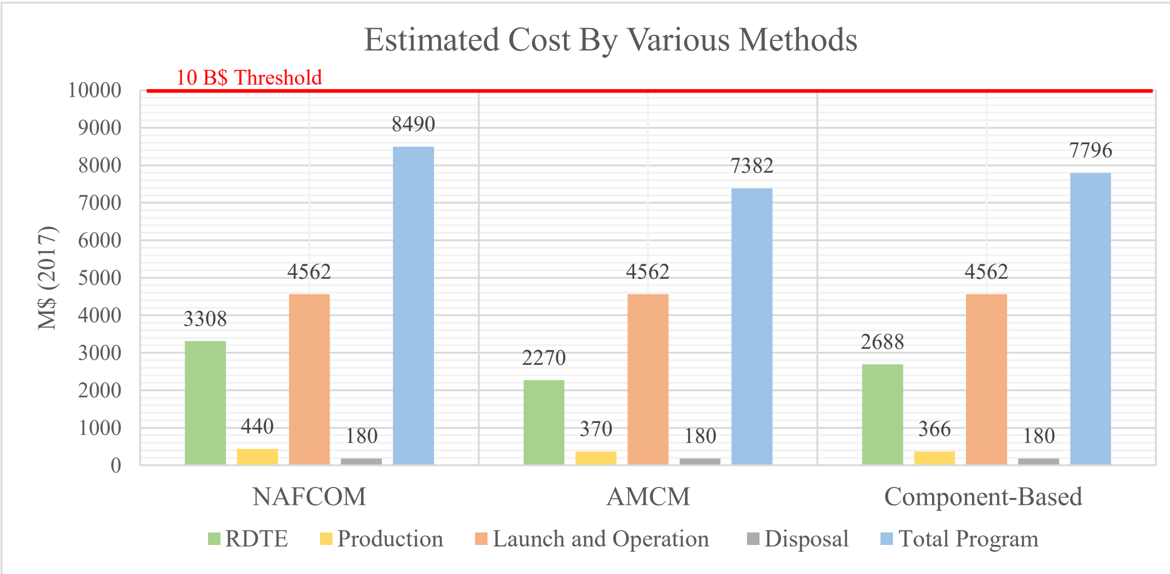


Figure 16.2: Comparison of Estimated Costs at Phases Calculated by Cost Models

16.3 Ariana Lifetime

Since it is necessary to know the lifetime of Ariana and the number of missions which the vehicle can operate; the initial lifetime estimation of each component described in the following section. After determination of the critical components and subsystems, the strategy for a potential replacement will be discussed.

16.3.1 Estimating Lifetime of Each Subsystem

In each subsystem, there are some critical components with limited lifetime, to estimate the lifetime of the vehicle and make decision about any replacement strategies the

Table 16.3: Lifetime Estimation of Each Component

Subsystem / Component	Estimated Lifetime (#of Missions)
1. Propulsion	
1.1. Main Engines	10
2. Control	
2.1. Thrusters	30
2.2. Sensors	90
3. ECLSS	
3.1. Habitat	10 Crew Missions
3.2. ECLSS	10 Crew Missions
4. Thermal	
4.1. MLI	Infinite
4.2. Cryocoolers	400
4.3. Electroheaters	250
5. Power	
5.1. Batteries	150
5.2. Solar panels	20 to 30
6. Structure	Infinite (Based on Inspections and Maintenance)
7. Communication	90
8. C&DH	90

The number of crew missions are assumed to be half of the total missions; therefore, the most critical component are engines in the propulsion system that would last for ten missions by the On-Off strategy in section 9.7

16.3.2 Mitigation to Extending Lifetime

After investigating the pros and cons of the strategies, it has been decided to send another propulsion module to DSG for replacement after ten cycles of Ariana's missions. This module will be sent to DSG in around 2034, and the old propulsion module can be used as external storage for exported propellants from the Moon's surface.

