Multi-Use Lunar Transportation Vehicle Utilizing Deep Space Gateway

AIAA 2019 Undergraduate Spacecraft Design Competition Submission

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AIAA	American Institute of Aeronautics and Astronautics	MIMU	Miniature Inertial Measurement Unit
ARS	Air Revitalization System	MLI	Multi Layer Insulation
ATCS	Active Thermal Control System	MMH	Monomethyl Hydrazine
ATP	Astrophysics Theory Program	N	Newton
BER	Bit Error Rate	NASA	National Aeronautics and Space Administration
CAD	Computer Aided Design	NCC	Nominal Corrective Combination
CDHs	Command and Data Handling System	NRHO	Near Rectilinear Halo Orbit
CM	Crew Member	NRI	National Robotics Initiative
DC	Direct Current	PCDU	Power Control and Distribution Unit
DoD	Depth of Discharge	PCS	Pressure control system
DSG	Deep Space Gateway	PCU	Power Contol Unit
DSN	Deep Space Network	PDR	Preliminary Design Review
ECLSS	The Environmental Control and Life Support System	PEG	Powered Explicit Guidance
EPS	Electric Power System	PWMS	Potable Water
EVA	extravehicular activities	QPSK	Quadrature Phase Shifting Keying
FDS	Fire Detection and Suppresion	RCT	Ramdomized Controlled Trial
FSI	Flight Suit Interface	RFP	Request for Proposal
GMAT	General Mission Analysis Tool	SAR	Single Axis Rotation
H	Hour	SFCG	Space Frequency Cooperation Group
HGA	High Gain Antenna	SHReD	Supplemental Heat Device
IMU	Inertial Measurement Unit	Si	Silicon
IR	Infrared Waves	SLS	Space Launch System
Isp	Spesicif Impulse	STK	Systems Tool Kit
ISS	International Space Station	t_0	Initial Time
ISRU	In Situ Resource Utilization	t_f	Final Time
K	Kelvin	TA	True Anomaly
kW	KiloWatt	TCS	Thermal Control Subsystem
Kg	Kilogram	TLC	Trans Lunar Coast
Km	Kilometer	TLI	Trans Lunar Injection
Km/s	Kilometer per Second	TOF	Time-of-Flight
M	Meter	Ts	Radiative Sink Temperature
M/s	Meter per Second	USD	United States Dollar
m_f	Final Mass	V	Velocity
m_p	Propellant Mass	W	Watt
LCG	Liquid Cooling Garment	WM	Waste Management
LEO	Low Earth Orbit	ΔV	Change in Velocity
LGA	Low Gain Antenna	α	Radiator Solar Absorptivity
LLO	Low Lunar Orbit	q	solar Incident Solar Flux

1. Introduction and Overview

1.1 Mission Definition

In this subdivision, the Request for Proposal (RFP) [29] for the Reusable Lunar Surface Access Vehicle given by the American Institute of Aeronautics and Astronautics (AIAA) is reviewed. The goal of the competition, as stated in the RFP, is to design a space vehicle that will be operated from the Deep Space Gateway (DSG), which will be used as a staging point for the necessities of the spacecraft (i.e., refueling, assembly, reparation, etc.), will be able to transport a cargo of fifteen metric tons (15,000 kg) to the lunar surface and ten metric tons (10,000 kg) back to the DSG, or transfer a crew of four (4) to the lunar surface and return them. The ability to easily and swiftly reach the lunar surface is a stepping stone towards ultimate goal of deep space exploration. Therefore, this mission has an exceptional value for the future space operations.

1.2 Requirements

P1	The spacecraft should make multiple trips utilizing the Deep Space Gateway.
P2	Mission shall carry a payload of 15,000 kg to Lunar surface and 10,000 kg back to DSG in cargo mode
P3	Mission shall safely transport four (4) humans to the Lunar surface and back to the DSG in crew mode
P4	Vehicle design should allow for capability of switching between two modes
P5	The spacecraft should support the crew for the duration of the trip and an additional 24 hours on the surface
P6	The spacecraft should access on any specific point on the Lunar surface
P7	Total cost of the mission shall not exceed $10 billion
P8	The vehicle should complete its first trip to the lunar surface until December 31st, 2028

Table 1.1: **RFP Requirements**

The mission requirements as they are stated in the RFP [29] are summarized in Table 1.1. These requirements include the payload capacity for cargo missions, and the ability of the vehicle to switch between different mission modes. During crewed missions, The safety and survivability of the crew is of top priority, and the spacecraft should be able to supply the astronauts sufficiently for the duration of the flight and at least 24 hours on the surface of the Moon. Another significant requirement is the ability to land on any desired point on the lunar surface, which greatly impacts communication architecture and ΔV cost. In order to satisfy all of the requirements, all of the critical decisions were made in order to create a safe reliable, operable, and reusable vehicle that will bring the Moon within our grasp.

1.2.1 Deep Space Gateway and NRHO

The Deep Space Gateway is an exploration and science outpost in orbit around the Moon [30]. It supports four crew members, 30 to 90 day crewed missions, a huge amount of payload, has a Robotic Arm to perform various operations [31], and is accessible via SLS. As stated by the RFP, the DSG was assumed to be in a stable Near Rectilinear Halo Orbit (NRHO). The NROs are halo orbits with large amplitudes over either the north or south pole with shorter periods that pass closely to the opposite pole[4]. Therefore, an orbit from the NRHO family was chosen, and calculations were based on it. The simulation of the chosen NRHO was done using both STK and GMAT softwares with the input parameters given in Table 1.2. GMAT is a powerful open-source space mission analysis tool developed by NASA, and STK is a more extensive software developed by AGI. Both of them are able to simulate the solar system with high accuracy, and were used throughout the study for trajectory calculations and validations. Both orbit families (north and south) always favor one pole, and neither north nor south NRHOs have an advantage over each other in terms of communication coverage. Hence, coverage was not a deciding factor in orbit selection. The recent findings of ice on the Moon's solar pole [33], and NASA's plans of sending astronauts to this region [34] were the main driving factors during this process. The L2-South orbit of interest stays stable for around 25 days and requires very small correction maneuvers afterwards.

Table 1.2: Moon-centered Earth-Moon rotating frame values [30]

	Periapsis	1/8 Rev	1/4 Rev	3/8 Rev	1/2 Rev
rx (km)	-247.122	6060.483	11467.119	14868.184	16023.074
ry (km)	0.000000	19452.284	16269.487	8955.187	0.000
rz (km)	4493.209	-34982.968	-56381.822	-68055.505	-71816.650
vx (km/s)	0.000000	0.082677	0.059130	0.030451	0.000000
vy (km/s)	1.444467	0.006820	-0.077120	-0.111682	-0.121971
vz (km/s)	0.000000	-0.368434	-0.212112	-0.100368	0.000000

1.3 Vehicle Configuration

A variety of vehicle configurations were considered. Since the vehicle will be used to transport humans, the overall configuration along with every subsystem were carefully chosen to maximize safety while maintaining utility. Each configuration was contemplated in accordance with the requirements P1 through P4.

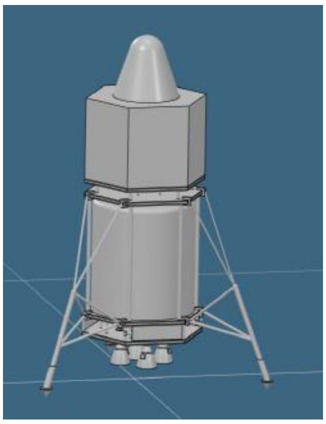
First configuration includes separate crew and cargo modules, and a service module that is comprised of the necessary systems. The three components would be assembled in orbit, and then sent to rendezvous with the DSG by using its own propulsion system. After the initial docking, the three components would never be together again, and the vehicle would operate mission-specific. That means the service module would dock with the crew capsule for crewed missions and vice versa, then deploy the module it would be carrying to the lunar surface and bring it back. This method would eliminate the unnecessary weight and components for different types of missions, rendering the vehicle mission-specific. Also, having separate modules for crew and cargo would allow one module to keep operating even if the other one was out of service (e.g., requires maintenance). However, it would be difficult to utilize (i.e., docking/undocking a couple of times every time it is used) and would increase the cost and complexity of the system. Unloading the cargo would be an issue as well. Another drawback of the system would be the constant occupation of the DSG, as one module would always remain docked.

Second configuration is a two-part vehicle consisting of a crew module and a service module with removable cargo compartments attached on the main body instead of a third separable module. The crew capsule would dock with the descent module for crewed missions and if desired, the cargo compartments would be removable with the aid of the Robotic Arm on the DSG. For cargo missions, the crew capsule would stay docked to the DSG, and the service module would carry the cargo. This configuration would decrease the complexity of the first one, make for a lighter and easy to implement system, and would be efficient in terms of volume and fuel usage, while allowing for unloading the cargo remotely and practically. The most significant drawback would be the size of the vehicle as it would impact the launch from Earth strategy greatly.

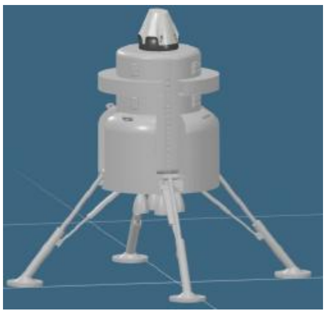
Evaluating the considerations above with a trade study, the most optimal configuration for the mission was decided to be the second one. A simplified summary of the trade can be seen in Figure 1.1. A full four-view of the final configuration can be seen in Appendix B.

	Weight	Complexity	Size	DSG Occupation	Total
Config 1	2	1	3	1	1.5
Config 2	3	2	2	2	1.9

Figure 1.1: Trade Study for Vehicle Configuration



(a) **First Configuration Proposal**
From top to bottom; crew module, cargo module, service module.



(b) **Second Configuration Proposal**
From top to bottom; crew module, service module with cargo compartments on the sides.

Figure 1.2: Configuration Considerations

1.4 Executive Summary

The purpose of the mission is to return humans to the Moon, long years after the Apollo missions. Only this time instead of simply visiting and returning, we will develop a way to travel back and forth between the DSG and the Moon in order to be able to develop and test new, advanced technologies and systems in an environment that is both in close proximity to the Earth and operational. Another benefit that will emerge from the mission is the ability to perform scientific investigations on the lunar surface, which will give us an improved knowledge about the regolith composition, ice at lunar poles, solar system volatile history, comet impact and solar activity progression [34]. Oxygen, water and other materials can also be extracted from the lunar soil. This team chose systems in such a way to prioritize the safety of the crew, while allowing the vehicle to make multiple trips and making use of the latest developments. For this very reason, a combination of emerging and proven technologies were used. Total cost of the vehicle including Design, Development, Test, and Evaluation (DDT&E) and Theoretical First Unit (TFU) costs is estimated to make up $9 billion out of a budget of $10 billion. Therefore, the mission makes use of the available budget as efficiently as possible while providing a suitable margin.

The name of the vehicle is *Adjustable Expeditive Lunar Landing Conveyor*, or JELLY for short. It was chosen because it is a simplistic name that briefly states the purpose of the vehicle, and also because the vehicle slightly resembles a jellyfish with its landing gear undeployed. Furthermore, jellyfish are immortal by nature, which reflects the reusability of the vehicle.

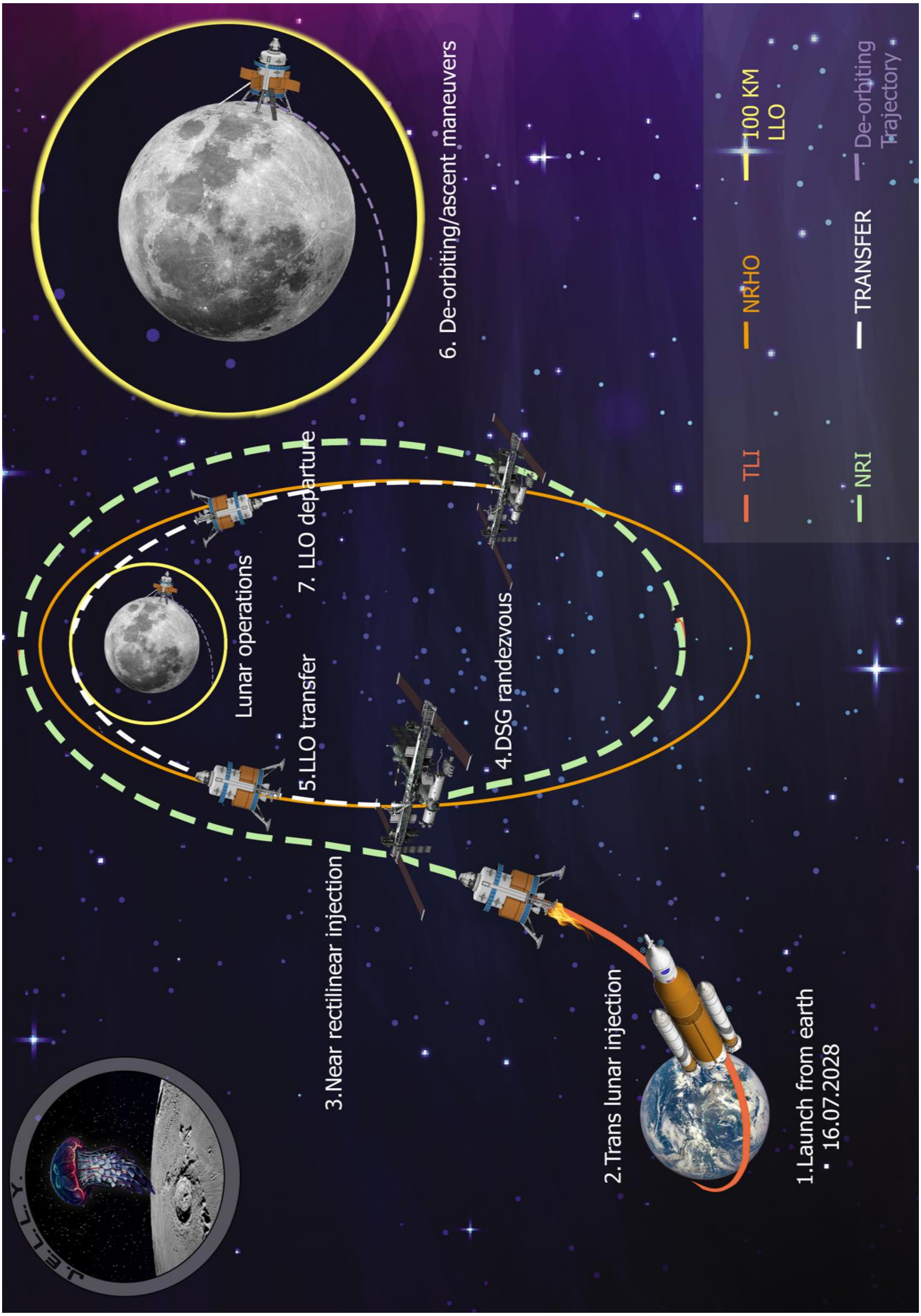


Figure 1.3: **ConOps for the whole mission.** This figure summarizes JELLY's Concept of Operations

On July 16, 2028, the anniversary of Apollo 11, NASA's SLS will launch from Earth carrying the full JELLY vehicle, assembled pre-launch, with a launch mass of 50 tons to TLI. After a TOF of 4 days and 7 hours, the vehicle will use its own LOX/LH bipropellant propulsion system to perform a NRI to insert itself into the orbit of the DSG, and proceed to rendezvous with it. The total ΔV and TOF for this segment is given in Table [?]. During the process, JELLY will communicate with Earth via DSN and also with DSG to perform the orbital insertion and rendezvous maneuvers. After successfully completing this part, phase two will begin, which consists of the operations between the DSG and Moon.

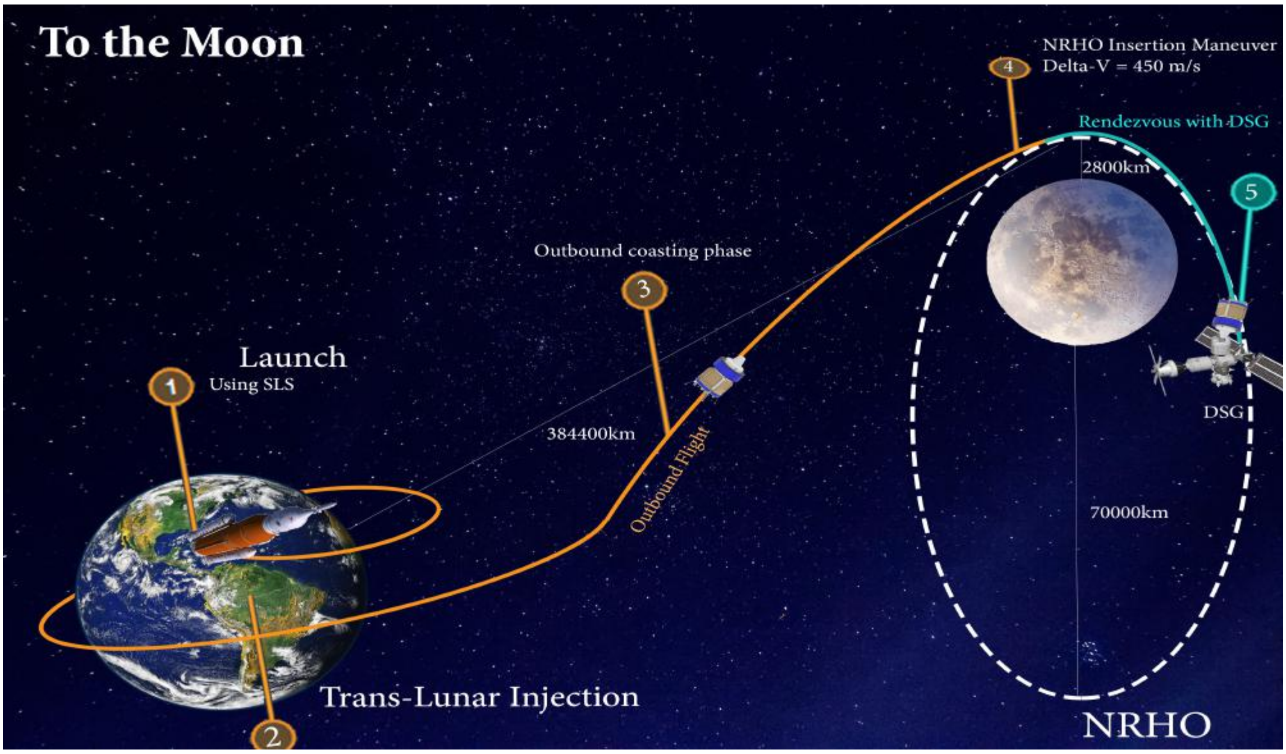


Table 1.3: **ConOps for the Launch Segment.** This figure depicts the first phase of the mission, from launch until rendezvous with the DSG.

From P4, the vehicle is able to switch between cargo and crew modes. For orbital maneuvers, there are two factors at play, ΔV and time of flight. The primary driver for cargo missions is ΔV, but both factors are important for crewed missions. thus, both modes have different mission design with the most optimal ΔV/T.O.F. combinations. All of the calculations and design decisions were made according to the maximum ΔV and maximum mission duration.

A detailed explanation about both mission modes are given in the following chapters. This section focuses on the common structure of each mode.

Phase	ΔV (m/s)	T.O.F. (days h:m)
Launch		
NRI	390	4d 16:07
Initial Docking	400	1d 03:00
Total	790	5d 19:07
Lunar Operations (Max ΔV case)		
LLO transfer	950	2d 02:20
Landing	1950	0d 00:40
Ascent	2000	0d 02:00
NRHO transfer	800	2d 12:00
Total	5700	4d 17:00

Table 1.4: **ΔV and TOF Budgets** This table summarizes the maximum ΔV and corresponding TOF values

Once JELLY completes docking, it can begin operating between the DSG and the Lunar surface after all the maintenance checks are done and the fuel tank is filled up. On July 22nd, JELLY with a crew of four will deploy from the DSG at 240 ° True Anomaly of its orbit, and begin the Lunar access sequence. The transfer will take around only 5 hours. In order to make a safe and precise landing, the vehicle will first inject itself to a Low Lunar Orbit (LLO). It will then begin its descent and complete the landing on the same day. After spending 24 hours on surface, the ascent phase will begin. JELLY will start its engines on July 23rd and ascend to a 15x100km elliptical orbit followed by a circularization maneuver. This whole process will take around 2 hours. Then an orbital maneuver, calculated by Lambert's problem, will be performed to raise the orbit in order to rendezvous with and dock to the DSG.

From P6, JELLY has to has access to any specific point on the Lunar surface. Characterizing particular landing sites, along with the abort capability from them, is important for crewed missions and global surface access is required for cargo/lunar sample return operations. In order to do so, a particular LLO must be achieved when leaving the DSG. There are two options in this case, either by performing a plane change maneuver or by waiting for the right opportunity while in a LLO. The plane change maneuver is directly related to the orbital velocity and the required inclination change from equation 1.1, and has a significant ΔV cost.

$$V_{pc} = 2V sin[\frac{\Delta i}{2}] \tag{1.1}$$

A detailed analysis of ΔV and TOF requirements for inclination change is given in section 2.2. By departing from the DSG close to the apogee of its orbit, this cost can be reduced to a minimum which is well within the capability of JELLY. Thus for missions that require plane changes, a higher time of flight strategy will be employed. Also in order to avoid landing during lunar nights when there is no light, no solar energy or heat, most of the missions will be planned beforehand to land when the desired site is illuminated by the sun. JELLY, however, is still capable of operating 24 hours on the surface without sunlight for emergency cases.

Figure [] shows the communication coverage of the lunar surface from NRHO.

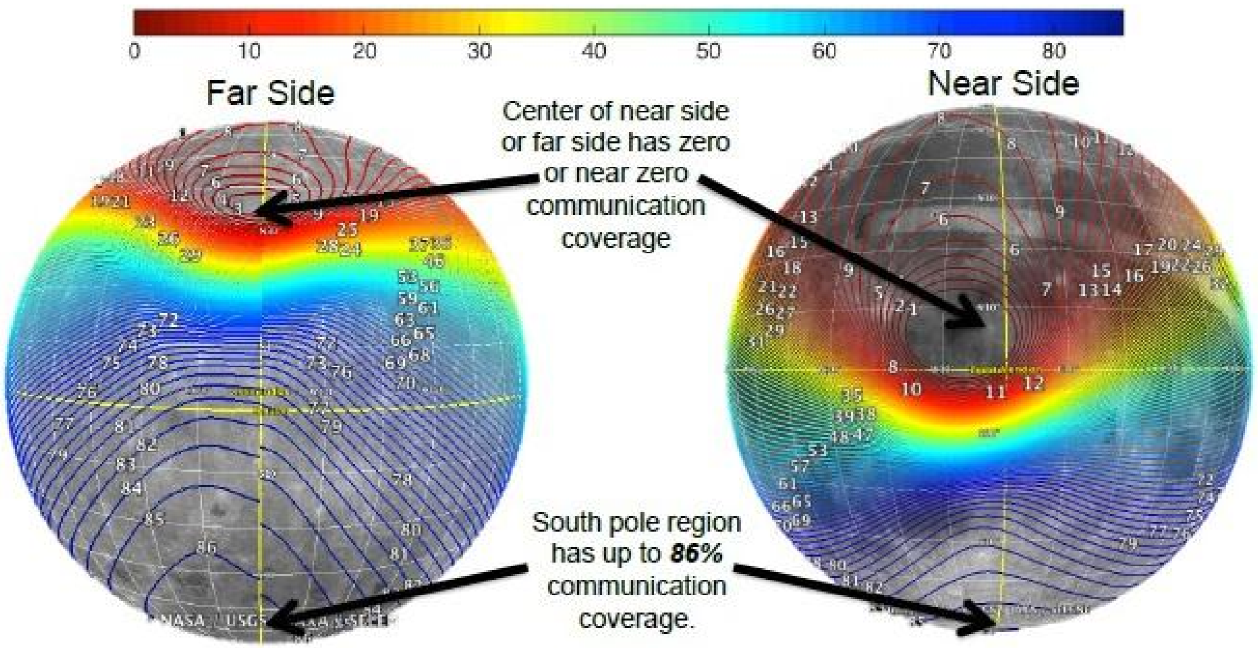


Figure 1.4: **NRHO Communication Coverage**[4] Percent of coverage of lunar surface from NRHO

When JELLY is on the near side and there is no access to the DSG, it will communicate directly with the DSN in Earth using its LGA in S-band, so the only problem is establishing communication on the far side. When JELLY is in a LLO, it will have blackouts form the Earth that last for one hour during which it can communicate with the DSG for most of the time. After landing on the surface of the far side, JELLY will lose line of sight with the DSG in about 10 hours. Although not having anything in LOS, since JELLY has its position information, it does not require any communication to perform its lift-off operations.

2. Trajectory Design

In this section, the launch, on-orbit rendezvous, landing, lift-off, and docking sequences and trajectories are explained.

2.1 Launch From Earth

Since JELLY has a very large diameter (10m) and a launch mass (around 48 tons), the only option to launch the assembled spacecraft is by using Space Launch System (SLS) Block 2B [Pietrobon, Steven. (2017). Fly Me To The Moon On An SLS Block II]. It is highly desirable, in terms of safety and simplicity, to perform the launch in a single stage without any LEO assembly as the SLS has the sufficient power to perform a trans-lunar injection (TLI) maneuver [Harbaugh, Jennifer (July 9, 2018). "The Great Escape: SLS Provides Power for Missions to the Moon". NASA. Retrieved March 4, 2019.] for payloads larger than 45 tons. Thus, by only utilizing the SLS capabilities, JELLY will be set on a trajectory toward the Moon. Although there still is not an exact date set for SLS Block 2, some sources state that it will be available as early as 2028. [67]

After the initial launch to 185 km circular parking orbit and the following TLI insertion, SLS will have completed its mission, and JELLY will continue its journey by itself. In about 4.5 days, by making Trajectory Correction Maneuvers (TCM) when necessary, the spacecraft will be in the vicinity of the Moon. Utilizing its own propulsion system, JELLY will perform a Near Rectilinear Orbit Insertion (NRI) maneuver in order to capture itself to a NRHO just below the orbit of DSG. In order to achieve the most optimal ΔV for this maneuver, launch simulations for different months were done using GMAT. Since the availability of SLS Block 2B is not exactly known, and to add a schedule margin to complete the mission before the deadline, the first and last two months of the year 2028 were eliminated. Table [] lists the average ΔV for the chosen

Date	Average ΔV (m/s)
March 2028	960
April 2028	400
May 2028	393
June 2028	395
July 2028	390
August 2028	405
September 2028	397
October 2028	640

Table 2.1: ΔV Costs For Various Launch Times

interval. Thus, July 16, 2028 was chosen for the launch date because it is the most optimal one, and also because it is the anniversary of Apollo 11.

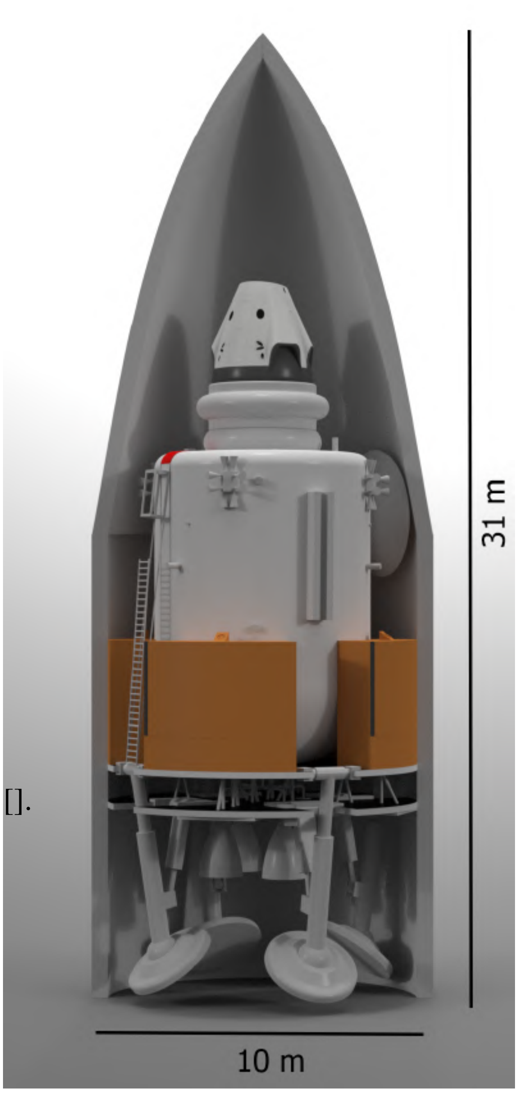


Figure 2.1: SLS Launch Configuration

After the NRI maneuver, JELLY will immediately start the docking operation utilizing its remaining propellant and RCTs. Since the DSG is not operational yet, its position is unknown at this point, but by altering the time of launch it is possible to keep the required docking time minimum. At this point, the process is assumed to be 27 hours, which is the time it takes for the Dragon capsule to dock with the ISS [**?**]. JELLY will also have sufficient propellant to provide a ΔV of 400 m/s at this step for performing a burn to catch-up with the DSG.

Visualization of Launch can be seen from Figure[] and the summary of the launch phase is given in Table [].

Date	ΔV (km/s)	T.O.F.	Operation	Vehicle
16 Jul 2028 15:00:00	9.21	0d 00:12:00	Launch	SLS
16 Jul 2028 17:00:00	3.127	4d 14:07:15	TLI	SLS
-	0.1	-	TCM	JELLY
21 Jul 2028 07:07:15	0.390	-	NRI	JELLY
21 Jul 2028 07:10:00	0.4	1d 03:00:03	Docking	JELLY

Table 2.2: Launch Operations Summary

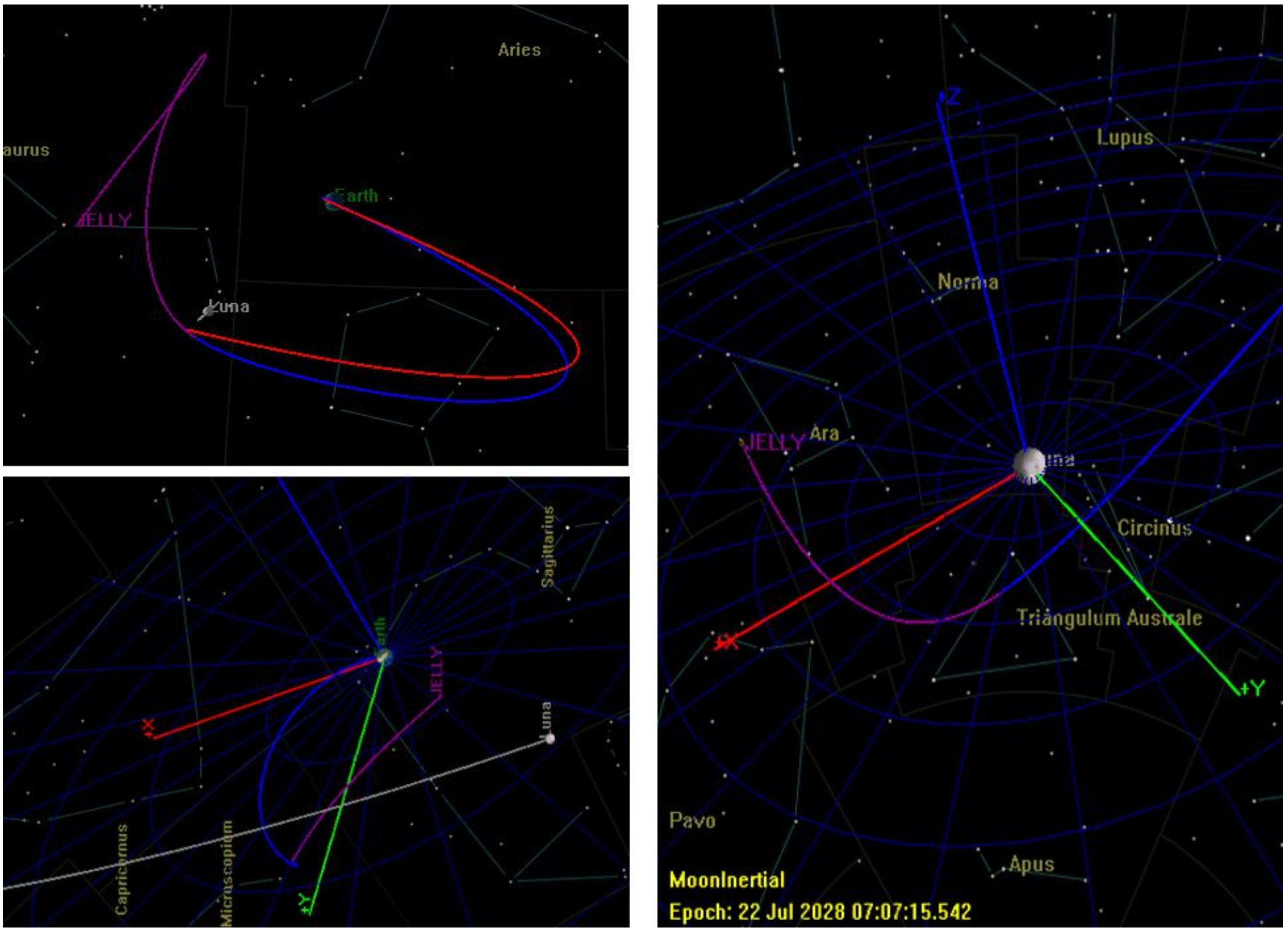


Figure 2.2: Launch Visualization

2.2 From DSG to LLO

One of the most challenging parts of the mission design is finding the optimal trajectories to reach lunar surface. From P6, in order to reach anywhere on the Moon JELLY will use a 100 km circular LLO. After arriving at the specified orbit, JELLY will loiter until the right opportunity arises and consequently make a de-orbiting maneuver. In order to find the most optimal trajectory, all possible trajectories were simulated using the tools GMAT and STK.

2.2.1 Polar Sites

Since NRHO can be considered a polar orbit, JELLY is able to reach polar low lunar orbits without performing any plane changes. First, all possible transfer trajectories were examined. Then, to find the optimum departure true anomaly, orbit of the DSG was divided into true anomalies with 5 degree intervals. For instance, at Figure 2.3, transfer trajectories with departure true anomalies from 180, 200 and 220 can be seen, respectively.

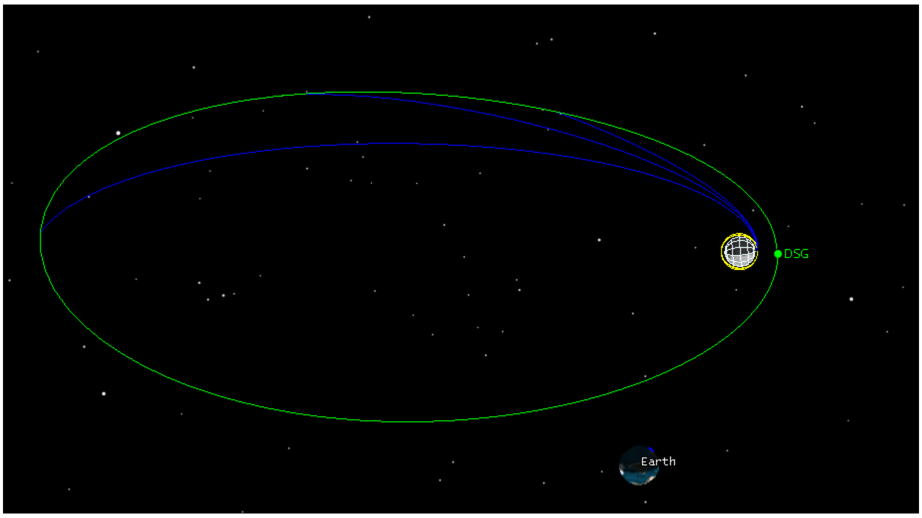
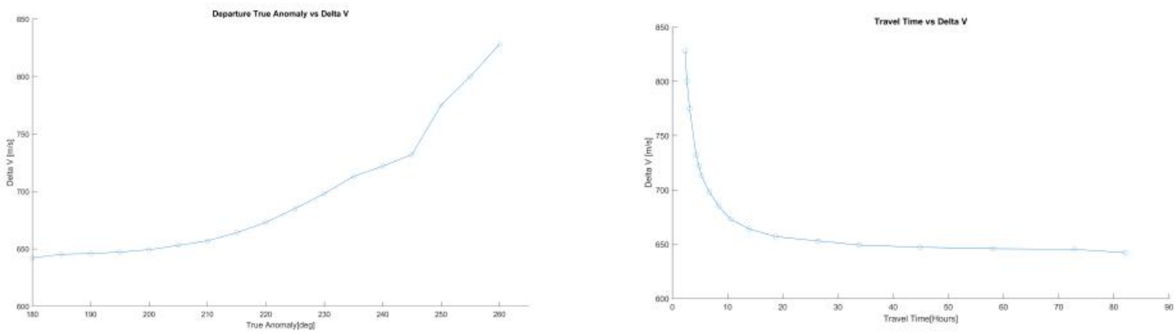


Figure 2.3: **DSG Departure Scenarios.** STK Simulation of the Transfer Trajectory

Results show that minimum ΔV for a transfer trajectory from NRHO to LLO can be obtained at 180 with a Hohmann transfer. Increasing the departure true anomaly results in a decrease in travel time to LLO, but also in an increase in ΔV cost.



(a) ΔV - True Anomaly (b) Departure ΔV - Time of Flight

Figure 2.4: Departure True Anomaly ΔV and Travel Time Relations

For crewed missions minimum transfer duration with reasonable ΔVs were considered. Departure true anomaly around 230-250 ° resulted in a less than half day transfer with a ΔV cost of 720-750 m/s.

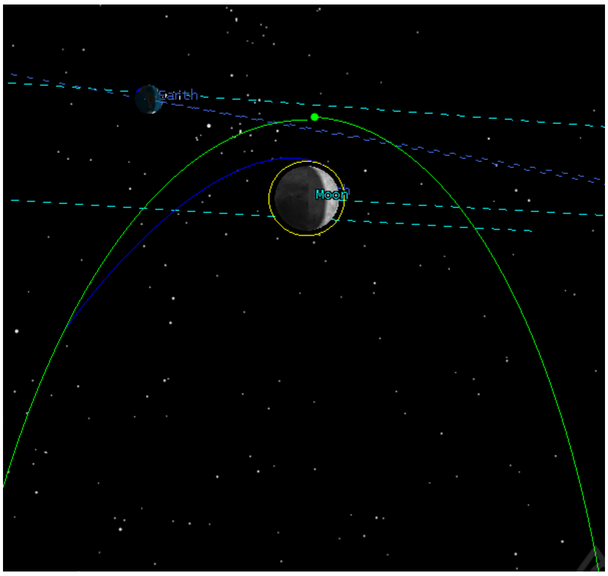


Figure 2.5: NRHO to LLO transfer at a T.A of 240°

For cargo missions transfer duration is less important than the ΔV cost. Therefore, departure travel time is chosen as 1.5 days with cost of 650-700 m/s.

Mission	Departure True Anomaly	Travel Time	DeltaV
Crewed	230-250	< 0.5 days	720-750 m/s
Cargo	190-210	1.5 days	650-700 m/s

Table 2.3: Chosen ΔV - TOF values for DSG to LLO

2.2.2 Between Equatorial-Polar Sites and Equatorial Sites

There are two possible strategies to reach non-polar sites of the moon surface. One is to travel to a polar low lunar orbit first as mentioned polar sites strategy. Due to natural precession of the polar orbits, any location on the surface is accessible without any plane change maneuvers [22]. For instance, 10 days coverage of a 100 km Polar Low Lunar Orbit can be seen in Table 2.7

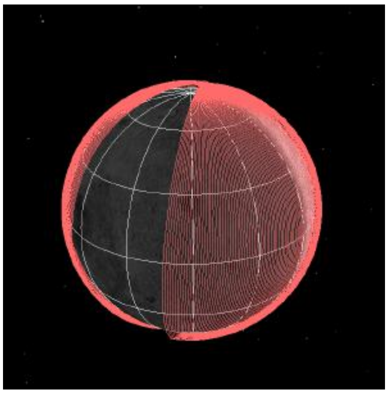


Figure 2.6: 10 days Precession of 100 km PLLO

Thus, JELLY can reach anywhere on the moon if the mission is willing to wait, but this strategy increases the mission duration up to 15 days and exceeds the margin for crewed missions. The other strategy is to perform a plane change maneuver which requires a high ΔV. Trade studies show that making plane change maneuvers to satisfy requirement P6 is much better than waiting at the low lunar orbit in terms of astronaut safety, total mass and so on.

Consequently, JELLY will perform a combined maneuver(inclination and perilune change) to reach a non-polar LLO.

Inclination change is directly related to the orbital velocity, hence departure true anomaly must be closer to the apoapsis in order to decrease the ΔV cost. Some of these maneuvers can be seen in Table 2.4

Table 2.4: Various Combined Inclination Change Maneuvers to 100 km LLO

True Anomaly	**Inclination Change**	**Inlication Change ΔV [m/s]**
220	22	120
240	22	167
240	30	200
200	90	245
180	90	153

As it can be observed on Figure 2.7, in order to increase vehicle efficiency, optimal ΔV - Time of Flight combination was targeted. In this case departure true anomaly is a function of required inclination change.

Consequently, if JELLY has to land on polar sites there will be no inclination change and departure true anomaly will be around 240, and if it has to land on equatorial sites departure true anomaly will be closer to apolune to decrease inclination change cost.

JELLY mission strategy is summarized in Table 2.5.

Table 2.5: Landing Site Strategy Summary

Landing Site	**Inclination Change**	**Departure True Anomaly**	**Total Delta V [m/s]**	**Travel Tim to LLO**
Polar Sites	0	240	720-750	<0.5 days
Between Polar and Equatorial Sites	0-90	240-180	720-950	0.5 to 3 days
Equatorial Sites	90	180	950	3 days

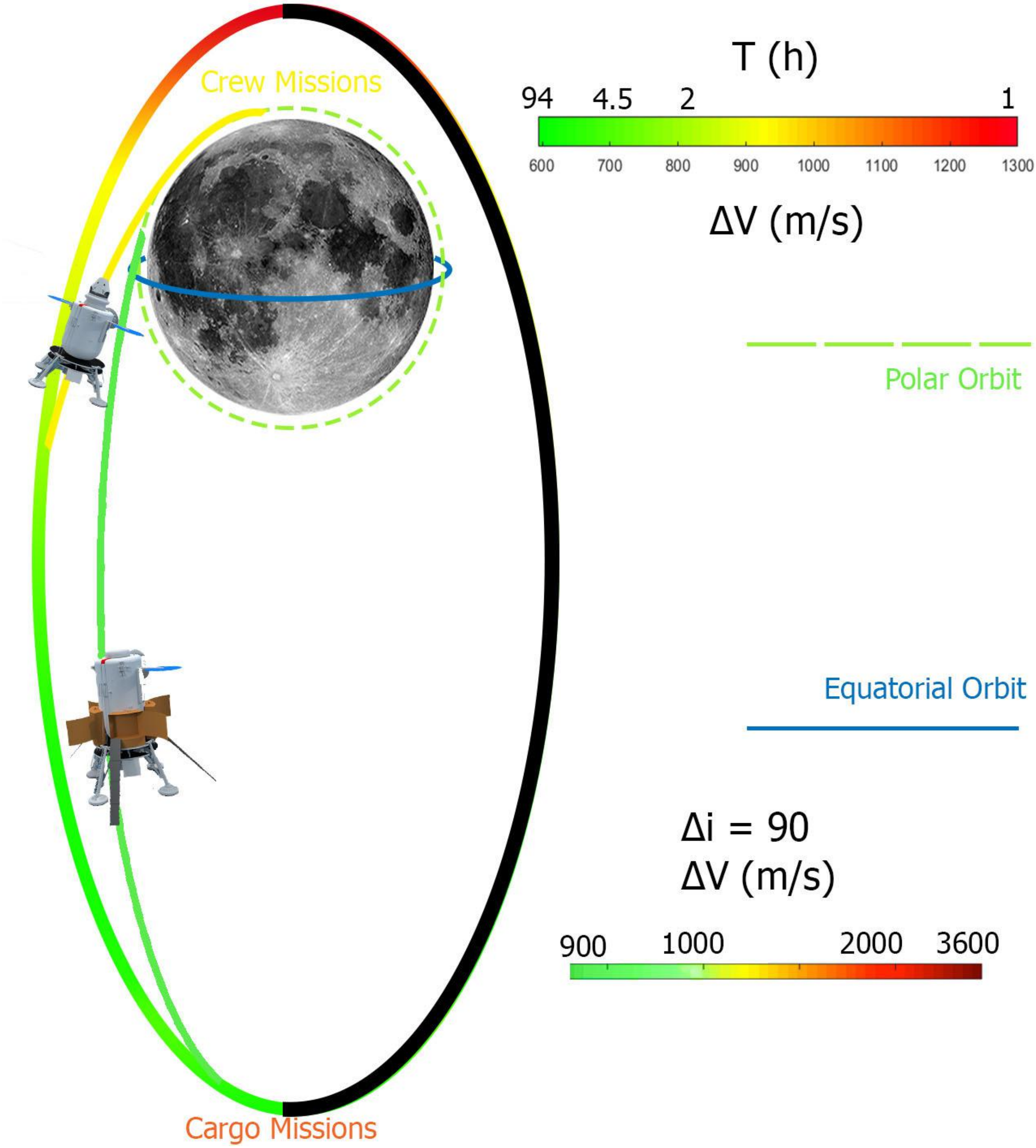


Figure 2.7: Departure True Anomaly and Combined Manuever Relations for Mission Modes

2.3 Lunar Landing

2.3.1 Low Lunar Orbit

100 km LLO to surface of the moon was chosen using the information obtained from Apollo missions and from recent studies about lunar landing. Due to its close passages to the surface, low period, and 40 minute travel trajectory to the lunar surface along with the experiences from Apollo missions, 100 km LLO can be used as a transfer orbit. [23] [24] [25]

2.3.2 Landing Trajectory

After arriving at the 100 km circular LLO and waiting for the right opportunity, landing trajectory can be achieved by one impulsive maneuver. De-orbiting maneuver costs 80 m/s and results in a surface transit duration of 40 minutes.