16.4 Schedule

- RDTE: This phase appertain developing employed technologies and accessing to a high level of technical maturity. BEAM habitat modification for the mission, RL10 CECE engine, lunar rovers and refuling procedure, are the main focused technologies for development and tests. A testbed with the Moon simulated environment would be constructed to determine the actual lifetime of engines and plan for further missions.

- Production: Fabrication of the vehicle's final version and subsystems that should be ready for launch.

- Operation: Operation of the vehicle classified as the following phases:

  – Launch: Two block 1B cargo configuration of SLS will be launched within less than six months from each other.

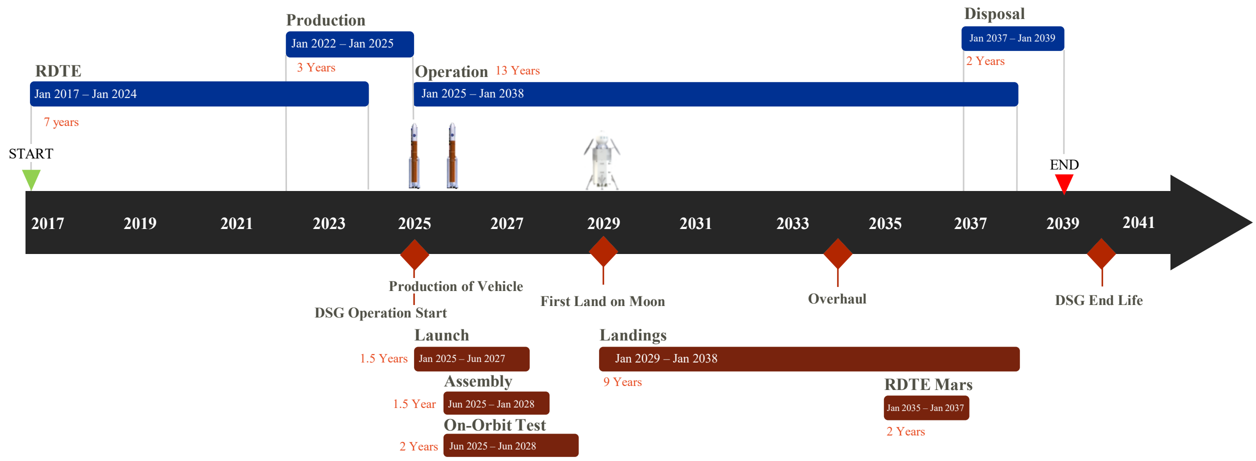


Figure 16.3: Timeline of the I-MARS Program

- Assembly: After the PM arrives on DSG, the habitat inside the PLM arrives by SM, and the process of assembling the modules would be started.
- On-orbit tests: To ensure the correctness of all systems and lifecycle of Ariana's components, on-orbit test would be performed.
- Landings: Before the start of crewed landings, it is essential to analyze the risk of the mission, perform some necessary tests in landing procedure and deliver some vital modules to the Moon's surface. So, first few missions (approximately three missions expected) will be cargo missions.
- Overhaul: Referring to section 9.7, a ten mission lifetime is estimated for overall engines and in purpose to extending lifetime of the vehicle, after the PM's end life, it will be replaced with a new one.
- Mars Mission Research: According to the Mars' missions, it would be carried out tests of subsystems or lander in the lunar environment until 2038.

• Disposal : At the end of Ariana's life cycle, the disposal phase will be started by passivating propulsion and power systems and the final plan of disposal will be putting Ariana on Heliocentric graveyard orbits.

Risk Assessment

It is obvious that successful missions occur on a regular and frequent basis. Risk assessment can be described as avoiding loss of crew, loss of vehicle, and loss of science mission. The lunar surface mission needs some strict consideration which is discussed below. Craters are the most common feature of the lunar terrain and considered as a hazard for safe landing as well as surface mission success [83]. Besides, harsh environmental conditions, like radiation, high temprature variation, affect the crew and electronic devices' performance. Also, program management must be implemented to mitigate the technical and schedule mission risks. Determination of anticipated risk and its impact on the mission helps to choose the best mission strategy in order to increase its success. The main objective of the mission is the crew's health and safety. Some risks such as human death might not be mitigated, but more money could be allocaeted on crew's training and medical tests.