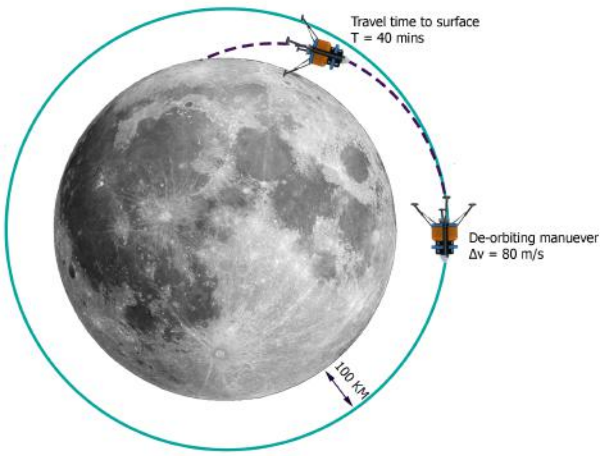


Figure 2.8: Landing Trajectory from 100km LLO to Surface

2.3.3 Landing

The landing strategy of Apollo missions consisted of several phases including breaking, approach and transition to landing. [26] Lunar Module Descent Mission Design.] However, this strategy was inefficient due to lack of experience and technology. With the developments in GCN technologies, better strategies can be developed. Recent studies show that instead of dividing the trajectory into different phases, landing in a constant parabolic trajectory is a more effective strategy. [26]

It is possible to achieve 10% efficiency by constantly increasing the flight path angle as the JELLY gets closer to the surface starting at an altitude of 15 km and a flight path angle close to 0. [26]

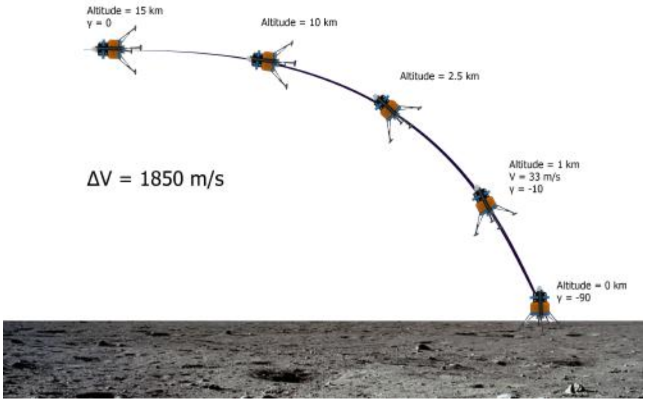


Figure 2.9: Safe and Efficient Landing Strategy

Table 2.6: Landing Cost Summary

Manuever	$\Delta V [m/s]$
De-orbiting	80
Landing	1850

2.4 Lunar Ascension

After performing its operations on the surface for 24 hours, JELLY will start its ascent from the lunar surface in order to rendezvous back with the DSG. The ascension phase consists of three stages; vertical rise, 15x100 elliptical orbit insertion, and circularization.

In order to perform an orbit injection, the vehicle must be flying parallel to the lunar surface, so a flight path angle must be given at the right time. The force of gravity will then gradually transform the flight from vertical to horizontal. The equations 11.6, 11.7, 11.8, and 1.36 from Orbital Mechanics for Engineering Students book that describe a “gravity turn” and take the mass and gravitational acceleration variations into account were numerically solved in order to simulate the ascension trajectory of the vehicle and calculate the gravity losses which aided in finding the required propellant[7]. Since the Moon has no atmosphere, there is no drag loss and consequently no heating, which makes the whole ascension/descension processes relatively easier.

Results indicate that for an initial flight path angle of 72° given at 100m altitude will result in a final path angle of 5.4° at 48 km altitude with a final velocity of 1.666 km/s after a gravity loss of 0.191 km/s. The Orbit velocity of an 15x100 km Orbit at this point is 1.661 km/s, therefore we can observe that it is possible to inject into Orbit with these conditions and proceed with the calculation of the required amount of propellant.

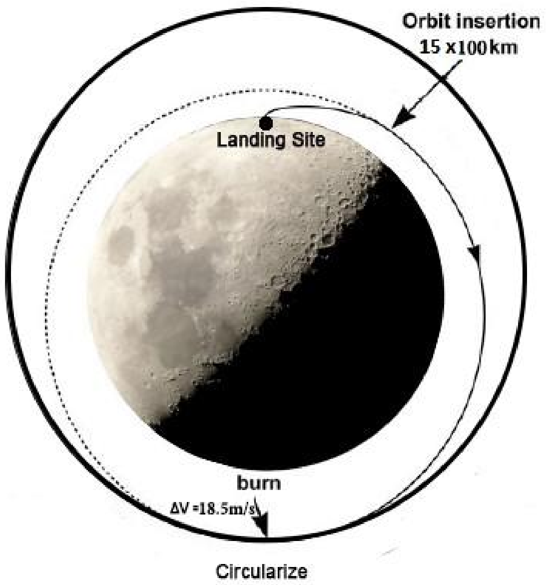


Figure 2.10: Ascension Visual

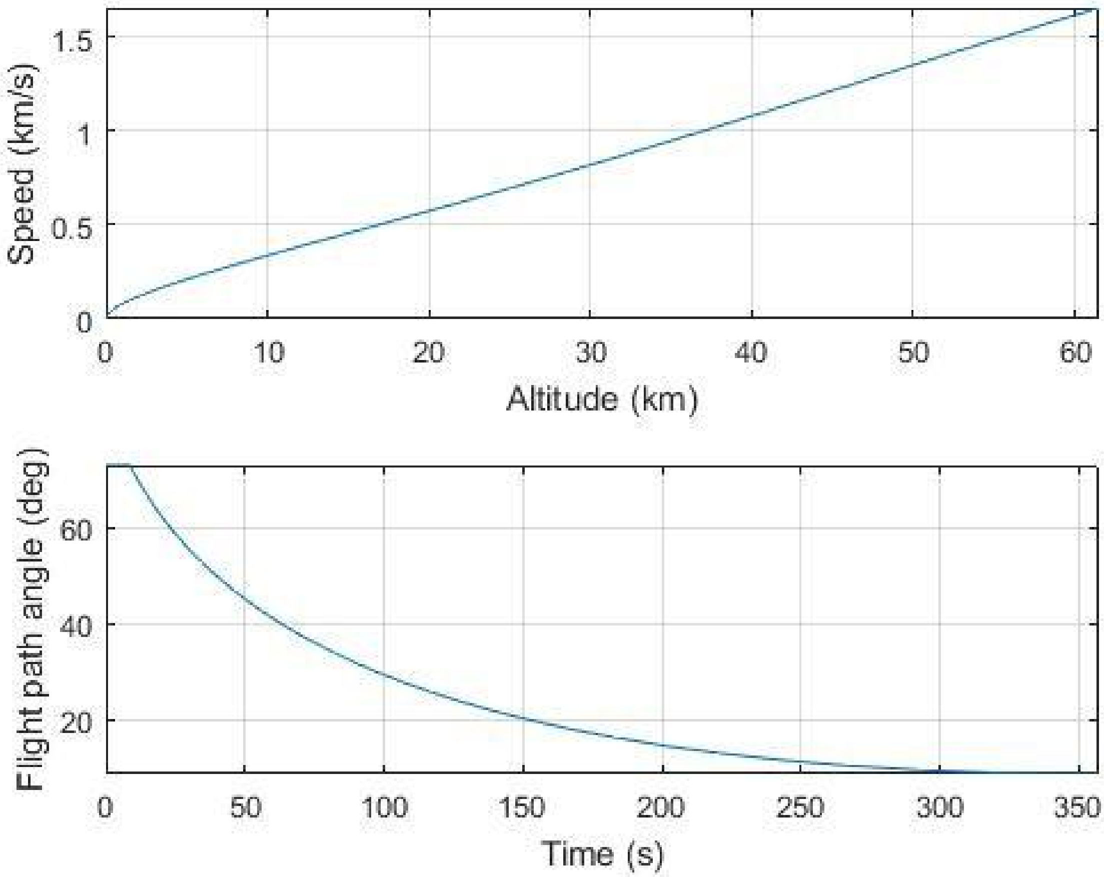


Figure 2.11: Ascension Calculations

The vertical rise stage lasts until the vehicle reaches an altitude of 100m, which takes approximately 9 seconds. Then a Single Axis Rotation (SAR) maneuver that takes around 10 seconds is performed by calculating a single-axis time optimal rotation between the initial attitude and the given final attitude command, followed by a Powered Explicit Guidance (PEG) stage to insert the vehicle to the targeted orbit[8] . The desired orbit has an apolune of 100 km and a perilune of 15km. The reason for choosing this perilune is because it is considered the lowest safe altitude[9]. Following the insertion into this Low Lunar Orbit (LLO), in order to fix any out-of-plane errors that might have occurred during ascent, JELLY performs a nominal corrective combination maneuver (NCC). Then a burn will occur when JELLY reaches the first apolune to circularize the orbit.

This is the main strategy that will be used in order to launch JELLY to a LLO prior to rendezvous with DSG. However, there is one small problem. The vehicle must achieve the correct ascending node as its orientation with respect to the initial polar orbit changed slightly due to the rotation of

Moon around its own axis. It can be done by either yaw steering during ascent to achieve both the ascending node and inclination of the target orbit, or by performing an ascent to a combination of desired inclination and node, and subsequently an on-orbit plane change maneuver with minimum ΔV. Out of these two methods, the latter is the most beneficial one in terms of total ΔV cost2. Therefore, JELLY will utilize the plane change method, which results in a total ΔV penalty of 100 m/s for 24 hours.

Trajectory Event	Estimated Duration (s)	ΔV (m/s)	Resultant Apoapsis/Periapsis (km)
Liftoff	10	-	-
Orbit Injection	370	1872	100/15
Plane Change	-	100	100/15
NCC	3420	10	100/15
Circularization	3540	18.5	100/100
Total	2h	2000	100/100

Table 2.7: ΔV - TOF Table for Lunar Ascension Phase

2.5 From LLO to DSG

Another critical part of this mission is returning to the DSG from the lunar surface. A trajectory between Low Lunar Orbit to DSG must be defined in order to achieve this goal. Simulations show that the rendezvous position of JELLY and DSG does not change with departure true anomaly. Decreasing departure true anomaly will increase the travel duration, but the DSG will also travel in its orbit during this time. Thus, no matter what the departure true anomaly is, after JELLY spends 24 hours on the surface, the DSG will be at a true anomaly around 140- °. Fuel efficient transfer trajectory between LLO and DSG orbit was obtained by solving Lambert's problem, which is used to find the optimal transfer trajectory if travel time and two position vectors are given. [27] [28] The simulation calculated the initial position vector of JELLY in LLO and the final position vector of DSG for the optimal transfer duration.

Figure 2.8 shows the trajectory from LLO to DSG. Optimal trajectory was found as 60 hours of travel duration and 750-800 m/s of ΔV. To summarize, every mission with a duration of 4-5 days will use this trajectory. An unexpected increase in mission duration, however, has no impact on the JELLY design. In fact, a delay may result in a decrease in ΔV due to the higher true anomaly for rendezvous.

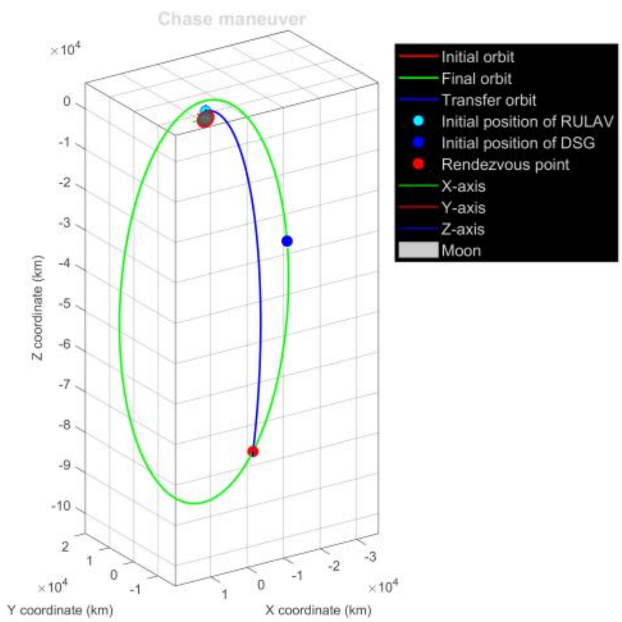


Figure 2.12: Rendezvous Trajectory from 100km LLO to DSG

2.5.1 Summary of Transfer Trajectories and ΔV Budget

Table 2.8: Crewed Mode Transfer Summary

	Near Side Equator		**Near Side Polar**		**Far Side Polar**		**Far Side Equator**	
	ΔV (m/s)	T (h)	ΔV (m/s)	T (h)	ΔV (m/s)	T (h)	ΔV (m/s)	T (h)
NRHO to LLO	950	66	750	5	750	5	950	66
LLO to Surface	1950	1	1950	1	1950	1	1950	1
Surface to LLO	2000	1	1900	1	1900	1	2000	1
LLO to NRHO	800	60	800	60	800	60	800	60
Total	5700	128	5400	67	5400	67	5700	128

Table 2.9: Cargo Mode Transfer Summary

	Near Side Equator		**Near Side Polar**		**Far Side Polar**		**Far Side Equator**	
	ΔV (m/s)	**T (h)**	ΔV (m/s)	**T (h)**	ΔV (m/s)	**T (h)**	ΔV (m/s)	**T (h)**
NRHO to LLO	850	88	700	68	700	68	850	88
LLO to Surface	1950	1	1950	1	1950	1	1950	1
Surface to LLO	2000	1	1900	1	1900	1	2000	1
LLO to NRHO	800	60	800	60	800	60	800	60
Total	5600	150	5350	130	5350	130	5600	150

3. Crew Mode

3.1 Requirements

When operating in crew mode, the RLSAV should fulfill the requirements given in Figure 3.1.

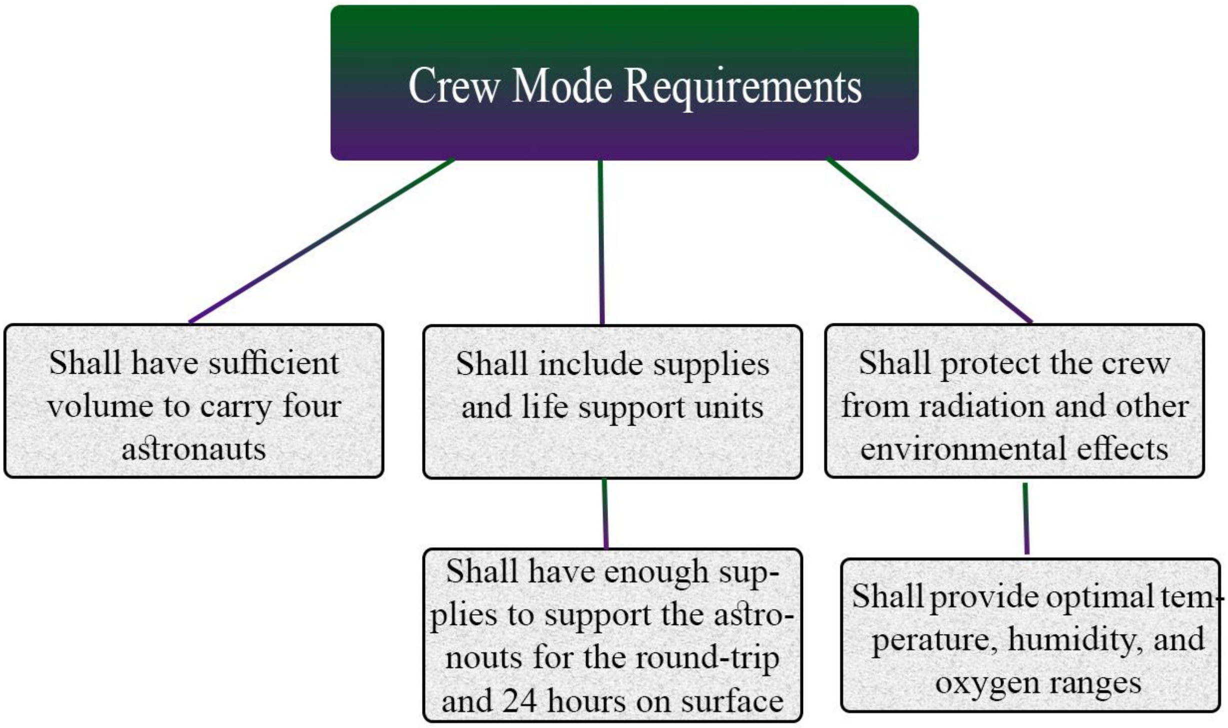
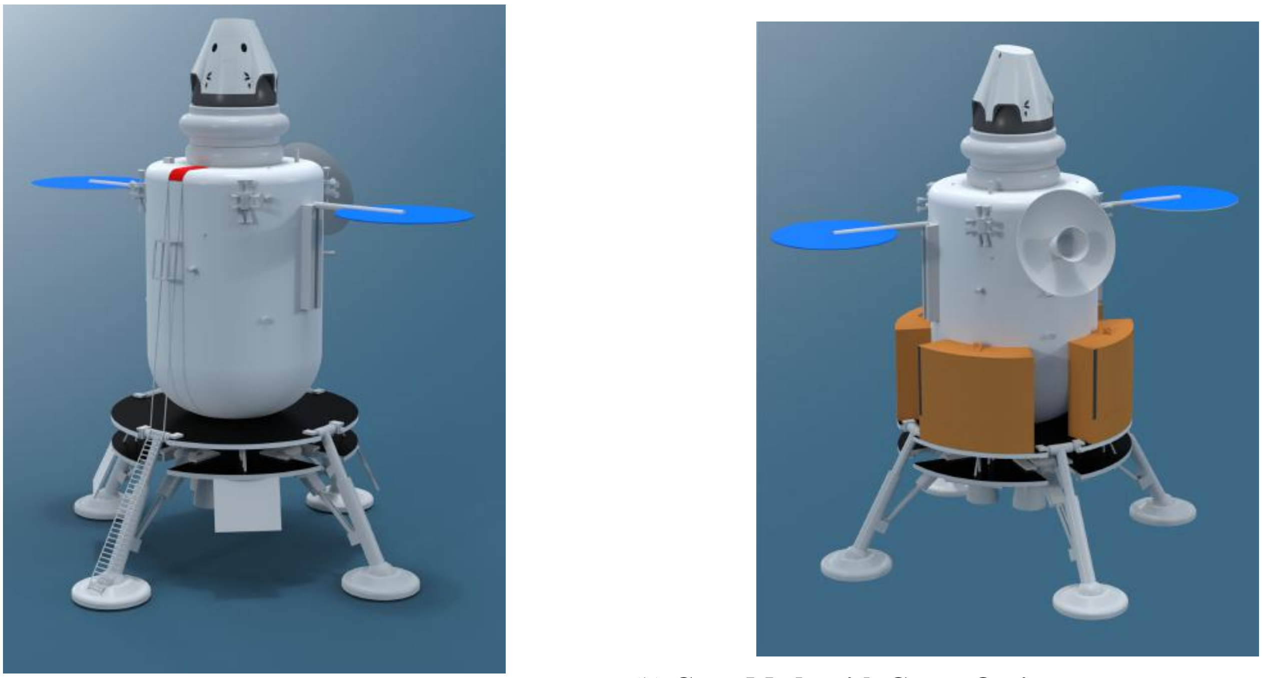


Figure 3.1: Crew Mode Requirements

The vehicle was designed to be able to switch between crew and cargo modes easily to satisfy RFP requirements (P4). Therefore, it was determined to use a crew command capsule for crewed missions and leave it docked to the DSG for cargo missions. This way, it was aimed for the vehicle to operate effectively in the desired mission mode with maximum performance and minimum unnecessary components. Figure [] shows the configurations of JELLY spacecraft for crewed missions.



(a) **Crew Mode Without Cargo**
Cargo compartments are removed in this configuration to reduce propellant usage when not necessary

(b) **Crew Mode with Cargo Option**
This configuration allows some payload to be carried with the crew

Figure 3.2: Configurations for Crewed Missions

Since humans will be transported, using an existing crew capsule was preferred for numerous reasons with safety being top priority, as a capsule with space heritage would be a lot more reliable than any system in design phase. Other factors included low cost, and no extra manufacturing and testing processes. Thus, a trade study was performed to determine which capsule was going to be used. Table 3.1 lists the available choices and Figure 3.3 shows the Analytic Hierarchy Process. Note that the "Pure Capsule Mass" is the mass calculated by removing the unnecessary components for the mission such as parachutes and heat shields.

Capsule	Manufacturer	Dry Mass (kg)	Pure Capsule Dry Mass (kg)	Max. Crew Capacity	Active Crew Support Time (days)	Design Life (days)	TRL
Dragon 2[10,11,12]	SpaceX	9252	4200	7	7	210	8
Orion[13,14]	Lockheed Martin	10160	8595	6	21	210	8
CST-100[15,16]	Boeing	13000	8100	7	2.5	210	7
Federation[17]	Roscosmos	12000	7000	4	14	200	4

Table 3.1: Crew Capsule List

CST-100 was instantly dropped out of the list because it did not satisfy the minimum mission duration which was considered 5 days for the worst-case scenario.

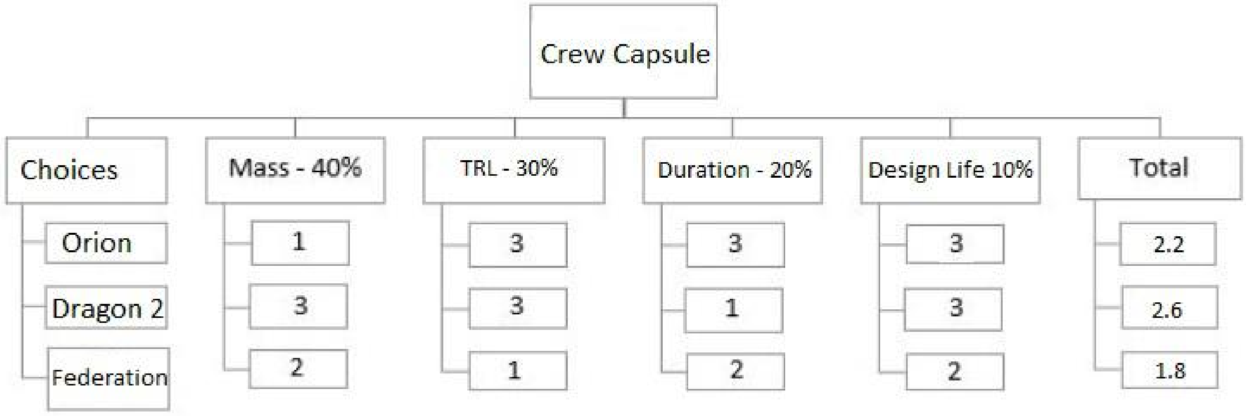


Figure 3.3: Crew Capsule Trade Study

Consequently, crew capsule of Dragon 2 was chosen to be the crew capsule as it satisfies the mission requirements. It has the capacity to support 7 astronauts for 7 days, and can support 4 astronauts for even longer which is sufficient for the mission duration plus a margin. Moreover, it is fully autonomous and can provide real-time information such as position, destination, or environmental information [53]. Therefore, Dragon can be used as a back-up system for crewed missions which greatly reinforces the reliability and safety. A schematic of the Dragon 2 is given in Appendix A.

3.2 Life Support

The life support system is an integral part of the mission and it deals with maintaining people in a closed environment far away from the Earth. Breathable air, usable water, and food are the most fundamental requirements the life support unit should provide for the crew. The life support system was designed to satisfy the crew's needs by providing them a safe and reliable environment. All requirements for a four-day long Lunar Mission for a crew of four are listed in Table 3.2.

Time of Flight and Standby Time(Day)	4			
Crew Member(CM)	4			
ITEMS	**Mass (kg/CM-day)**	**Volume(m³/CM-day)**	**Mass(Kg)**	**Volume(m³)**
Food+packaging(15%)	0.71	0.00871	11.36	0.13936
Water	9.08	0.00908	145.28	0.14528
Water Tankage	0.834	0.000834	13.344	0.013344
Carbon Dioxdide	1	0.00277	16	0.04432
Carbon Dioxdide Tank	0.31	0.00101	4.96	0.01616
Oxygen	0.84	0.00277	13.44	0.04432
Oxygen Tankage	0.31	0.00101	4.96	0.01616
Drink	1.62	0.00162	25.92	0.02592
Drink Tankage	0.834	0.000834	13.344	0.013344
SpaceSuit	221.525	0.135	886.1	0.54
Portable Life Support System	24.04	0.19	96.16	0.76
Medical Supplies	40	0.2075	160	0.83
		Totals	1390.868	2.588208

Table 3.2: Crew Requirements

Also Figure 3.4 shows the variation of total supply mass for 4 to 7 day missions. Thus, sufficient amount of supply for the appropriate mission duration can be used accordingly.

Figure 3.4: Crew Requirement Change With Mission Duration

4. Cargo Mode

4.1 Payload Information

When operating in cargo mode, the vehicle will be able to carry a payload of 15,000 kg to the lunar surface and 10,000 kg back to the DSG. For this operation, JELLY simply leaves the crew capsule docked to the DSG and leaves with its cargo compartments filled up with the desired payload. Upon return, it will again dock with the Dragon Crew Capsule.

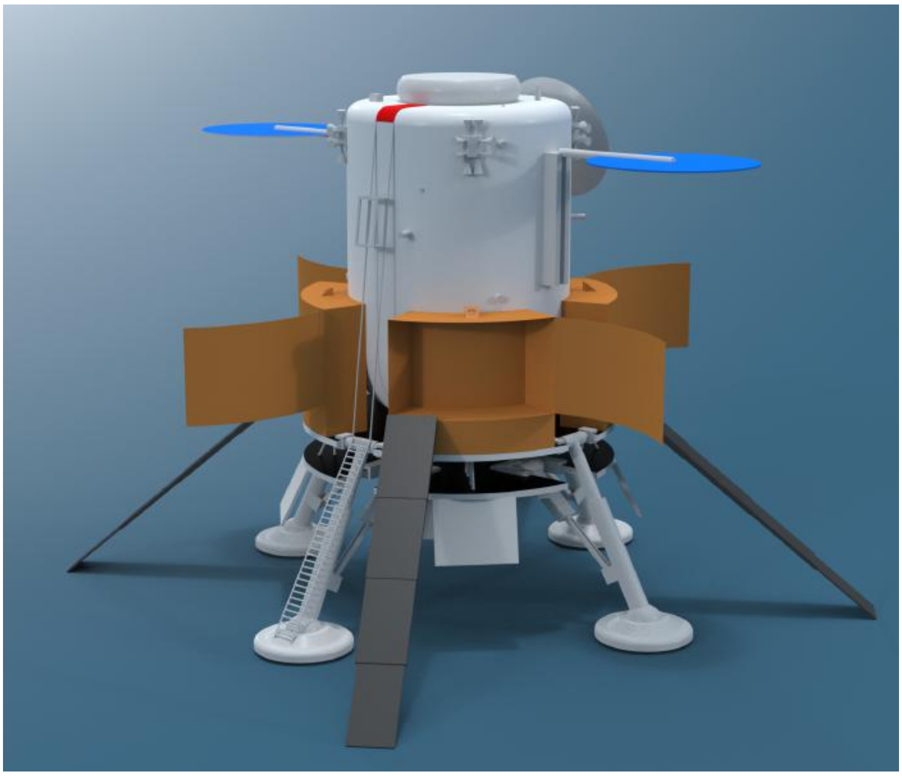


Figure 4.1: Cargo Mode Configuration for Unmanned Missions

Below are some information about the possible contents of the payload which are based on past missions and descriptions in the RFP, which explains how the decisions for the cargo compartments are made.

- **Habitat**

  The habitat elements provide a pressurized environment for the crew to live and work while performing their duties on the moon surface. The surface habitat consists of a Basic Habitat module, which provides the starting capacity for 4 crew members. For this mission, it is not possible to transport the entire habitat because of the design and capacity of spacecraft, but its individual components can be carried and assembled on the lunar surface.

- **Lunar Excavator**

  Performs lunar regolith mining and transport, regolith oxygen extraction and oxygen storage and distribution. Supports soil repellent cleaning and water production.

- **Rover**

  The Moon has one-sixth of the weight of the Earth, so it will have a hard work that resists and remains a lightweight mobile drill. The so-called regolith soil is abrasive and small, so when a drill hits the ice it will probably be in concrete consistency.

Table 4.1 lists a possible payload configuration along with the dimensions and masses of the contents.

Payload	volume(m^3)	mass(kg)	Dimension(m)
Oxygen Tankage	20	6138.61	4x2*2.5*1
Hydrogen Tankage	20	6138.61	4x2*2.5*1
Hydrogen	0	902	0
2 of Oxygen Excavator Plant	9	420	2x2*1.8*1.25
4 Mining rover	3.375	320	4x1.5*0.75*0.75
Habitat Components	1.1516	140	1*1*1.1516
TOTAL	53.5266	14059.2	

Table 4.1: Payload Configuration

4.2 Analysis of the Cargo Compartments

The minimum length of the cargo compartments are constrained with the vehicle diameter and is 1.25m. To analyze the cargo compartments, ANSYS Mechanical APDL was used. Figure[] shows that the compartments can carry the payload safely.

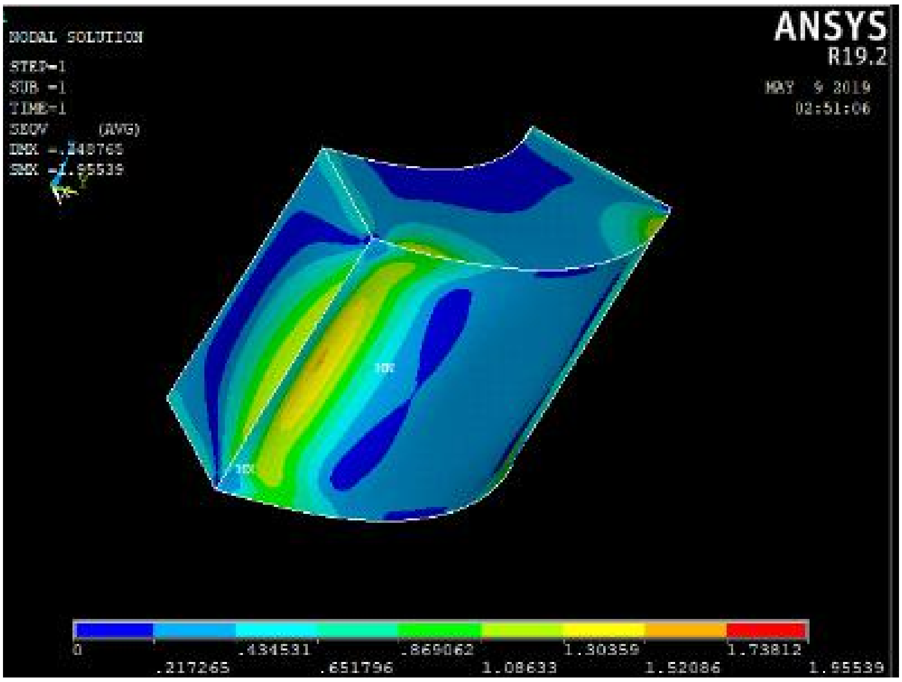


Table 4.2: Cargo Compartment Load Analysis

5. Design of Spacecraft Subsystems

5.1 Propulsion System

5.1.1 Requirements

Main purpose of the propulsion system is to provide velocity change range of 5450m/s to 5750 m/s to the spacecraft for one round trip, as indicated in Conops . It also should be able to produce the necessary thrust for the spacecraft to lift-off from the Lunar surface and accelerate it towards the target orbit with the necessary orbital velocity at that point. In addition, it is required to be capable of restarting since the spacecraft must land to lunar surface and take off again.

5.1.2 Trade Studies

The first decision to be made when designing the propulsion subsystem was selecting the type of the system. Ion propulsion was eliminated at the beginning because it is applicable to low thrusting cases, and it requires longer times to accelerate. Nuclear propulsion is also not suitable since it has a low thrust-to-weight (T/W) ratio and the mission is manned. Based on these criteria, the most convenient choice is chemical propulsion. Among chemical propulsion systems solid motors were out of consideration since neither restart nor shutdown is possible. In hybrid and liquid systems it is possible to restart the engines, but hybrid systems are not as reliable as liquid systems for such conditions. Thus, liquid propulsion system was decided to be used. Since the mission requires 10000 to 15000 kg of payload and 5450 to 5750 m/s of V, the amount of propellant is the major sizing factor for the vehicle in terms of both mass and volume. A trade study was conducted between 6 bipropellant pairs. Main criteria are toxicity since the mission is manned, reusability since the vehicle will be used for a long time, final mass and volume. Another important point is that oxygen and hydrogen extraction plans from water resources located at the lunar poles.[Concept for a Crewed Lunar Lander Operating from the Lunar Orbiting Platform-Gateway]General properties of the pairs are shown in Table . From there, LOX/ LH2 pair has the biggest volume but this was not an issue since with 484 $\mathrm{m}^3 tankvolumevehiclecouldfitwelltoSLS.AndalsoLOX/LH2hasthelowestmassamonganyotheroptionsand$

Table 5.1: Bipropellant Properties

Bipropellant Properties						
Bipropellant	**Isp (s)**	**MR**	**Toxicity**	**Reusability**	**Propellant Mass (kg)**	**Total Tank Volume [m3]**
LOX/LH	**449.7**	**5.5**	**None**	**Yes**	**133281**	**404**
LOX/LCH4	**350**	**2.77**	**Low**	**Yes**	**210871**	**267**
LOX/RP-1	**353**	**2.29**	**Moderate**	**No**	**207329**	**203**
N2O4/MMH	**336**	**1.73**	**High**	**Yes**	**228944**	**197**
N2O4/N2H4	**339**	**1.28**	**High**	**Yes**	**224842**	**187**
N2O4/Aerozine-50	**302**	**1.59**	**High**	**Yes**	**286686**	**247**

After deciding the propellant type, a deep research has been conducted among LOX/LH2 engines that have space heritage. Other constraints were the restart capability of the engine and throttling in a single mission. With all of this constraints, RL10C-1 engine appeared to be the most suitable one. Engine specifications for RL10C-1 are shown in Table .

Table 5.2: Engine specifications for RL10C-1

Isp (s)	449.7
MR	5.5
Thrust (N)	101820
Nozzle Diameter (m)	1.45
Length (m)	2.18
Mass (kg)	191

5.1.3 Propellant Mass Calculation

Main purpose of the propulsion system for the ascent phase is to lift the spacecraft and its 10,000 kg of payload from the Lunar surface, injecting them to a 15x100 km elliptical orbit, followed by circulization of the said elliptical orbit to 100x100 km by orbital maneuvers, and finally transferring the vehicle to NRHO for rendezvous with the DSG. In order to obtain the most accurate values possible for the propellant mass, calculations were made backwards with the first step being the transfer from the 100-km circular LLO to NRHO. Propellant mass necessary for this maneuver was calculated from the Tsiolkowsky rocket equation.

The next step was to calculate the propellant mass needed for the transfer from 15x100 km elliptical orbit to 100-km LLO, which was determined to be 1806 kg from Tsiolkowsky rocket equation. For the Lunar ascention phase, the code that simulates the lift-off was used both to calculate the propellant mass and to determine the engine quantity, as explained before . In order to reach the 15x100 km elliptical orbit with the necessary orbital velocity at that point, and a final flight path angle close to zero it was determined that 33000 kg of propellant and four engines will be required. This step was conducted by numerically solving the following equation which is a useful expression which includes the Tsiolkowsky rocket equation and losses caused by gravity.

The final step was to determine the propellant needed for Entry, Descent and Landing. This was also calculated from Equation 5.1. All steps are tabulated in Table .

Table 5.3: Propellant Mass and Trajectories

	Equation Used	**Final (Burnout) Mass**	**Propellant Mass**
Step 1: LLO to NRHO	5.1	bus mass + ascent payload	mp,1
Step 2: Surface to LLO	5.2	bus mass + ascent payload + mp,1	mp,2
Step 3:LLO to Surface	5.1	bus mass + descent payload + mp,1 + mp,2	mp,3
Step 4: NRHO to LLO	5.1	bus mass + descent payload + mp,1 + mp,2 + mp,3	mp,4

Table 5.4: Propellant Mass and Trajectories

	ΔV **(m/s)**	**Final Mass (kg)**	**Propellant Mass (kg)**
Step 1: LLO to NRHO	800	50958	10132
Step 2: Surface to LLO	2000	61090	34806
Step 3:LLO to Surface	2000	100900	57872
Step 4: NRHO to LLO	950	158770	38150

From ConOps, it can be seen that the most expensive round trip requires a V of 5750 m/s. This value of velocity change gives the propellant mass which is the determining point for system design. This critical propellant mass is calculated by following the steps indicated in Table . The results are shown in Table . Total usable propellant mass is 140960 kg.

In full throttle, four RL10C-1 produce 407280 N of thrust. Since the lift-off mass is 95896 kg, which is the dry mass plus ascent propellant, this gives a T/W ratio of 2.62. T/W ratio of greater than 1 is the condition for lift-off. The spacecraft experiences 1.62gm at lift-off and 4gm just before burnout.

5.1.4 Propellant Inventory

Table [] shows the propellant inventory for a round trip. Note that this inventory was prepared by using the amount of propellant required for the most expensive trip, as shown in Table []. In addition to trapped propellant, outage and loading error, a 10 of margin, which is the margin for V, was used as emergency propellant [Elements of Spacecraft Design, AIAA Education Series, 2002]. This gives 161399 kg of loaded propellant in total.

Table 5.5: Propellant Inventory

PROPELLANT INVENTORY			
	Fuel	Oxidizer	Total
Usable Propellant (kg)	21686	119274	140960
Trapped Propellant (3%)	651	3578	4229
Outage (1%)	217	1193	1410
Loading Error (0.5%)	108	596	704
Margin (10%)	2169	11927	14096
Loaded Propellant (kg)	24831	136568	161399

5.1.5 Pressurization System

3% of the propellant is added to tank volume as ullage. Thus, oxidizer tank volume is 124m^3 and fuel tank volume is 360m^3, a total volume of 484m^3. Titanium was chosen as tank material. General information about the propellant tank is given in Table 5.6.

Table 5.6: Propellant Tank Properties

Material	**Titanium**
Yield Stress	880 MPa
Volume	484 m^3
Pressure	2.7 MPa
Thickness	6 mm
Shape	Cylindrical
Diameter	7.5 m
Length	13.5 m
Tank Mass	8390 kg height

5.1.6 Tank Design

In cryogenic propellants, boil-off is a critical point regarding with the design. In long durations, boil-off can be a serious problem. Area that will be exposed to heat should be minimal in terms of cryogenic thermal management. Furthermore, using common bulkhead to divide the oxidizer and fuel tanks provides a significant solution to the boil-off problem. Venting H2 removes more heat from its tank than venting O2. This common bulkhead will direct all heating to the LH2 tank where this excess energy can be removed by H2 venting. Thus, LH2 tank cools the LOX tank, this will reduce the boil-off of the oxygen almost to zero. Cryogenic Operations for Long-Duration (COLD) technologies allows a boil-off rate of approximately 0.1% per day, which is sufficient for a Lunar mission of about 45 days.A turbopump drives the pressurant into the propellant tanks in order to regulate the tank pressure. Helium was chosen as pressurant since it is an inert and lightweight gas. Composite overwrapped pressure vessels (COPV) was chosen as pressurant tank material. Table outlines the properties regarding with the pressurization.

Table 5.7

Initial Pressure	34.5 MPa
Final Pressure	3.8 MPa
Volume	42 m^3
Pressurant Mass	2404 kg
Diameter	2.15 m
Material	COPV
Yield Stress	850 MPa
Thickness	43.5 mm
Tank Mass	4630 kg

5.2 Power System

Electrical Power System (EPS) is critical for the operation of all other subsystems and crew life support. Its responsibilities are power generation, regulation, and distribution. Thus, EPS's requirements must be determined firsthand, then optimal primary and secondary systems must be chosen.

5.2.1 EPS Requirements

Power source was determined from the power consumption of each subsystem, and power storage was determined from the energy required for maximum eclipse duration. Power estimation for the spacecraft was made according to the Dragon, Apollo, Orion, Altair power systems and the knowledge of "Current Space Station initial power requirements for housekeeping loads for crew of eight is 25 kW." [Manned Spacecraft Electrical Power Systems, Simon, IEEE, Vol. 75 No. 3 1987, pg. 285] Since the mission requires transporting 4 crew members, power need is approximately 10-15 kW. Power outputs of Dragon, Apollo, Orion,and Altair are approximately 10 kW. Along with all these information and considering that Dragon 2 was chosen as the crew module, total power requirement was estimated as 11 kW. For the payload power, maximum need can be seen during the crew mission so the power consumption of Dragon 2 is taken which is 4kW [http://spaceflight101.com/spacecraft/dragon/]. Subsystem power requirement was found from the difference of total power and payload power. As for the contingency from the Table [ ] of Brown's Elements of Spacecraft Design book, 20% contingency was taken into account since the spacecraft is in the stage of PDR class and DP. Category and added to its subsystem power requirement since Dragon already has contingency.

From Elements of Spacecraft Design		**Estimation**
Subsystems	*Power Allocations(%)*	
Thermal Control	28	28
Attitude Control	20	15
Power	10	10
CDS	17	12
Communications	23	20
Propulsion	1	10
Mechanism	1	5
Total	100	100

Subsystems	**Power (kW)**
Thermal Control	1.96
Attitude Control	1.05
Power	0.7
CDS	0.84
Communications	1.4
Propulsion	0.7
Mechanism	0.35
Total	7

Estimated payload power	4 kW
Estimated subsystem power	7 kW
Margin	1.4 kW
Total power	12.4 kW

The subsystems' power allocations and requirements are shown in the Tables **??** and **??**. Percentages are taken form Brown's book and with the necessary modifications new values for the spacecraft was estimated. [Elements of Spacecraft Design, Brown, pg. 34]

Brown's book the percentages are taken and with the necessary modifications new values for the spacecraft was estimated.