17.1 Anticipated Risks and Mitigation Strategy

The risks and mitigation strategy are listed in table 17.1 to calculate the possibility of each risk by multiplying the likehood and impact of each risk element to examine the risk of the mission.

In fever chart, the risks are divided into low, medium, and high risk. As can be seen in figure 17.1, Ariana would operate at low risk. Despite mitigation strategies and a high level of safety, the possibility of an incident is not deniable. The most critical concern in Ariana's manned missions is keeping the crew safe even in catastrophic situations; this will drive a strategy for the ejection process; which is based on the safe-life concept. Habitat and SM will be separate from the rest of the vehicle, and its speed decreases to about 690 (m/s) for safe landing on the lunar surface. The SM tanks are sized for 5438.1 (kg) for capturing cargo module to NRHO. However, it is filled with 1884.5 (kg) of propellants for ejection process. Ejection process specification are shown in table 17.2

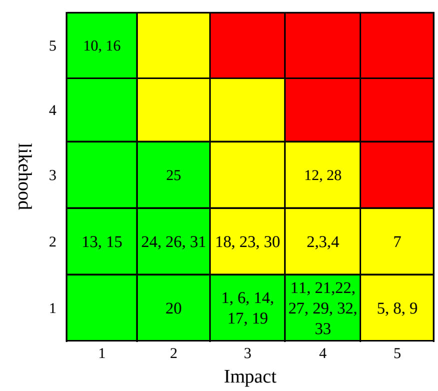


Figure 17.1: Fever Chart of Ariana's Missions

Table 17.2: Ejection Specification

Engine 8×R-40B				M_{dry} (kg)	Mass of Propellant (kg)	Mass of Helium (kg)	Available ΔV (m/s)
Isp (s)	Vac. Thrust (N)	Engine Mass (kg)	Mixture Ratio				
303	4000	13.6	1.65	7211.6	1884.5	10.5	690.00

Table 17.1: Ariana's Risk and Mitigation

No.	Risk	Likehood	Impact	Mitigation
1	ECLSS failue	1	3	Mature technology, Redundant elements, Utilizing passive O_2 reserve tanks
2	Crew's health care	2	4	Pre-mission training
3	Habitat system failure	2	4	Redundant components
4	Habitat equipment ignition	2	4	Fire detection and sensors, EVA suit, Creat vacum
5	Thruster failure	1	5	Gimball system aides, Can be compensated by combination of 34 other thrusters, Redundancy, Share the power to cover the mission objective
6	Electronic Instrument Error	1	3	Use of proven component, Redundant wires and switches
7	Micro Meteoroid and Orbital Debris (MMOD)	2	5	Perform maneuver to escape from MMOD
8	Inability to dock with DSG	1	5	Reliable thrusters(TRL9), Use EVA suite to go to DSG
9	Loss of attitude control	1	5	Redundant thruster
10	Impact of moving and rotating component	5	1	Closed loop control system for fixing attitude
11	Collision with DSG and Moon	1	4	Main engine Gimbal aid
12	Missing planned launch date	3	4	Schedule and cost margins consideration, Utilizing alternate maneuver
13	Error accumulation during orbital transfer	2	1	Consider margin for ADCS system, Reliable sensors
14	Engine Failure	1	3	Redundant engine and fault procedure scenario\9.7
15	high consentration O_2	2	1	Redundant pressure sensor
16	fuel boil off	5	1	Acepted risk, Thermal control
17	Engine restartablity	1	3	\9.5
18	Solar panel deployment failure	2	3	Use of proven technology, High margin EPS sizing
19	Battery shortage	1	3	Batteries size to provide sufficient power with high margin
20	Loss of communication	1	2	Use of proven communication device, High margin data transmission
21	HGA failure	1	4	Proven high TRL design, High link margin
22	Processor unit failure	1	4	Redundant instruments
23	Vehicle subsystem and component overheating	2	3	Heat dissipation by radiators, Use of MLI
24	Environment impact on vehicle's component	2	2	Redundant design for worst case
25	Active systems failure	3	2	Redundant thermal subsystems, Accurate monitoring, Passive system designed to be adequate
26	Environment Impact on vehicle's component and crew (radiation and temprature)	2	2	Strong sheilding against radiation, Redundant thermal design for worst case
27	Fatigue, crack formation	1	4	Monitoring, Inspection, Maintenance
28	Corrosion, rust	3	4	Shield layer entire the structure
29	Environment stresses and vibration hazard	1	4	Extensive and accurate, Test, Accurate material selection
30	lander leg fail to deploy/separate	2	3	Test in DSG
31	Damage during unloading operation	2	2	Redandant and alternative methods
32	Cost overrun	1	4	Allocation of 11% cost margin, 7 year RDTE budget shall be sufficient, High TRL subsystem was selected
33	Schedule overrun	1	4	Scheduling, Management.