From Elements of Spacecraft Design		Estimation
Subsystems	Power Allocations(%)	
Thermal Control	28	28
Attitude Control	20	15
Power	10	10
CDS	17	12
Communications	23	20
Propulsion	1	10
Mechanism	1	5
Total	100	100

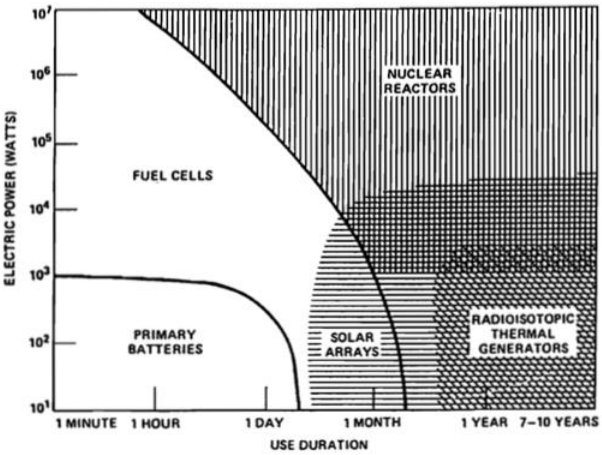
(a) **Estimated Power Breakdown Percentages**

Subsystems	Power (kW)
Thermal Control	1.96
Attitude Control	1.05
Power	0.7
CDS	0.84
Communications	1.4
Propulsion	0.7
Mechanism	0.35
Total	7

(b) **Power Breakdown Values**

5.2.2 Power Source

For the power source, solar panels, fuel cells, RTGs, and nuclear reactors were considered. Since the mission has long lifetime and mediocre power requirements, solar panels are well suited as seen in Figure **??** from Brown's book. This source also offers various advantages such as heritage, high reliability, and sustenance due to unlimited sunlight. Additionally, being renewable and environmental solution is a positive effect.



captionPower Sources

Solar Panels

ince the JELLY will perform a lot of lunar surface landing and orbital maneuvers, the solar arrays should withstand the effects and it should be able to stowed when necessary. Thus, ultraflex solar array system was chosen from the Northrop Grumman. It has heritage, high strength, high deployment reliability, lightweight. Ultraflex is also compatible with all solar cell technologies. For selecting the photovoltaic cells, a trade study was performed. Values other than mass of the C3MJ and C4MJ taken from the websites of the companies, those values found by linear interpolation of the given mass range according to thickness range in the XTJ prime data sheet. For this mission, efficiency is highly important so maximum weight given to that and from the lack of data small trade study is done and Spectrolab 4th Generation Triple Junction Solar Cell was chosen.

	Cell	**BOL Efficiency(%)**	**Mass(kg/m2)**	**BOL Voc (V)**
3***SPECTROLAB**	*C3MJ*	38.5	0.72	3.21
	XTJ Prime(50μm)	30.7	0.5	2.72
	C4MJ	39.8	0.76	3.125
2***AZURSPACE**	*Tj GaAs*	29.5	0.86	2.7
	Tj GaAs (80m)	29.5	0.5	2.7

	Cell	BOL Efficiency(%)	Mass(kg/m2)	BOL Voc (V)
3***SPECTROLAB**	*C3MJ*	38.5	0.72	3.21
	XTJ Prime(50m)	30.7	0.5	2.72
	C4MJ	39.8	0.76	3.125
2***AZURSPACE**	*Tj GaAs*	29.5	0.86	2.7
	Tj GaAs (80m)	29.5	0.5	2.7

5.2.3 Solar Panel Sizing

Table 5.9: Solar Panel Sizing

Solar Power Irradiance	1368 W/m^2
Efficiency	40%
Worst case solar angle	59.3
Degradation	2%
Lifetime	15 years
Area necessary to produce 13.2 kW	83m^2
Total area needed	83*3= 249m^2
Panel mass(non-structural)	136.95 kg

5.2.4 Power Storage

As per mission requirements, there is a possibility that JELLY may land on the far side of the Moon. In addition, there will be eclipse times when the spacecraft is operating in LLO. These conditions call for an alternative power supply for when the solar panels will be ineffective. For this reason, regenerative fuel cells, primary batteries, and secondary batteries were considered. From mass, cost, and volume analysis secondary (rechargeable) batteries were chosen.

Batteries

Among the rechargeable batteries it can be observed that Li- ion batteries are more advanced, lighter,and more durable with a high specific energy. Therefore, a trade study between Li-ion batteries was performed. From Table [] and Table [], Saft VL51ES Battery Cell was chosen.

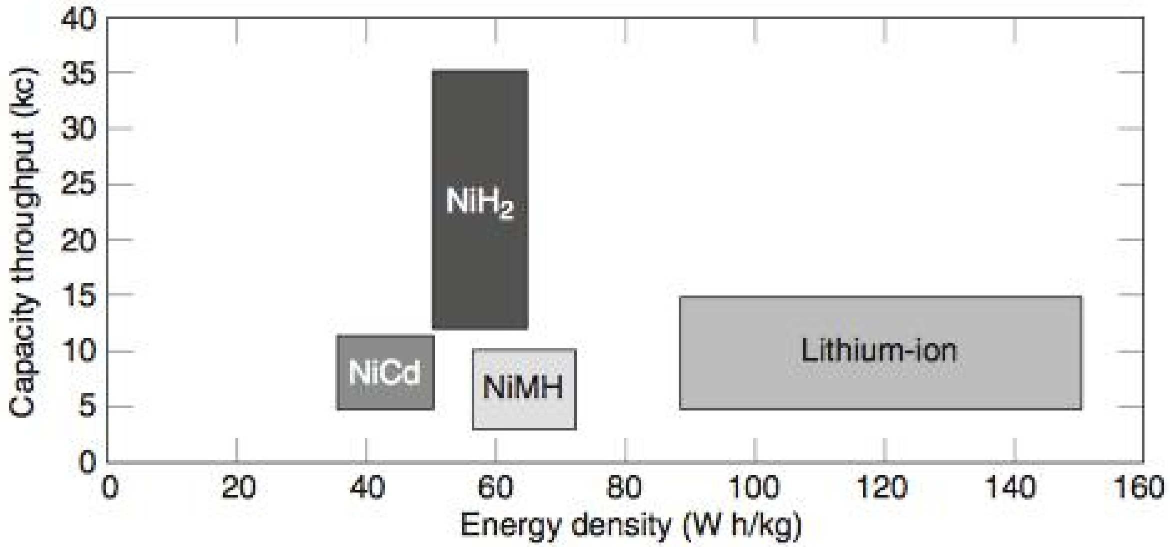


Figure 1.2. Relative life and energy density characteristics of common rechargeable battery types used in satellite applications.

Figure 5.1: Batteries

Lithium Batteries							
	EAGLE PICHER			SAFT			
	43 Ah Space Cell	Space Cell	SLC-028-01	VL 51ES	VES 180	VES 16	Weight
Energy(Wh)	195	245	128	182	175	16	2
Mass(kg)	1.27	2.02	1.02	1.08	1.11	0.155	1

Table 5.10: Lithium Battery Trade

Battery Sizing

Power requirement for the spacecraft was found according to the worst case scenario. Assuming 1 hour of eclipse during landing, 24 hours surface duration on the far side, 1 hour from lift-off to leaving eclipse region, and a margin of 1 hour, the worst case is assumed to be 27 hours of complete eclipse.

Thermal	27h
Attitude	3h
Power	27h
CDHS	27h
Communication	27h
Propulsion	30min
Mechanism	24
Payload	27h

From the required energy with the 65% depth and 0.95 efficiency Cp found as 455 kWh. 65% DoD were taken because of safety and longer life time. After adding 10% margin to Cp the cell's energy and the mass was solved. It is concluded that 2754 cell is necessary and total mass is 2974 kg. For battery packing 34 cells are connected in series to achieve desired operating voltage which sums up 122.4V. Parallel connections are as 18 of two in parallels and 15 of three in parallels. There are 33 batteries in total. Packed battery mass is found as 3361 kg.

5.2.5 Power Distribution and Control

Power distribution and control is necessary to deliver power to the appropriate subsystems and to batteries when it is required with the right voltage and current. For JELLY, power conditioning and distribution unit (PCDU) from Terma was chosen. It has been used before and has high transfer efficiency.

It also includes the regulators for the solar panels and batteries, and has up to 3kW output power capability. So 5 of them will be used in this mission to control the power output of JELLY.

5.2.6 Mass and Volume Sizing

Table 5.11

Battery 3051 kg
Solar panel support 463 kg
Solar Panel 37 kg
Power distribution and control units total mass 868 kg

5.3 Navigation, Guidance and Control System

The requirements for the JELLY guidance, navigation and control systems are derived from AIAA RFP documents. First requirement is the determination of the JELLY position, orientation and attitude through docking, coasting and landing. The JELLY has to orient itself in order to make orbit maneuvers, and has to stabilize itself during safe, smooth surface landing. During docking and undocking operations JELLY will be an active vehicle and DSG will be a passive vehicle. The detailed guidance navigation and control system requirements are listed at Table 5.12;

Table 5.12: Guidance Navigation and Control System Requirements

Determination of the JELLY position, orientation and motion through space.
The JELLY shall change its orientation for orbital maneuvers
The JELLY shall make safe landing to the surface of the moon.
The active JELLY shall perform relative navigation for rendezvous, proximity operations and docking with the passive Deep Space Gateway.

5.3.1 Attitude Determination

Star Trackers

Star trackers are able to find the vehicle's attitude and position with respect to the distant stars. During cis-lunar space and between DSG and surface of the moon, star trackers will guide the JELLY. The considered star trackers are listed at Table 5.13. Only flight proven star trackers are considered with an only one exception. [1]

Table 5.13: Star Trackers

Manufacturer	Star Tracker	Power [W]	Dimensions	Mass [kg]	Temperature [Celcius]	Life Time	Heritage
Jena Optronik	ASTRO APS [1]	12	154 mm x 154 mm x 237 mm	2	-30 to +60	18+ years	Proven
Jena Optronik	ASTRO 15 [2]	10	192 mm dia x 496 mm	6	-30 to +55	15+ years	Proven
Ball Aerospace	CT-2020 [3]	8	-	3	-	-	2019
Terma Space	HE-5AS [4]	7	-	2.2	-40 to +70	-	Proven

Ball Aerospace CT-2020 is selected from four options besides it is the only non-proven option. Ball Aerospace offers 0.1 arcsec three axis accuracy with low cost, low power consumption, high performance compact star tracker. The main reason for the selection is CT-2020 can able to operate its determination operations while the moon in the field of view. [5] This feature is very critical feature for Deep Space Gateway operations. Moreover, expected availability 2019 this makes selection reasonable. In order to increase coverage 4 star-trackers will be used. 3 of them on the body, evenly separated, and 1 on the top for decent and ascent stages at the surface.

Sun Sensors

Secondly, in order to increase the attitude determination accuracy and to prevent star tracker failures, sun sensors will be used. Sun sensors are able to find orientation of the spacecraft according to the sun. Furthermore, information of the sun position according to the JELLY can be used thermal and power subsystems.

Table 5.14: Sun Sensors

Manufacturer	Star Tracker	Power	Mass [g]	Temperature [Celcius]	FOV [Deg]	Accuracy [Deg]
New Space	NFSS-411 [6]	37.5 mW	35	-25 to +70	140	0.1 <
Solar Mems	SSOC-A60 [7]	36 mW	25	-45 to 85	120	0.3 <

New Space NFSS-411 is selected as sun sensor since it has higher accuracy and field of view. 3 of them will be mounted on the body, evenly separated. This will be enough for 360 degrees coverage, one more will be mounted to the nose to increase the accuracy during ascent and decent from the surface.

Inertial Measurement Units

Thirdly, in order to increase accuracy and determine the body's angular rates Inertial Measurement Units will be used. IMUs will be measure translational and rotational motions. The considered IMUs are listed at Table 5.15

Table 5.15: Inertial Measurement Units

Inertial Measurement Unit	Mass [kg]	Power [W]	Operating Temperature [Celsius]	Size [cmxcmxcm]
Honeywell MIMU	4.7	32	-30 to 65	233(diamter)x169
NGC SIRU	7.1	43	-10 to 65	28.9x18x14.9
Space & Navigation CIRUS	13.5	25	-20 to 60	396(diameter)x203

The Honeywell MIMU is selected due to its low mass, high accuracy and reliable flight performance. 6 of them will be used for redundancy.

Deep Space Network

DSN has provided navigation data derived from spacecraft radio signals since the early years of Apollo Missions. DSN has capability of measure spacecraft position at moon orbit with 150 meters accuracy at 1AU. [11] JELLY will use DSN to determine its position and velocity through coasting phase, whenever JELLY is able to communicate with DSN.

Rendezvous and Docking Sensor

Rendezvous and Docking Sensor will be used for docking and undocking sequences to determine relative position and speed with respect to the DSG. Reliable Jena-Optronik's RVS (Rendezvous and Docking Sensor) is implemented. [12]

Table 5.16: Rendezvous and Docking Sensor

Manufacturer	RDS	Power [W]	Mass [kg]	Range	FOV	Accuracy [Deg]
Jena - Optronik	RVS 3000 [12]	85	14	1.5 km	40x40	<0.05

Landing Sensors

Most critical part of the mission is landing phase of the mission, especially for crewed missions. High accuracy and safety are primary concerns. NASA is developing a high accuracy, low risk landing measurement system called Landing and Hazard Avoidance Technology (ALHAT). ALHAT is adopted to the JELLY design and will include Hazard Detection and Avoidance Sensor, Terrain Navigation Sensor, Altimeter and Velocimeter Sensors to map the surface, measure the relative velocity and altitude with respect to the surface. This information will be combined by IMU, star tracker and sun sensor data to improve position and orientation determination accuracy. [13] ALHAT system mass and power consumption is taken from prototype and test models. [14] [15]

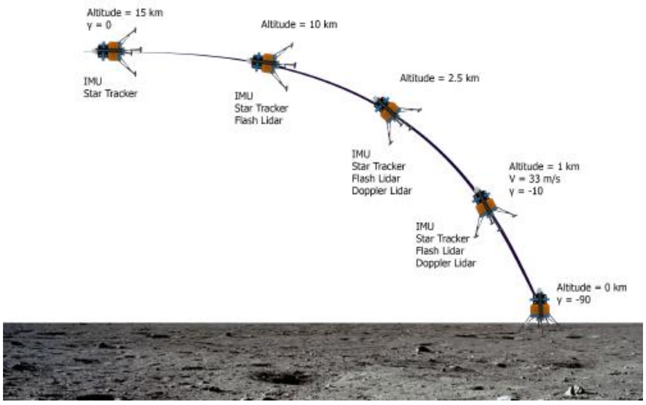


Figure 5.2: Landing Sensors on JELLY

5.3.2 Attitude Control

Table 5.17: Control System Requirements

The requirements for JELLY control system are to provide the necessary thrust for performing
Separation and Docking with DSG
Attitude change for orbital maneuvers
Attitude control for safe landing

The JELLY shall be able to land in its designated target without any risk to collide to surface. In order to control the JELLY only Reaction Control Systems are considered. Reaction Wheels and Control Moment Gyros are not considered since they would be heavy and mass is the primary limiting factor for the mission. Reaction Control Systems were used on the Apollo missions and they successfully completed their landing and docking operations. [18] Reaction Control Systems will be used for orientation change for orbital maneuvers, stabilization during ascent, landing, decent and stabilization during docking sequences. Previous spaceflights proved that RCSs are capable of all of the requirements mentioned. Moreover, docking and separation sequence will be assisted with Canadarm.

Reaction Control System

Studies show that evenly separated 4 groups of RCT configuration, as used on Apollo modules, is the most fuel efficient and accurate configuration. [17]

For the mission, most reliable engines are selected for trade study. The RCT selection factors are fast response time, high thrust to a minimum impulse ratio, long cycle life time. Due to high reliability, high performance and high thrust and long cycle time, R-4D engines are selected. To improve control during the landing one more RCT is added, 4 packs of 8 engines, total 32, will orient the JELLY, and will assist during docking and landing.

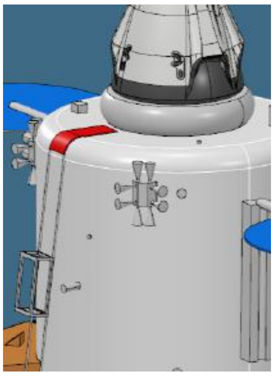
Table 5.18: Reaction Control Thrusters

Manufacturer	**RCT** [20]	**Thrust [N]**	**Mass [kg]**	**Power [W]**	**Isp [s]**	**Propellant**	**Total Pulses**	**Heritage**
Aerojet	MR-111C	5.3	0.33	8.3	229	Hydrazine	420000	Proven
Aerojet	R-6D	22	0.454	5	294	NTO / MMH	336331	Qualified
Aerojet	R-1E	110	2	36	280	NTO / MMH	330000	Proven
Aerojet	R-4D	490	3.4	46	312	NTO / MMH	20781	Proven
Aerojet	R-40B	4000	6.8	70	293	NTO / MMH	50000	-

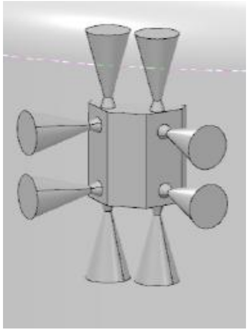
Number of reaction control thruster and required propellant calculated from Apollo and Space Shuttle missions. [19] Flight reports from Apollo are avaiable. Every translational, rotational, docking and landing thruster data were used for calculation. JELLY manuevers were estimated based on concept of operations such as how much translational, rotational meanuver JELLY needs. Furthermore, Brown's approach to calculations of RCSs is implemented. This calculations are combined with Apollo and Space Shuttle control systems capabilites. As a result, in order to reach same capability(angular accelaration capability), 32 R-4D thrusters are placed.

Apollo data, engine burn times and number of pulses, scaled to JELLY concept and expected number of pulses for a mission to the surface and back to DSG is estimated 1200 pulses and 1950kg propellant.

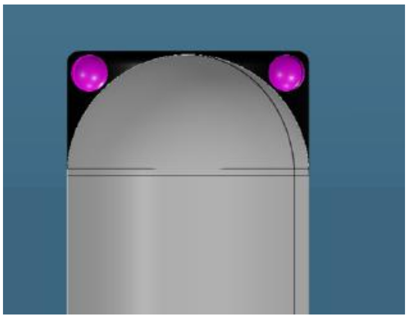
There will be 2 tanks and each tank will feed 2 RCS groups. If one tank fails other will all of them.



(a) Reaction Control Thrusters on JELLY



(b) RCT Configuration



(c) RCS Tanks

Figure 5.3: Reaction Control System

RCS thrusters are replaceable as groups. [18] For reusability, after every 15 trips(21000 total pulses) to surface RCS thrusters will be replaced as groups.

Canadarm

Canada Space Agency officially confirmed that they will participate with next generation Canadarm on DSG. [21] Canadarm will capable of Autonomous Docking System, so, docking and undocking sequences will be strengthening with the next generation Canadarm.

Figure 5.4: Canadarm and Dragon[NASA]

5.3.3 Summary

All these systems allow the JELLY to meet the requirement for the GN&C system.

Guidance, navigation and control System sensors on JELLY

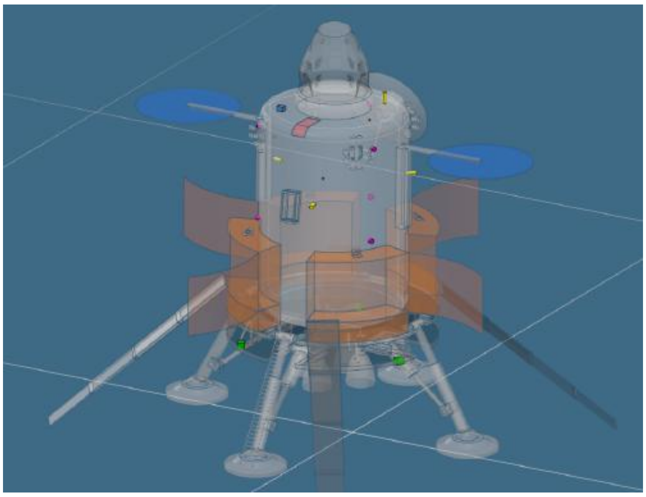


Figure 5.5: Guidance, navigation and control System sensors on JELLY
Yellow:Sun Sensors, Purple: Star Trackers, Brown: IMUs, Green: ALHAT, Blue: RVS 3000 Docking Sensor

Summary of guidance, navigation and control system can be seen at Table 5.19

Table 5.19: Total Power and Mass Allocation for Guidance, Navigation and Control Systems

Product	Quantity	Mass [kg]	Power [W]	Line Mass [kg]	Line Power [W]
Aerojet-R-4D	32	2	5	64	160
Propellant	2	975	-	1950	-
Ball Aerospace CT-2020	4	3	8	12	32
ALHAT*	1	15	70	15	70
RVS 3000	1	14	85	14	85
New Space 411	4	0.035	0.037	0.14	0.148
Honeywell MIMU	6	4.7	32	28.2	192
Total Mass [kg]	2083.34				
Total Power [W]	539.148				

5.4 Structure and Mechanisms

5.4.1 Requirements

The most important goal for the structure of the vehicle was to make it as light as possible for both crew and cargo mission modes, while maintaining the robustness and safety of the system. As the vehicle is expected to be able to switch between these two modes, and there is no additional payload requirement during the crew mode, making the cargo compartments removable is a feasible option in order to reduce the propellant usage when it is unnecessary. Thus, a design that allows the removal of the excess parts was employed.

First thing to do for designing a suitable structure system was finding out the limiting boundary conditions. The first one of those was the fuel tank. As it is explained in the propulsion section, the fuel tank has a volume of

The most optimal shape must be determined considering the landing operations, as a very long tank makes it difficult to maintain stability during landing. Second limiting factor was the dimensions of the launch vehicles. The longest diameter of the vehicle can not exceed 10m, which is the planned payload width of SLS Block 2. Since the cargo compartments were planned to surround the tank, a significant amount of space should also be left for them. Therefore, a diameter of 7m and a height of 10m for the tank was found appropriate. The fuel tank will also provide structural support. Thus, a thickness of 3mm with some variations with Titanium as the material was considered, which resulted in a mass of 3900 kg for the tank itself.

When it comes to the main structure, it is clear that some plates should carry the tank along with the cargo modules and also the landing gear. Thus, material selection becomes a major issue, as it should be extremely strong but not too heavy. Naturally, because of their high strength/density ratio advanced composites were the first choice. In the recent space missions, Aluminum honeycomb carbon-fiber composite material was the leading choice. It can be modified easily for different conditions like the thickness of laminas and areas of honeycomb hexagons. The challenging part of composite materials is manufacturing process, as they are ten times harder than simple alloys about FEM analysis. Because reality is going further than real in composite materials.

5.4.2 Design

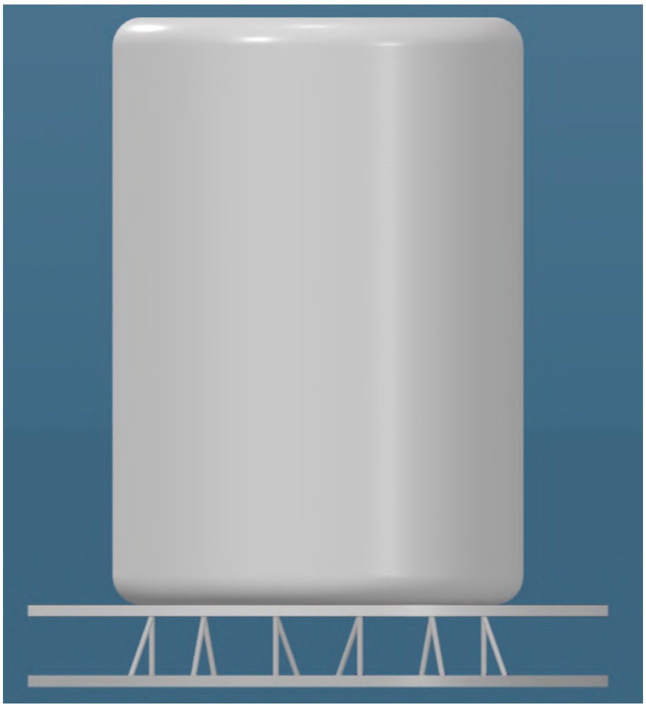
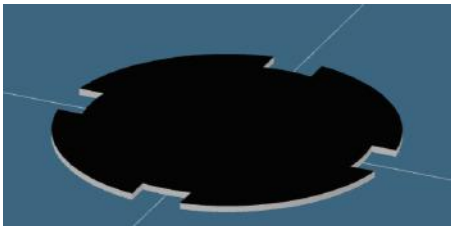
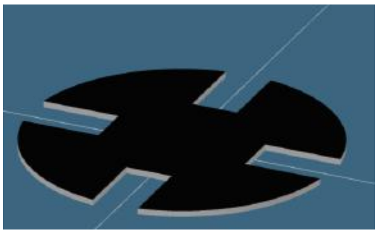


Figure 5.6: Fuel Tank and Truss System

Fig.[] shows the main structural elements of the vehicle. There are two plates and a truss system between them for two reasons. First was to create a stronger connection between landing gears and the vehicle. Bottom and top plates give a reliable landing system from two fixtures. Second reason was to give extra structural safety. The 1-meter space between the plates were reserved for engine connections and feeding systems. Plates can be made thinner by increasing fixing points of the truss system. Deep dive in parts one by one



(a)



(b)

(c)

Figure 5.7: Structure Elements

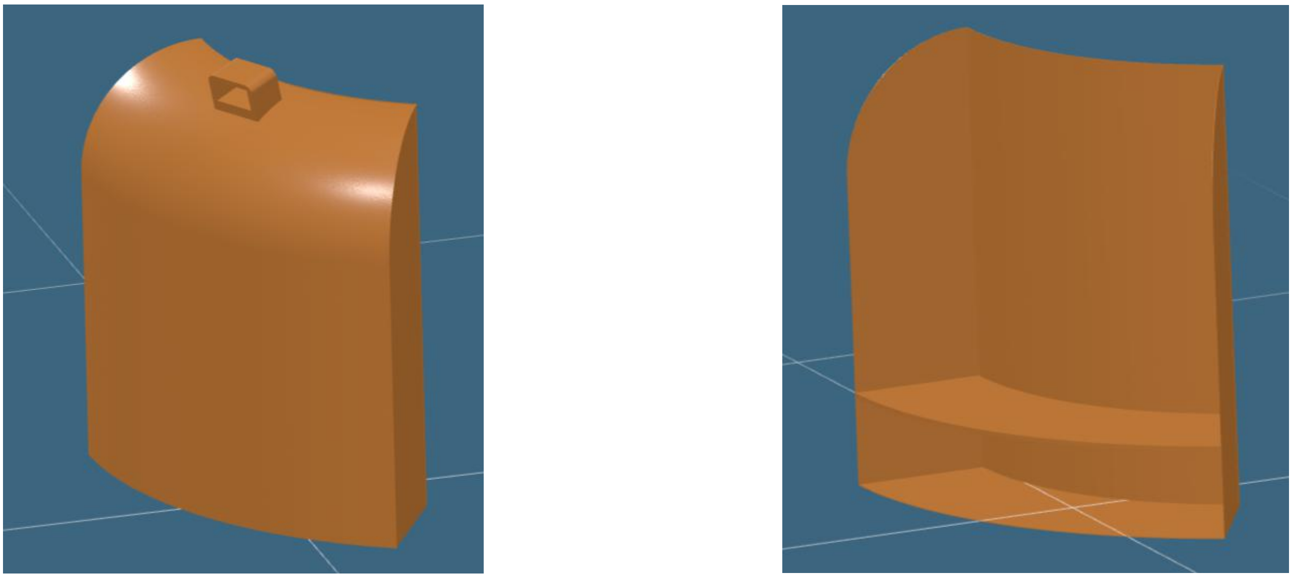
Fig.[] (a) and Fig.[](b) show the upper and lower load carrier plates, respectively. They have an outer diameter of 10 m, pockets for landing gear, and are made of 2 mm carbon-fiber and 50mm Al-7075 honeycomb core material. Mass of plates are 1000 kg and 850 kg, respectively.

Fig.[](c) shows the truss system which is made of Aluminum 7075-T6. Total mass of the truss system (4 of the one is in the figure)

is 320 kg. Detailed analysis is given below. Plates' analysis depends on assumptions because of the complex structure of the composites, especially in this case.

Since the DSG will have a robotic arm, it is of our best-interest to make use of it. Therefore, the cargo compartments were designed to be easily removed by the robotic arm. A clip and lock mechanism between the cargo module and the upper plate was employed for this purpose. The compartments were made from carbon-fiber with a thickness of 3 mm and a mass of 260 kg each. There is also a 1-meter long space reserved for batteries.

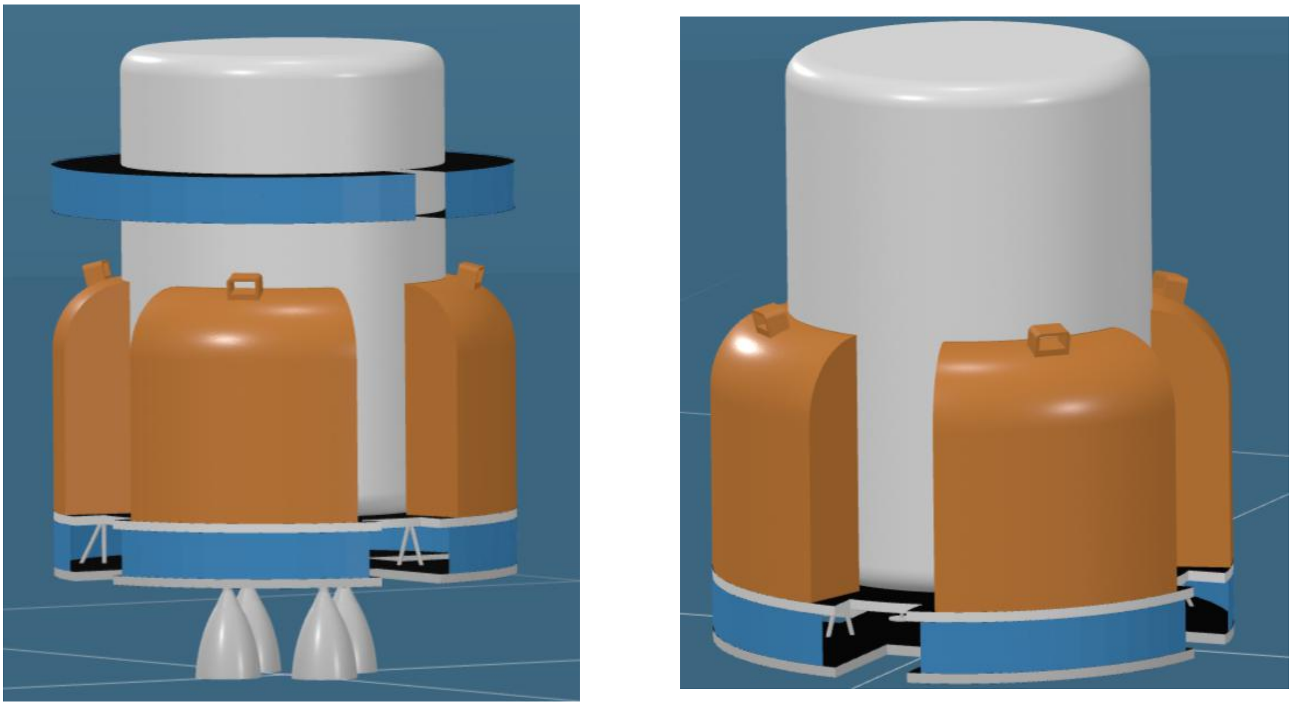
her biri di mi? 3000den nasıl 260a in- dirdin wow



(a) (b)

Figure 5.8: Cargo Compartments

Engines were added and an extra volume was created using 2mm-thick carbon-fiber for the electronics, batteries and flight computers. The extra component has a height of 1m and a mass of 123 kg. Also, the space around the truss system, and the surface of the electronics bay were covered with solar panels.



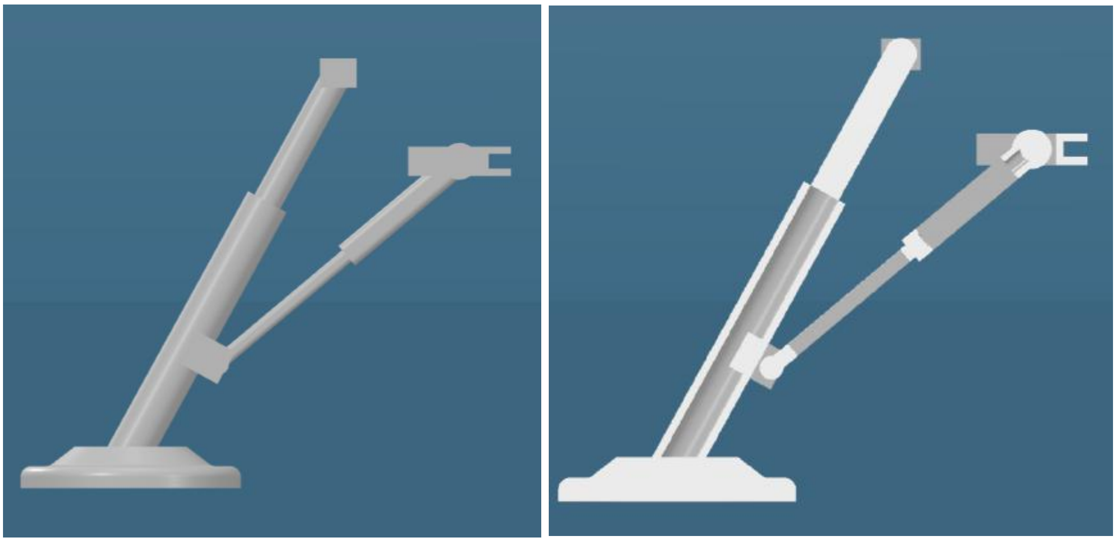
(a)

(b)

Figure 5.9: Structure Overview

There are two possible configurations for the landing gear, static or deployed. The former requires assembly at LEO or DSG, as the diameter of the vehicle exceeds 10m if it is employed. The latter one has a mechanism that deploys the gear during landing.

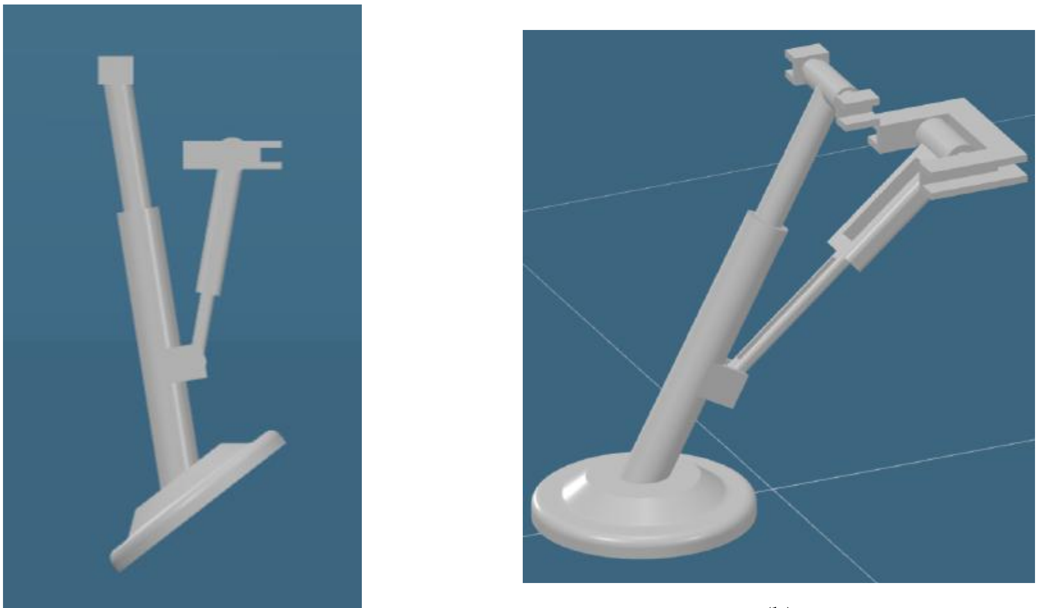
Titanium was used for the landing gear, and the dimension ratios were determined by examining other lander missions like Apollo Lunar Module. The landing system consists of a primary and a secondary strut. A suspension system used in big passenger airplanes was integrated into the primary one for safe landing.



(a) (b)

Figure 5.10: Landing Gear Struts

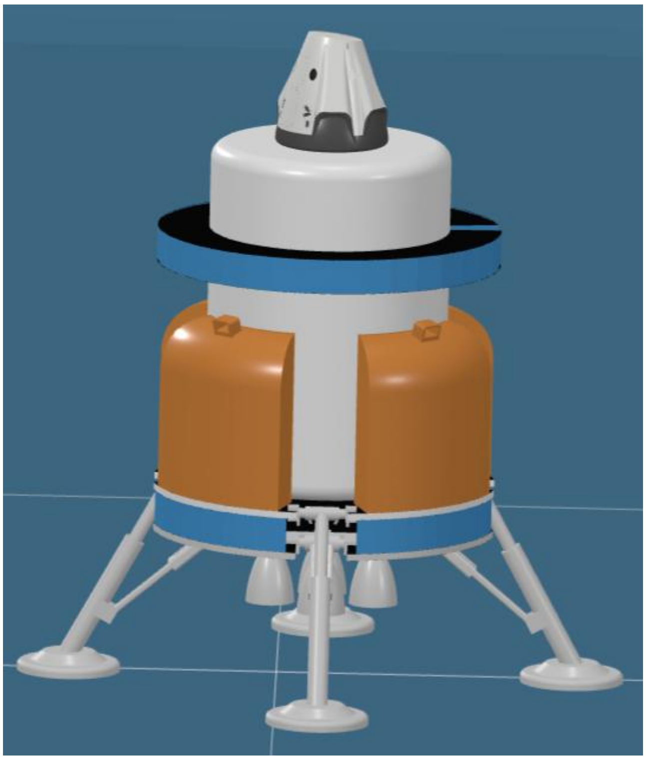
Undeployed and deployed conditions of landing gear assembled to vehicle can be seen in figures. This design allows for the whole vehicle to be launched at once. Landing gears will be deployed after launch via pressurized system without using any power.



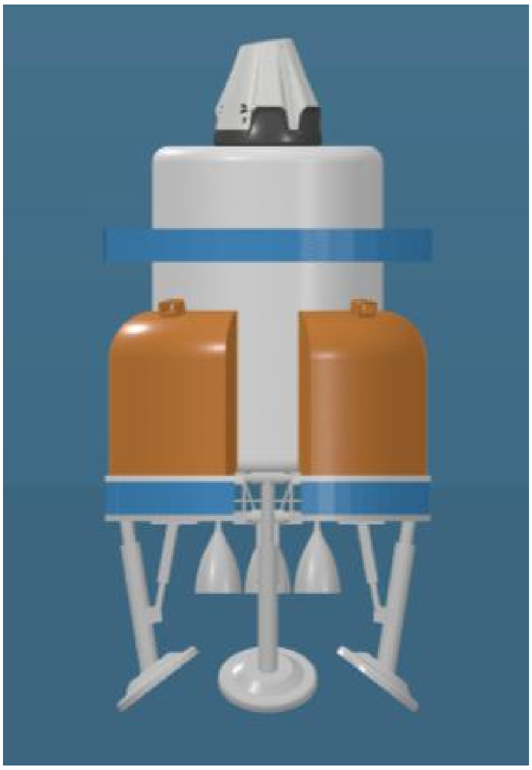
(a) (b)

Figure 5.11: Landing Gear Overview

Next, SpaceX's Dragon crew capsule was integrated on top with a docking system. The resultant height of the vehicle is 17m, and the diameter is 14m with the landing gear deployed. Mass budget of structures can be seen in Table [].



(a)



(b)

Figure 5.12: JELLY CAD Drawings

		Weight (kg)	Subtotal (kg)	Description of Material	TOTAL (kg)
Main Structure	Lower Plate	850	2170	Aluminum Honeycomb Reinforced Carbon Fiber Composite	13463
	Upper Plate	1000		Aluminum Honeycomb Reinforced Carbon Fiber Composite	
	Truss System	320		Titanium	
Electronics	Electronic Plates	123	123	Fiberglass	
Landing Gear	Primary Strut	900	6080	Titanium	
	Secondary Strut_1	260		Titanium	
	Secondary Strut_2	260		Titanium	
	Fixtures	100		Titanium	
Cargo Capsules	Cargo Capsule	260	1040	Carbonfiber	
Ladder	Ladder	150	150	Aluminum	
Tank	Tank	3900	3900	Titanium	

Figure 5.13: Structure Mass Budget

5.5 Thermal and Environmental Control System

5.5.1 Introduction

The thermal control subsystem (TCS) is primarily responsible for keeping other subsystems within the temperature limits they need at all mission phases, for rejecting the thermal energy generated by the crew or other subsystems and for protecting the spacecraft from the solar radiation or heat reflected from other planets.

5.5.2 Thermal Control System Overview

As in other planetary missions, in the lunar mission, once passing the atmosphere, the spacecraft is exposed to three environmental heat sources. The direct sunlight from the Sun, reflected sunlight from other planets (albedo) and infrared (IR) waves. The passive thermal control system will be used to protect the spacecraft from these external heat sources. As the passive thermal control subsystem does not require power to operate, it is generally lighter and budget-friendly. Since it is simpler than active control, thermal design should be as passive as possible. We desire to use a thermal coating with a low solar absorptivity (α) and high emissivity (ϵ). Silverized Teflon, 5 mil tape will be used in the vehicle[18]. Also, to support propulsion and altitude control system equipment and avionics, aluminum alloy skin will be used for thermal dissipation. Lastly, multi-layer insulation (MLI) will be used as a passive thermal control to protect spacecraft from solar and planetary excessive heating. In addition to passive thermal control, as it is a manned mission and some components have critical temperature limits, it is also necessary to use active thermal control.

5.5.3 Thermal Control System Design Requirements

Some temperature limits for our mission design can be seen in Table []. These lunar environmental limits guide us in designing the thermal control mission design.

Mission Phase	*Temperature (Kelvin)*
Lunar Surface (Maximum)	400 K
Lunar Surface (Minimum)	100 K
Lunar Lower Orbit	290 K (avg)
Trans Lunar Coast	65 K (avg)

Table 5.20: Temperature Values for Mission Phases[19]

In addition to these environmental temperature limits, our mission design should also maintain all components within their temperature limits. Component list and their typical temperature ranges are shown in Table []. There are 2 types of temperature limits to take into consideration during thermal design. Operational limits, which are components' thermal limits during operation and survival limits that are necessary thermal limits for the time being. Besides, because of the lack of atmosphere, the spacecraft may be subjected to sudden temperature changes. Thus, during the mission design, gradient requirements were also deliberated[20].

	Components	Operational Range (°C)	Survival Range (°C)
Life Support Subsystem	Crew Quarters	18 – 26	15 – 30
	Food Storage	-20 – 4	-25 – 10
Electrical Subsystem	Batteries	0 – 15	-10 – 25
	Power Box Baseplates	-10 – 50	-20 – 60
	Solar Panels	-150 – 110	-200 – 130
Propulsion Subsystem	Tanks	15 – 40	5 – 50
Attitude Determination and Control	Reaction Wheels	-10 – 40	-20 – 50
	Gyros/IMUs	0 – 40	-10 – 50
	Star Trackers	0 – 30	-10 – 40
Communication	C&DH Box Baseplates	-20 – 60	-40 – 75
	Telemetry & Command units	-10-50	-15-55
Antennas	Antenna Gimbals	-40 – 80	-50 – 90
	Antennas	-100 – 100	-120 – 120
	Onboard computers	-10 – 50	-15 – 55

Table 5.21: Temperature Requirements[20]

5.5.4 Heat Acquisition

Heat acquisition is the process of acquiring excess thermal energy from various heat dissipating components including electronics, avionics, computers, and metabolic loads from the crew members for a manned mission. Cold plates and heat exchangers are the two main components used for heat acquisition in our thermal mission design. Cold plates are used to acquire excess thermal energy from avionics components and maintains these devices within their ideal temperature limits. Heat exchangers, air/liquid heat exchanger means

acquires energy from air loop and transfers it to liquid thermal control loop. Liquid/liquid heat exchanger can also be used. 3 types of liquid/liquid exchanger; liquid cooling garment (LCG) mainly between crew's LCG to thermal control system. Interface heat exchanger which transfers energy from internal pumped fluid loop to external pumped fluid loop. Lastly regenerative heat exchanger, used to maintain the system set point throughout the entire mission[21].