Appendix

Subsystem	Capture to DSG (kW)	DSG to Moon (kW)	Moon's Surface (kW)	Standby (kW)
Payload (Cargo)	**0.00**	**0.00**	**4.00**	**0.00**
ECLSS (Crew)	**3.70**	**3.70**	**3.70**	**3.70**
Communication	**0.08**	**0.08**	**0.08**	**0.80**
Propulsion	**0.14**	**0.14**	**0.10**	**0.10**
Power	**0.10**	**0.10**	**0.10**	**0.10**
Thermal	**0.80**	**0.80**	**0.80**	**0.80**
Attitude & Control	**2.10**	**2.30**	**0.10**	**0.10**
Total Cargo Mode	-	**3.42**	**5.18**	**1.18**
Total Crew Mode	**6.92**	**7.12**	**4.88**	**4.88**

Table 18.1: Power Budget

		Cargo Mode	Crew Mode
Structure	**PM**	**4265.0**	**4265.0**
	PLM	**1043.2**	**1043.2**
	Landing Legs and Actuator	**1260.0**	**1260.0**
Habitat		**0.0**	**2000.0**
Service Module	**ECLSS support**	**0.0**	**5200.0**
	Cold Gas System	**0.0**	**158.8**
	Thrusters	**0.0**	**108.8**
	Propellant	**0.0**	**1884.0**
Thermal Protection		**1320.0**	**1780.0**
Main Propulsion System	**Engines**	**795.0**	**795.0**
	Propellant Tank	**626.0**	**626.0**
A/ODCS	**Thruster**	**489.6**	**489.6**
	Tanks	**25.3**	**25.3**
	Propellant	**1038.0**	**1038.0**
Power System	**Batteries**	**1200.0**	**3200.0**
	Solar Panels	**100.0**	**100.0**
	Super Capacitor	**150.0**	**150.0**
C&DH		**15.0**	**15.0**
Communication		**71.1**	**71.1**
Dry Mass		**12397.1**	**24209.8**