V. Heat Transport

Heat transport is used for movement of energy from one region to another. Pumped fluid loops are main components for heat transport in our thermal control design. The key point of heat transport design is deciding which fluid will be used for fluid loop and how many loops will be used. Single fluid loop architecture is mass and cost efficient, but because of the wide temperature limits that we have in our mission, we desire to use a fluid that has a low freezing temperature. However, majority of this group of fluids are toxic. Therefore, to provide a safe environment for the crew 2 separate fluid loops will be used. Internal non-toxic *propylene glycol and water* working fluid will transfer the thermal energy from cold-plates and heat exchangers. Additionally, external *HFC-245fa* working fluid which is relatively more toxic but has a lower freezing temperature, will transport thermal energy to the radiators.

VI. Heat Rejection

Heat rejection is the final step of thermal control and constitutively it is the part that rejects the excess thermal energy acquired from thermal control system to the space. Radiators and sublimators are two main components of our heat rejection design. Radiators and sublimators work similarly, but the reason we need both in our system is because we cannot use radiators while in LLO due to warmer environmental temperature of this region. Therefore, instead of radiators we need to use sublimators which are sole heat rejection devices. In brief, radiators will be used on Lunar Surface, and when radiators cannot be used or are not enough for rejecting, supplemental Heat Device (SHReD) will be used. To understand the radiator need, we calculate sink temperature from the following equation;

$$T_s = \left[\frac{(\alpha/\varepsilon)(q''_{IR} + q''_{solar})}{\sigma}\right]^{1/4}$$

T_s = Radiative sink temperature (Kelvin)
α = Radiator solar absorptivity (unitless)
ε = Radiator infrared emissivity (unitless)
q''_{solar} = Incident solar flux (W/m^2)
q''_{IR} = Incident infrared flux (W/m^2)
σ = Stefan-Boltzmann constant (W/m^{2}K^4)

(5.1)

According to our coating decisions, $\alpha = 0.08$ and $\epsilon = 0.81$. As a result of these calculations sink temperatures for missions phases are listed in Figure [x].

Trans Lunar Coast (TLC)	• Extremely cold environment • Sink Temperature < 75 K
Low Lunar Orbit (LLO)	• SC shadowed by moon Sink Temperature ~ 60 K • SC between moon and sun Sink Temperature ~ 290 K
Lunar Surface Operation (LSO)	• Sink Temperature varies between ~ 217 K to 239 K

Figure 5.14: Sink Temperatures

Because highest continuous heat rejection requirements will be during Lunar surface operations, we designed our TCS based on it. This means radiators are sized for this mission phase and we only use them for heat rejection during the Lunar Surface Operations (LSO). We will use 4 identical deployable radiators each side (90 degrees apart) on our spacecraft; 2 of them will be horizontal and 2 of them will be vertical to get a more effective design. Moreover, two identical sublimators will be used with the second one serving as backup.

5.5.5 Thermal Balance

Other than external heat sources, we also have 2 main internal heat sources, electric systems and crew itself. According to current standards, each crew member will produce a metabolic heat between 100 W (rest) to 200 W (performing). Since the crew consists of 4 people, maximum human heat production will be 800 W[21]. Also, as calculated before,

will be the maximum power to operate electrical systems[20].

Since thermal balance will be achieved when the total heat received is equal to the total heat radiated. Internal heat received calculated in previously. Then external heats that we will be received can also calculated from;

$$Q_{solar} = J_s \alpha_{eff} A_s \tag{5.2}$$

$$Q_{albedo} = J_a \alpha A_a \tag{5.3}$$

$$Q_{planet} = J_p \epsilon A_p \tag{5.4}$$

Thus from the main thermal balance equation, $Q_{external} + Q_{internal} = Q_{radiated}$, required area of radiator can be calculated from:

$$Q_{radiated} = \epsilon \sigma A T^4 \tag{5.5}$$

- **Worst Hot Case Scenario:**

  During the mission as a worst-case scenario we focused when the spacecraft will land on the sub-solar point of the Moon, which has a temperature of 400 K. In a worst hot case scenario, spacecraft will need the most extreme heat rejection. Thus, it is a significant point for radiator design[18].

- **Worst Cold Case Scenario:**

  For lunar surface operations (LLO) worst cold case scenario will be at far side (night side) at a temperature of 100 K. In worst case scenario, we might need some more thermal energy so that we can cover one side of the radiators with black paints to emit heat.

5.6 Environmental Control & Life Support

5.6.1 Introduction

In order to accommodate the crew and cargo, and to protect them while transporting, a suitable environment must be provided. In addition to this, it is necessary to take precautions against significant changes that might occur in temperature, pressure and radiation. There are different sensors in the spacecraft to detect these changes, which may occur as a result of external influences. The environmental control and life support system (ECLSS) in the spacecraft ensures necessary conditions are met and maintained for both cargo and crew[22].

5.6.2 ECLSS Requirements

ECLSS main requirements are thermal control, atmosphere monitoring, and water management as shown in Figure [].

Trans Lunar Coast (TLC)	• Extremely cold environment • Sink Temperature < 75 K
Low Lunar Orbit (LLO)	• SC shadowed by moon Sink Temperature ~ 60 K • SC between moon and sun Sink Temperature ~ 290 K
Lunar Surface Operation (LSO)	• Sink Temperature varies between ~ 217 K to 239 K

Table 5.22: ECLSS Requirements

To provide these requirements, main systems for ECLSS can be listed as following [23];

- Active Thermal Control System (ATCS)
- Air Revitalization System (ARS)
- Fire Detection and Suppresion (FDS)
- Flight Suit Interface (FSI)

- Potable Water (PWMS)
- Pressure control system (PCS)
- Waste Management (WM)

For lunar lander missions, there are 3 main external effects that need to be considered. Lunar dust, pressure, and radiation and temperature gradients. Radiation and temperature effects and reducing methods were explained in thermal control subsystem section. Pressure differences during the mission can be prevented by using PCS and pressure sensors. Pressurizing and depressurizing, especially before and after extravehicular activities (EVA), are critical for life support. Split operations may be necessary for some missions, and for some bad case scenarios like single suit failures or crew members getting injured. During split operations, to keep crew module in short-sleeve cabin conditions, special features are required. Suit-port concept can be used for this case, which is budget and mass friendly, but less safe than airlock systems. Airlock systems are also designed for split operations, and they help preventing the spacecraft's internal systems from lunar dust. Other options for split operation case include anti contamination coatings, repulsion system for dust particles, and robotic cleaning system for mechanisms [23].

5.7 Command and Data Handling System

Spacecraft's Command and Data Handling System (CDH) is its brain, without it spacecraft would just be a mass in the space. CDH system should be reliable, it should consist of systems that are already proved themselves in critical applications (preferably in spacecraft missions), and they should use industry standards to have most reliability and heritage. Due to our mission and spacecraft design, we need to have radiation hardened (radiation tolerant) computer equipment that should be much less susceptible Single Event Upsets (SEUs), Single Event Latch-up, Single Event Gate Rapture, Single Event Burnout and should be able to recover from SEUs by power resetting while having no downtime (multiple computers). The CDH system should be redundant, it should be less susceptible to failure, it should have failure detection and correction, should have multiple fail safes, and in the event of a catastrophic failure it should have alternative backups to not create a down time. Should be able to check perform self-checks, health checks. It should be operational at all times whether the system is in dock to DSG or not. C&DH's crucial duties are facilitating communications between various subsystems, so having every subsystem talk to each other, ranging from dragon crew modules controls to Thermal control system's request for rotation. It should be able to handle requests coming from Crew module, to DSG to mission control, should be able to process real time data to give autonomous decisions rather to let docking to proceed to open lights. It basically controls every subsystem, and gives decision according to its programming. For all this a computer and then an architecture should be selected. We decided to use an existing solution for our computer needs, since these existing solutions are already tested thoroughly and one of them also has a very long successful heritage. Comparison between RAD750® 6U Compact PCI extended single-board computer, and RAD5545™ SpaceVPX single-board computer.

Table 5.23: Spacecraft Computers [58] [58]

		SpaceWire ports	SpaceWire rate	PCI lane	I2C	JTAG	RAM (MB)	MIPS	Cores	FPGA	Heritage	Power (Watt)
%	**Rad750**	4	132Mb/s	No	Yes	Yes	44	300	1	no	YES	25
	Rad5545	12	320Mb/s	Yes	Yes	Yes	4096	5600	4	yes	NO	35

From the comparison JELLY computer has been selected as RAD5545, even though it has no recorded space heritage, it has been designed and manufactured as a direct replacement to RAD750 computer by the same manufacturer; therefore, there is a good amount of confidence for this computer. It would be safe to assume that this computer will perform as expected after various testing

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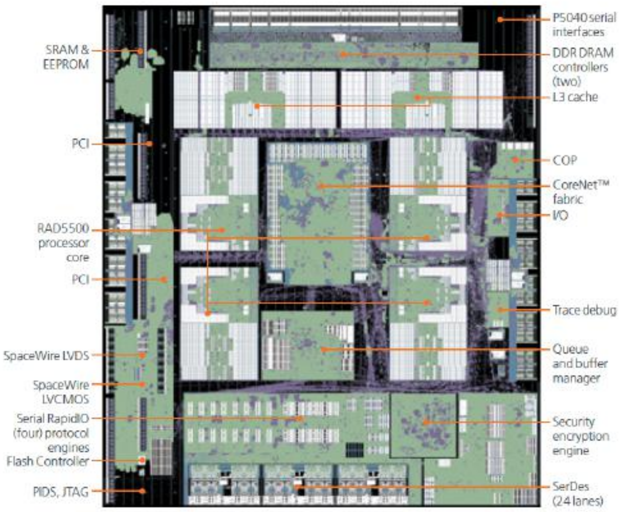

Figure 5.15: Die of RAD5545 [60]

RAD5545 provides robust IO, multicores, high levels of performance, generous amount of ECC system ram, included radiation hardened FPGA, and many more. Schematic is in Figure[]. Having 4 cores, helps with error correcting and supervision of the other cores, this is typically done by using more computer, but now we can run the cores with the same program, to cross check each other to detect errors broken components etc. This will provide very high redundancy without much work.

Having an FPGA will provide good future proofing, and very fast processing for repeatable work, since they allow to use programming to create real logic gates. JELLY will use 4 RAD5545 computers, utilising 2 core of each of the 3 computers' to run an instance of the whole spacecraft main operating system; thus each computer board will have 2 instances of the OS running on 4 core main processor. 4th computer would be used to check the decisions of the other 3 computers, and if 3 of the computers are in agreement, then it would execute the task, but if not then it would hard reset the one in disagreement, and run a diagnostic while executing the result that was most favourable. If in the unlikely event every one of the computers are in disagreement, then previously correct task would be carried while whole computer system would undergo sequenced reset. System would be able to just run on one computer, with reduced reliability. These 2 OS instances running on each of the processors of 3 computers will be cross checking respected computer's coupled OSes to see if a calculation is same on both of them, and if not that computer will be checked for errors, corruptions,

hen hard reset. In the unlikely event that problem still consists then that computer would be declared out of commission until it is replaced or declared otherwise from a manual input. To communicate between computers, subsystems and crew module SpaceWire protocol is utilised, since it is already sported by processors, and it is actively used in space applications. It's robustness and highly fault tolerant design [62], and ease on simple hardware such as sensors [61] makes it ideal for our use case.

Data Handling

Data handling requirements are relatively large compared to space missions, also communication budget of the spacecraft does not allow to every recorded data to be radioed; thus, aim is to record all the data generated locally, and then transfer them when docked to DSG. Doing this requires large and robust storage that is fault resistant and has enough storage for extended periods of time while also being radiation resistant. Due to this considerations, a solid storage method has been selected.

While NASA have been utilising Off-the-shelf parts for data storage, they have been forced to solve uncertainties of these products, by their findings every memory chip exhibits different characteristics under radiation, even from the same wafer. Some of them work better while others fail. [63] In order to reduce this risk and testing, also not being forced to design our own electronic boards, and their various tests, it has been decided to use an existing space grade solution. As a turnkey solution, that meets our needs is Mercury systems TRRUST-Stor ® VPX RT2nd Generation Radiation-Tolerant Large Geometry SLC NAND SpaceDrive [64]. This product is still in the development phase (even though it is not far away from being complete) at 2019, it is to be released at Q4 of 2019 [65], and considering Mercury systems reputation and their other product portfolio, it is very unlikely they would miss their target. So, we have selected RH6940NM2S as our storage device which has 940 GB of storage. Six of these units are used in combination, this will combine to 5640 GB of storage.

Data Calculation

When undocked from DSG, spacecraft cannot send its all data via radio because of the link budget. So, data created during this time is recorded to local storage and then dumped to DSG when docked.

Data sizes are rough estimates since they could change with the contents of the data, and final programming of the vehicle. Monitoring data includes data for the whole JELLY spacecrafts internal temperature, humidity, radiation levels etc. Voice data includes crews' conversations to incoming messages. Engineering data is the data that could be seen useful for health checks to debugging of the spacecraft, including when a door is opened to, at which position the landing arms are at. Image data are the pictures taken by the various cameras on the spacecraft. Video data is the data coming from 10 different cameras on the spacecraft, their position or their numbers are not exact, but their amount should not be a problem for recording needs. Over estimation of these data amounts would be able to provide for data needs of JELLY. This amount of storage should be able to cover 10.6 days of mission (most likely more) with full recording, and it should be able to do more with either with more compression or selective recording.

Liftime

There is no life time given for RAD5545 computer, but its predecessor RAD750 computer has been used in Mars Reconnaissance Orbiter since 2005 and it is still functional [66]. This gives us confidence on our main computer since it is a direct successor to that. There is redundancy on this computer system, and health checks; thus, an end of life failure can be tolerated, and computer can be changed. Our storage type NAND flash have a finite life, with our storage devices we have a lifetime of 38300 missions according to datasheet's whole drive overwrite capacity. [64] Expecting this much missions is non-realistic, since before that we would reach end of life time of other devices. To be cautious CDH system should be monitored constantly, and in case of a failure they should be replaced.

Due to the design of the system, when multiple computers or store devices fail, it is still possible to operate JELLY spacecraft without large restrictions.

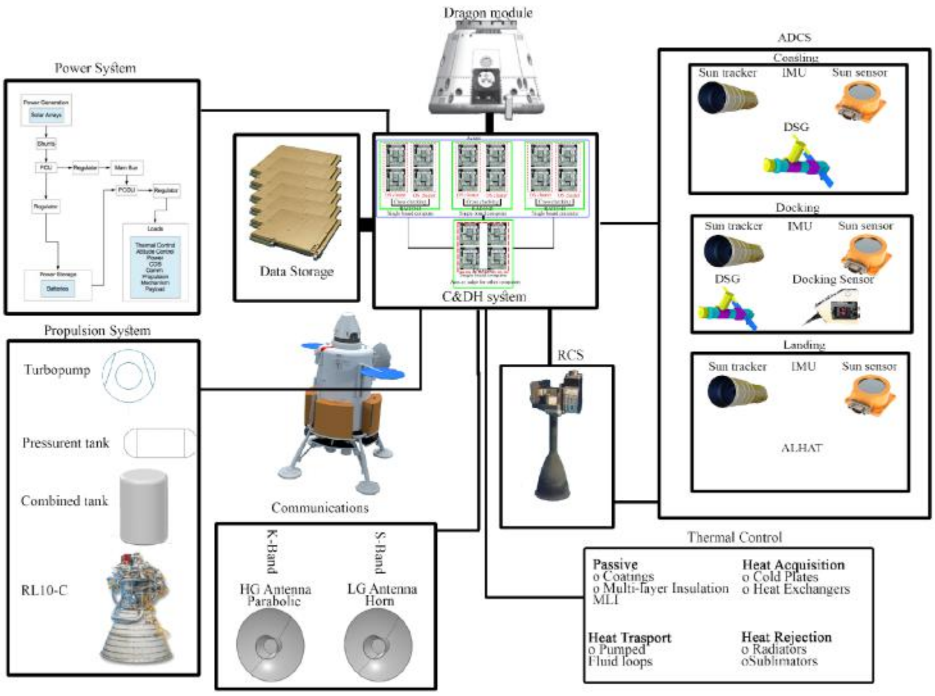

Figure 5.16: Command and Data Handling Structure

5.8 Communication System

Main purpose of the JELLY communication subsystem is to transmit signals to the Earth when travelling from Lunar surface to the LLO, and to both Earth and DSG throughout the rest of the mission. Communication subsystem is critical in terms of mission continuity and safety.

Between the Lunar surface and Lunar region, Jelly transmits and receives monitoring, voice, health status, TTC and navigation data to/from the NASA Deep Space Network (DSN) via Low Gain Conical Horn Antenna (LGA) in S Band frequency domain, which consumes 20 W of energy during this process. These data are then transmitted to NASA Jet Propulsion Laboratory (JPL) from DSN.

Minimum energy consumption is achieved under these conditions. For the worst case scenario, distance between the DSN and JELLY is determined as 400000 km, and the Earth is constantly in the line of sight. After the orbit insertion, communication is provided primarily with the DSG, otherwise with the DSN. The furthest distance between JELLY and the DSG will be 75000 km in the worst case scenario. If data exchange is desired in the second phase, data transmission is performed on Ka Band via HGA antenna which consumes 1000 W of energy. During this transmission, voice, health status, TT&C, and navigation data are exchanged. On the near side of the Moon, the Earth can be seen constantly, and the DSG is in line of sight. On the far side of the moon, only the DSG is in LOI. A back up system is included for reliability.

5.8.1 Ground Station

S Band and Ka Band are recommended by Space Frequency Cooperation Group (SFCG). A specific manual for the Moon communication was published about this subject. [68]. S band uplink frequency is 2100 MHz, downlink frequency is 2200 MHz. Antennas in the DSN support the S Band. These antennas are located in Goldstone, Canberra and Madrid, and each of them gives a full coverage [69]. LGA antennas on JELLY communicate with the ground station. A parabolic antenna with a diameter of 5 m is assumed to be located on the DSG as Ka Band receiver [70].

5.8.2 High Gain Antenna

HGA, which works compatible with Ka band, is used as the primary communication equipment. HGA on JELLY transmits and receives Ka Band signals. It has has a diameter of 3 m and recommended by NASA for Lunar communications [0].Ka Band is recommended by SFCG. High Gain Antenna is more directional. Radiation pattern is higher in smaller angles. HGA is more suitable for targeting a specific point [0]. For the worst case scenario, efficiency of parabolic antennas were assumed as 0.6. The atmospheric attenuation is zero since there is no atmosphere on the Lunar region. Free Space Path Loss (FSPL) is calculated as 162.24 dB. Communication delay between the HGA and DSG is 0.25 seconds. Ka-band uplink frequency is 22.55 GHz, Downlink frequency is 25.5 GHz'dir.

5.8.3 Low Gain Antennas

LGA1 is mounted on HGA and they rotate together. LGA1 is a Conical Horn Antenna which has low power consumption, and a diameter of 1.2. LGA1 is used to communicate in S Band. A second LGA is placed on JELLY for emergency cases as backup antenna. LGA2 will be used to propagate the S band signals with low power consumption. Signals that LGA2 transmits will be received by the DSG and the DSN. LGA efficiency is assumed as 0.511 [0]. FPSL between LGA and DSG is calculated as 155 dB based on the maximum distance which is the DSN ground station. Communication delay is calculated as 1.33 seconds.

5.8.4 Modulation

Bit error rate during communication was determined as 10-5 . Required Eb/No ratio is 9.56 which is determined by the simulation prepared in MATLAB Communication System Toolbox based on Additive White Gaussian Noise (AWGN) channel. AWGN is a simple signal channel for basic calculations. Observing the simulation results, QPSK ¾ is selected as the modulation type as it appears to be the most optimal one considering the data rate for the mission.

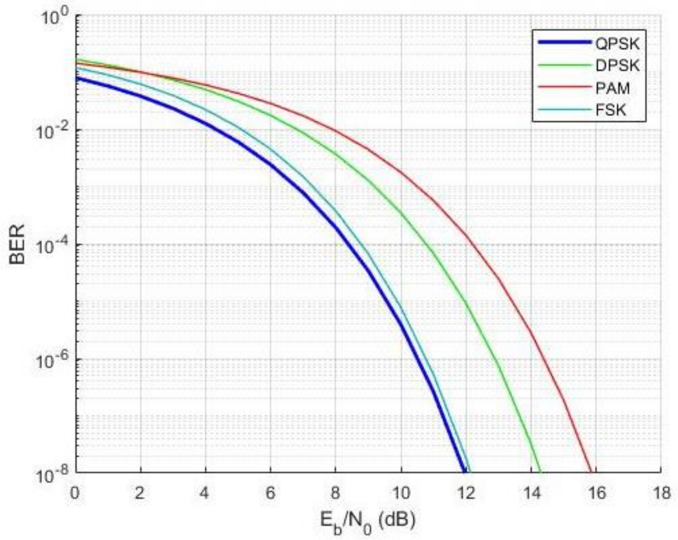

Figure 5.17: Modulation type simulation

5.8.5 Link Budget

Both S Band and Ka Band amplifiers are present in the system. Data transmission rates in Ka and S Bands enable these conditions.

2*	HGA Downlink	HGA Uplink	LGA Downlink	LGA Uplink	Backup Downlink	Unit
	Ka Band	Ka Band	S Band	S Band	S Band	
Frequency	25.55	22.55	2.2	2.1	2.1	GHz
Bit Error Rate	1.0E-05	1.0E-05	1.0E-05	1.0E-05	1.0E-05	
Range	75000	75000	400000	400000	400000	Km
Symbol Rate	6660	1666	2000	100	2000	Ksps
Transmitter Power	100	100	20	20	20	Watt
Transmitter Power	20	20	13.13	13.13	13.13	dBW
Transmitter Cable Loss	-0.5	-0.5	-0.5	-0.5	-0.5	dB
Transmitting Antenna Gain	55.490	58.842	25.890	54.874	25.486	dB
EIRP	74.990	78.342	38.520	67.504	38.116	dB
Space Loss	-162.249	-161.164	-155.490	-155.086	-155.086	dB
Atmospheric Attenuation	0	0	-0.5	-0.5	-0.5	dB
Polarization Loss	-0.15	-0.15	-0.15	-0.15	-0.15	dB
Receiver Antenna Gain	59.927	54.405	38.628	33.787	54.874	dB
Pointing Loss	-3.601	-2.805	-0.002	-1.311	-0.002	dB
Receiver Cable Loss	-0.06	-0.06	-0.06	-0.06	-0.06	dB
Total Received Power	-31.142	-31.431	-79.054	-55.816	-62.808	dB
Receiver System Noiser Temperature	200	50	200	50	200	K
System Noise Density	-205.59	-211.61	-205.59	-211.61	-205.59	dB-Hz
Carrier Power to Total Power Ratio	-15.207	-15.207	-15.207	-15.207	-15.207	dB
Received Carrier Power	-46.349	-46.638	-94.261	-71.023	-78.014	dB
Carrier Link Margin	159.241	164.972	111.329	140.588	127.575	dB
Data Power/Total Power	-9.319	-9.319	-9.319	-9.319	-9.319	dB
Data Power Received	-40.461	-40.750	-88.373	-65.135	-72.127	dB
Data Symbol Rate	-98.235	-92.217	-93.010	-80.000	-93.010	dB-Hz
Eb / No achieved	66.894	78.643	24.206	66.475	40.453	dB
Eb / No required	9.56	9.56	9.56	9.56	9.56	dB
Data Link Margin	57.334	69.083	14.646	56.915	30.893	dB
Wavelength	0.012	0.013	0.136	0.143	0.143	Meter
Beamwidth	0.183	0.207	8.182	0.303	8.571	Degree
Transmitter Antenna Type	Parabolic Dish	Parabolic Dish	Conical Horn	Parabolic Dish	Conical Horn	
Transmitter Diameter	3	5	1.2	34	1.2	Meter
Transmitter Antenna Efficiency	0.600	0.600	0.511	0.600	0.511	
Receiver Antenna Type	Parabolic Dish	Parabolic Dish	Parabolic Dish	Parabolic Dish	Parabolic Dish	
Receiver Diameter	5	3	5	3	34	Meter
Modulation Type	QPSK	QPSK	QPSK	QPSK	QPSK	
Data Rate Per User	10000	25000	3000	150	3000	Kbps
Data Type	Monitoring, Voice, TT&C, Eng. Data, Image and Video	Monitoring, Voice, TT&C, Eng. Data, Image and Video	Voice,Health Status,TT&C and Navigation	Voice, Health Status, TT&C and Navigation	Voice, Health Status, TT&C and Navigation	
Transmitter - Receiver Locations	Lunar Surface - Lunar Orbit	Lunar Orbit - Lunar Surface	Lunar Surface - Earth or Lunar Surface - Lunar Orbit	Earth - Lunar Surface or Lunar Orbit - Lunar Surface	Lunar Surface - Lunar Orbit or Lunar Surface - Earth	

Table 5.24: Link Budget

6. Mission Analysis

6.1 Mass Budget

There was no total mass constrain on the spacecraft itself, as the mission allows for multiple launches and assembly on the DSG. However, there was a constraint on the vehicle size. SLS Block 2B will have a fairing diameter of 10m, and a large mass means a large diameter. In Table 6.1, mass per equipment and subsystem is given along with a margin that allows an increase in the mass during the production stage. The table also includes the total dry and wet masses for crew and cargo missions. First estimations were made from Brown's textbook [57] using PDR Class-1 margins and estimating from old data. SLS Block 2 will be able to carry payloads larger than 45 tons to TLI, thus it was feasible to keep the launch mass around that number. Launch propellant includes the necessary amount of propellant for NRI and catch-up burn maneuvers. When the vehicle is fully loaded, propellant mass is about 288% of dry mass.

6.2 Schedule and Lifetime

Schedule for the project was estimated using NASA project lifecycle [54].

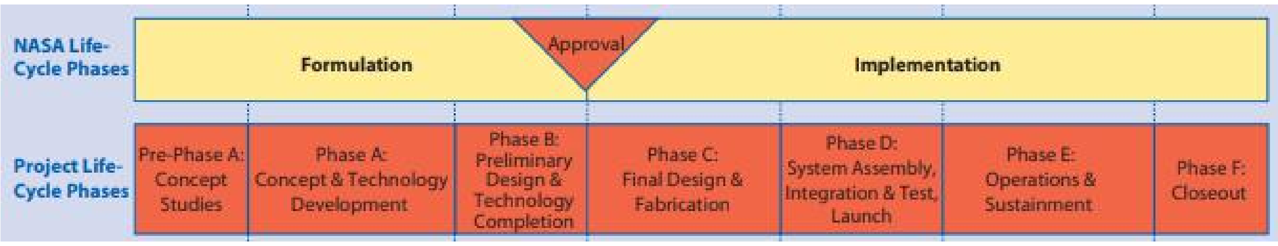

Figure 6.1: NASA Project Lifecycle from the Systems Engineering Handbook

As RFP states (P1), JELLY is able to go under refurbishment from the DSG. Most of the components of the vehicle are replaceable, therefore it will keep operating until it takes an external damage, the DSG completes its operations, or a better one is produced. The lifecycle limitations of components are discussed in their respective sections. The SLS will be ready around 2028, which gives us around 8 years to complete the design. Advanced studies for critical missions such as this one may extend for several years [54]. Thus, the progress should be steady and careful by assessing every step for this project which costs billions of dollars and involves the safety of humans. Hence, Phase A will begin

Subsystem	Equipment	Mass (kg)	Total Mass (kg)
GN&C	32x RCT	64	2083.35
	4x Star Tracker CT-2020	12	
	ALHAT	15	
	RVS 3000 Sensor	14	
	New Space 411 Sun Sensor	0.15	
	Honeywell MIMU	28.2	
	Loaded Propellant	1950	
OBDH	4x RAD5545	80	720
	6x SSD	140	
	10x Camera	50	
	Cabling	450	
Propulsion	Tanks and Feed System	14020	14784
	Engines	764	
Power	Solar Cells	37	4419
	Solar Panel Support	463	
	Batteries	3051	
	Power Distribution and Control Units	868	
Structure	Plates and Trusses	2170	11884
	Landing Gear	6080	
	4x Cargo Compartment	1280	
	4x Conveyor	880	
	Tank Case	850	
	Ladder	150	
	Lift	150	
	Dragon Port	324	
Thermal	Heat Exchangers	18.71	262.5
	Cold Plates	5.175	
	Radiators	62.1	
	Sublimator	26.19	
	Multi-layer Insulation	107.64	
	Pumps and Accumulators	16.56	
	Plumping and Valves	15,39	
	Instruments and Controls	5.13	
	Fluids	5.13	
Communication	3m Dish Antenna	350	436
	1.2m Horn Antenna	3	
	Ka-Band Transmitter	2.5	
	S-Band Transmitter	0.12	
	Cabling	80	
Margin		6369	18.40%
Total Bus Mass			40,958
Crew Mode Payload	Dragon 2	4200	7484
	Airlock	1300	
	ECLSS	595	
	Crew Supplies	1391	
Loaded Propellant			147,039
Total On-Orbit Dry Mass			49,067
Total On-Orbit Wet Mass			196,106
Cargo Mode Payload	Various components	15000	15000
Loaded Propellant			161399

by the end of 2021, providing the team with enough time to evaluate the decisions made in Pre-Phase A. During phases A and B, technology development progress will be tracked, and the production and testing phases will be planned. Phase C will be the beginning of the subsystem level design with an increase in the staff, followed by another increase for Phase D where the systems will go under integration and testing procedures. The time interval until the final stage are held long to prevent any scheduling errors.

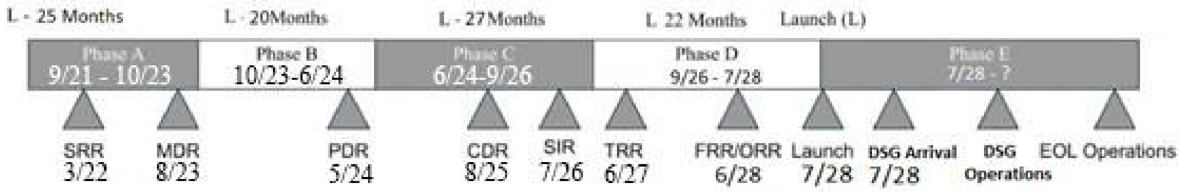

Figure 6.2: JELLY Project Lifecycle. This Figure Describes the Schedule to follow for the JELLY mission

7. Cost and Risk Analysis

7.1 Cost Estimation and Methodology

The mission has a large NASA budget of $10 billion. The single most expensive element is the launch, though each element has a considerable contribution as it can be observed from Table 7.1 which contains the detailed cost breakdown. Cost estimation process was done using parametric estimation method from the textbook Space Mission Analysis and Design [11] and verified using old Lunar and Martian missions. Figure 7.1 gives the WBS that captures each cost element.

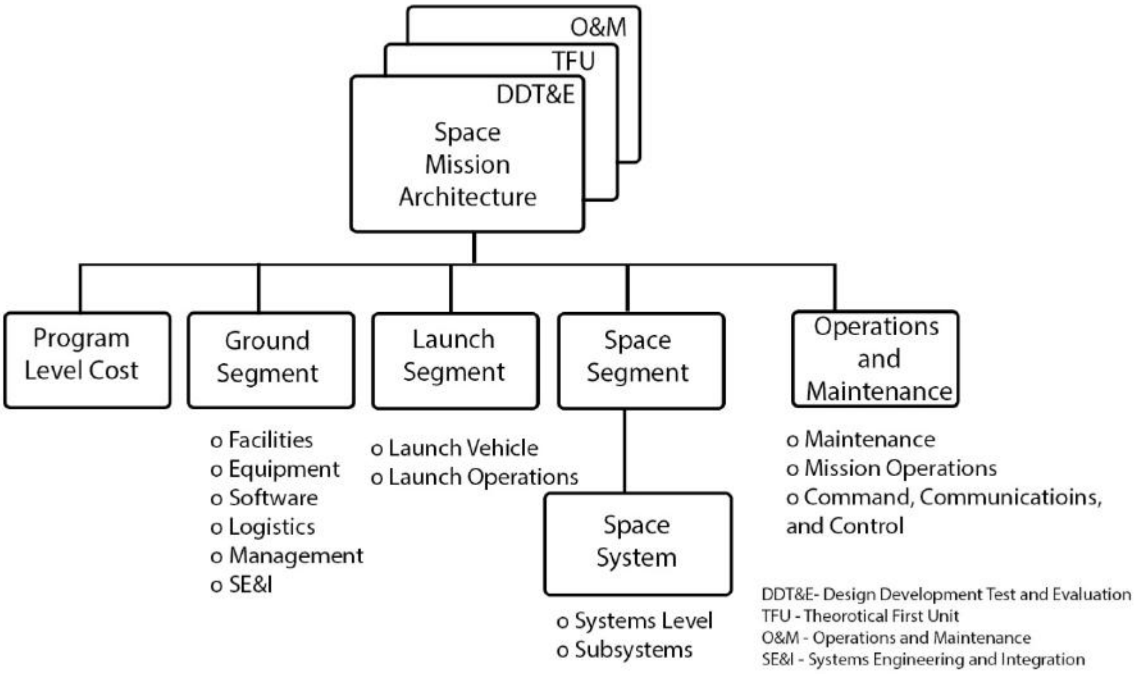

Figure 7.1: Work Breakdown Structure of Cost Elements

Estimates show that the total cost for the mission including launch, DDT&E, and TFU cost sum up to $8,992 billion. This estimation results in a margin of $1 billion, or 10% of the total budget which can be used to cover for any errors and non-predicted costs, to aid the TRL developments of the technologies, or for the back-up launch strategy.

Element	DD&E Cost ($ millions)	TFU Cost ($ millions)	Total Cost ($)	Mission Budget (%)
Launch Segment			$1,721.5	17.2
SLS [[56]]	-	$1,500	$1,500	15
Launch Operations	-	$221.5	$221.5	2.2
Spacecraft			$5,186	52
Dragon 2 [[55]]	-	$130	$130	1.3
Structure	$364	$132.3	$496.3	5
Thermal	$33	$7	$40	0.4
EPS	$278	$250	$528	5.3
Communication	$140	$100	$240	2.4
C&DH	$197	$121	$318	3.18
GN&C	$146.5	$101	$248	2.48
Propulsion	$120	$85	$205	2.1
Integration, Assembly & Test (IA&T)	-	$470	$470	4.7
Program Level	$890.3	$1,456	$2347	23.5
Ground Support Equipment	$164	-	$164	1.6
Ground Segment			$1,000	10
Facilities (FAC)			$73.4	0.7
Equipment (EQ)			$200	2
Ground and Flight Software (SW)			$408	4.1
Logistics			$61.5	0.6
Management			$73.4	0.7
Systems Engineering			$122.5	1.2
Product Assurance			$61.5	0.6
Earth Terminals, Antennas, and Communication Electronics			$0.1	0.001
Operations and Support			$1,085	11
Maintenance			$545	5.45
Contract Labor			$320	3.2
Government Labor			$220	2.2

Table 7.1: Cost Breakdown by Element

7.2 Risk Management

Risk assessment procedure begins with identifying the possible risks, and their respective impacts on the mission. Table 7.2 lists the risk levels and Table 7.3 lists the possible risks for the mission.

Level	Type
1	Minor
2	Moderate
3	Critical
4	Catastrophic

Table 7.2: Risk Definitions. In this table, risk types and threat levels are shown

Case No	System	Risk Event Table	Risk Level	Solution
1	Thermal	Coating Failure	3	Operation is affected. Direction must be directed towards the region where environmental conditions are appropriate again.
2	Structural	Landing Gear Failure	2	Operation is not affected, RCTs will be used to stabilize the spacecraft and JELLY will return to DSG
3	Structural	Dragon Hatch Stuck	1	Crew will be supplied by Dragon until Canadarm opens the hatch
4	Structural	Docking Failure	3	Docking system has high reliability and it will go under lots of test procedures
5	Guidance	RCS Failure	4	RCS has space heritage and is reliable.
6	Guidance	Active/Passive sensors failure	1	Spare systems in the vehicle can continue the operation
7	Guidance	ALHAT system faİlure	3	Abort the mission and return to DSG to await refurbishment.
8	Communication	Antenna Failure	2	Redundant antenna will be used. A new path must be installed to transmit signals to Deep Space Gateway/Earth.
9	Propulsion	Thruster Failure	2	The opposite one is shut down for stabilization, weight is minimized by removing payload and cargo components to decrease required thrust.
10	Propulsion	Exploding	4	Complete ground testing of tank and fitting system. Engines have over 99% reliability.
11	Power	Solar array failure	1	The energy consumption of the systems in the vehicle goes into emergency mode and consumption is minimized. Batteries can sustain the vehicle for 27 hours
12	Orbital	Soft landing failure	3	The systems in the vehicle are repaired in DSG and reusable again.

Table 7.3: Risk Events, Threat Level, and Solutions. This table lists the possible risks and solutions

Probability Level	Case
0-5%	4,5,7,10
5-15%	1,12
15-30%	2,8,9,11
30-35%	3,6
>35%	none

Table 7.4: Event Occurrence Possibilities

An analysis was done combining the event occurrence possibility and threat level of the risk elements, and the results indicate that the mission can proceed. However, it is highly dependent on SLS Block 2 as no other launch vehicle has adequate fairing diameter. The NASA Transition Authorization Act signed by President Trump on March 2017 gives assurance for the progress made by NASA to complete SLS by the given date, however this is not a full guarantee as previous examples has shown. Thus, as a back-up strategy, the launch will be performed in two steps. SLS Block 1 will first launch the main components of JELLY. Then with a subsequent launch, the cargo compartments and the disassembled lower plate of the spacecraft will be sent to the DSG where the whole system will be assembled with the aid of Canadarm. The $1 billion margin on the budget allows this strategy.

8. Summary

On the anniversary of the Apollo 11, JELLY will be launched from the Earth with SLS Block 2. SLS Block 2 upper stage will perform the TLI. After 4 days of TOF, LOX/LH2 bipropellant system will perform NRI for insertion to orbit of the DSG. Once this phase is completed, main phase will begin, which is the round trip operations between the DSG and the Lunar surface. After leaving the DSG, Jelly will first go 100-km LLO, then begin the descent phase. After spending 24 hours on the Lunar surface, it will begin the ascent phase. It will reach 15x100 km elliptical orbit, then 100 km LLO. Finally, JELLY will perform its final maneuver to reach NRHO. JELLY has the ability to transport cargo or crew up to 4 people and able to land on any point on the Lunar surface. During the mission, design of JELLY provides a healthy and efficient working environment for the cabin crew while protecting both the cargo and cabin crew from external influences. 4 RL10C-1 engines will provide the required thrust to the spacecraft for the descent and ascent phases. The spacecraft consumes 161 metric tons of propellant in order to provide 5750 m/s of velocity change. JELLY is covered with several coatings and multi-layer insulation to be protected from radiation, lunar dust and temperature. Besides that, active thermal control system will conserve all other equipment's required temperature ranges during the all mission phases. Four identical radiators on all four sides of the vehicle work during the rejection of the residual thermal energy. JELLY uses sun sensors, star trackers, IMU, docking and landing sensors for attitude determination. 4 packs of 8 RD-40, 32 RCT in total is used for attitude control. High Gain Parabolic Antenna and Low Gain Horn Antenna transmits and receives signals from/to the spacecraft, S Band and Ka Band frequencies are used. JELLY's CDH system is design with existing (or soon to be released) parts to be robust, hardened, and failure tolerant to provide nonstop operation at all conditions, utilizing multiple RAD5545 radiation hardened single board computers, and data storage units. EPS was designed with safety in mind to provide uninterrupted power to JELLY at all conditions, obtaining its power from mainly ultraflex solar array system with Metamorphic Fourth Generation CPV solar cells to achieve most efficiency, while using Saft VL51ES Battery Cell to have maximum specific energy and life time for energy storage. Main bus voltage was selected as 120V DC for its specific benefits.

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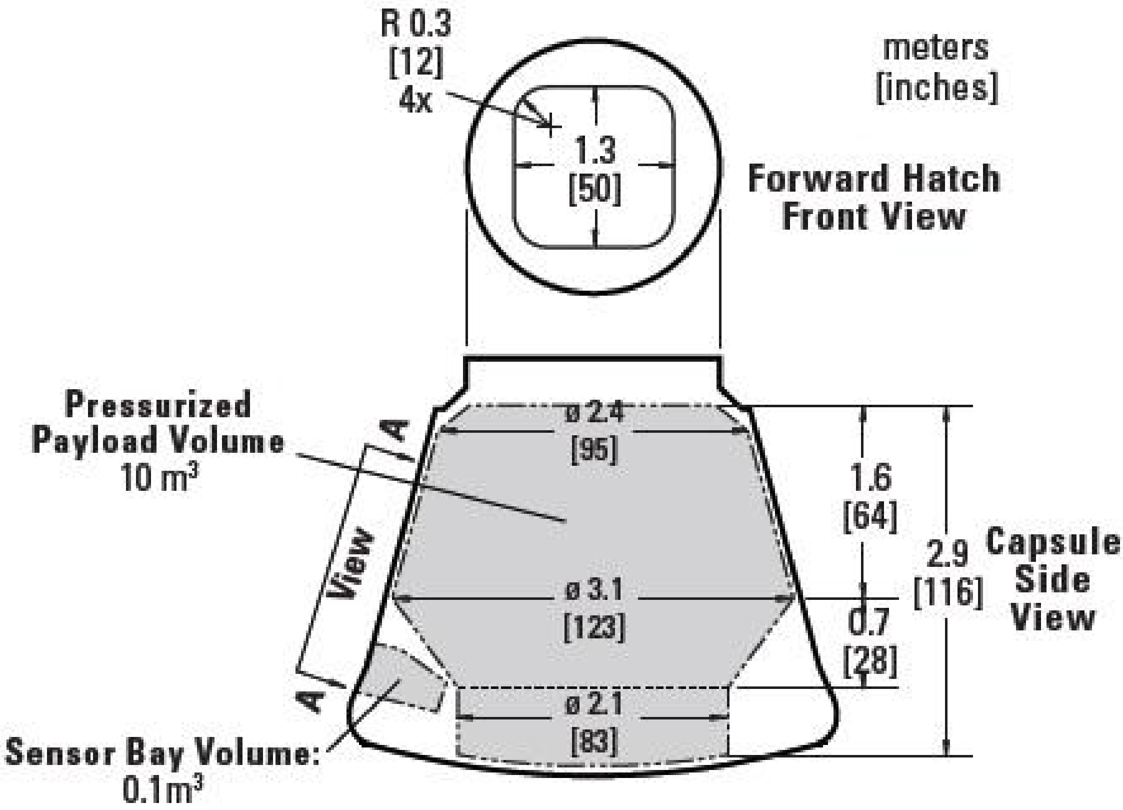
Appendices

Appendix A - Dragon Crew Capsule Schematic[13]



Appendix B - JELLY CAD Models

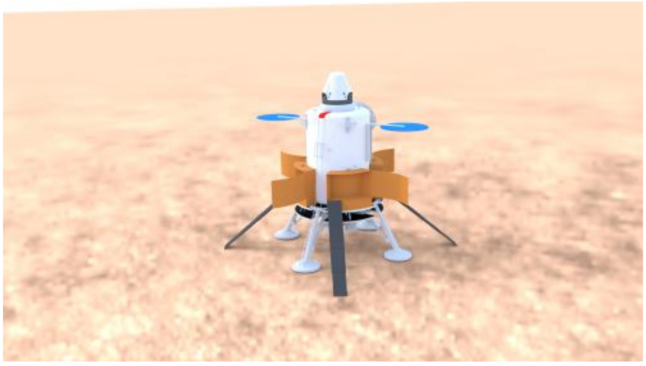


Figure 8.1: Landing Configuration

Figure 8.2: Flight Mode