Table 18.2: Dry Mass Budget

Model	ALiSSE criteria	
	Mass (kg)	Power (kW)
Air Management System	**828.154**	**0.973**
Atmosphere control system	119.4	0.07
Atmosphere pressure control system	119.4	0.07
Atmosphere revitalization system	303.442	0.902
Carbon dioxide removal system (incl. consumables)	137.502	0.395
Carbon dioxide reduction system (incl. consumables)	0	0
Oxygen regeneration system (incl. consumables)	0	0
Trace contaminant control subsystem (incl. consumables)	26.81	0.246
Trace contaminants monitoring system	29.7	0.11
Major constituents monitoring system	35.11	0
Airbone microorganisms monitoring system	55	0.11
Airbone particles monitoring system	1.87	0.006
Gaseous wastes line	2.71	0
Gaseous wastes monitoring system	1.95	0.005
Gaseous wastes storage system	0	0
ISS-like vent and relief valve (VRV)	5.44	0.03
Particulate and microbial contamination control system	7.35	0
Fire detection and suppression system	9.29	0.001
Fire detection system	1.54	0.001
Fire suppression system	7.75	0
Gas storage	396.022	0
Inert gas storage system	11.405	0
Inert gas supply system	4.661	0
Nitrogen (gas)	20.512	0
Oxygen storage system	98.126	0
Oxygen supply system	4.579	0
Oxygen (gas)	256.739	0
EVA Support	**283.882**	**2.64**
EVA support system	283.882	2.64
EVA CO2 removal system (incl. consumables)	6.864	0
EVA maximum absorbency garments	1.038	0
EVA radiation monitoring system	0.2	0
EVA umblical assembly	43	0.14
Oxygen recharge compressor assembly for EVA	162.48	1.5
Airlock recycle pump	70.3	1
Food Management	**238.971**	**0**
Food processing and storage system	238.971	0
Food and food ingredients regeneration system	15	0
Food storage system	38.871	0
Food (partially hydrated)	185.1	0
Food production system	0	0
Food complement unit (incl. consumables)	0	0
Produced food quality monitoring system	0	0
Crew accomodation system	**256.455**	**0**
Crew emergency provisions	6.255	0
Portable breathing apparatus (PBA)	6	0
Radiation monitoring system	0.255	0
Crew support system	250.2	0
Clothing	145.8	0
Laundry equipment	0	0
Shower	97.5	0
Wet wipes	6.9	0
Thermal Management System	**186.506**	**0.005**
Temperature and humidity control system	156.97	0
Cabin air THC assembly	119.62	0
Avionic air THC assembly	12.4	0
Cabin ventilation assembly	24.95	0
Intermodule ventilation assembly	0	0
Internal thermal control system	29.536	0.005
Internal thermal control system	29.536	0.005
Waste Management	**144.3**	**0**
Solid waste processing system	144.3	0
ISS commode/urinal	45	0
Solid and concetrated liquid wastes processing system	27	0
Solid waste storage system	72.3	0
Water Management	**3254.34**	**0.092**
Water storage	3157.475	0.016
Potable water storage system	751.275	0
Potable water	2385	0
Waste water storage system	0	0
Water supply system	19.74	0.016
Overboard water vent assembly	1.46	0
Water recovery system	96.865	0.076
Water recovery system (incl. consumables)	0	0
Waste water collection/trasportation system	19.74	0.016
Waste water monitoring system	38	0.05
Waste water stabilization system	39.125	0.01
Totals	**5192.608**	**3.71**

Table 18.3: Outout of SCALISS Tool

Compliance Matrix

No.	RFP Requirement	Met	Explanation	Chapter
1	The design should be reusable	Yes	Vehicle's subsystems and its modular design provide multiple operations	All
2	The vehicle should enable the delivery of payload and/or crew to anywhere on the surface of the Moon from the DSG and back	Yes	Optimal transfer trajectories to different latitude/longitude landing sites and back to DSG for both crew and cargo modes are simulated	8
3	Design of the vehicle should allow for the capability of switching between the two modes	Yes	The modular configuration is developed	7
4	the vehicle should support a crew of four astronauts for the duration of the transit from the DSG to the lunar surface and should support the crew without resupply from surface assets for a minimum of 24 hours on the surface	Yes	Non-regenerative ECLSS is used to support the crew of 4 for 6.5 days on the Moon's surface	11
5	The vehicle should have a payload capacity of at least 15,000 kg to the surface of the Moon and can return at least 10,000 payload from the surface back to the DSG	Yes	Multiple restarts, High lifetime, and Deeply throttled engines are used to deliver the maximum cargo of 22.5 mt to the moon and 17.5 mt back to DSG	9
6	Describe in detail how the vehicle will be deployed to the DSG,	Yes	Two SLS block 1B launch vehicle's modules to TLI and the assembly would complete the deploy phase in DSG.	5.2
7	The cost for the vehicle shall not exceed $10 Billion US Dollars (in FY17),	Yes	The entire program costs 7.8 B$	16
8	The vehicle shall make its first trip from the DSG to the lunar surface no later than December 31, 2028	Yes	The first Launch is scheduled to be in 2025	5.1

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