MMAE 412: PoleSat Design Final Report

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Member Name	Contribution
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CubeSat Mission Parameters							
Mission Name: **PoleSat**	Mass: **20.18 Kg**	Cube Size: **12U**		Desired Orbit:	Acceptable Range:	400 km @ 51.6 deg incl: **NO**	Desired Mission Life: **3 years**
			Altitude	**717 (km)**	**715 - 719 (km)**		
			Inclination	**92°**	**91° - 93°**		

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1 Abstract

This report contains the planning, organization, design process, analysis, and final design of a CubeSat proposal in response to the CubeSat launch initiative. The objective of the PoleSat mission is to observe and monitor ice elevation at the North and South Poles. The team was inspired to design the PoleSat based on the fact that climate change is very relevant in everyday life as shifting climates is affecting not only the weather but lifestyle and agriculture around the world. With reports in the news about how climate change is threatening human life and must be addressed, the team felt inspired to design a CubeSat which would be meaningful to NASA scientists.

This objective will be accomplished by using Synthetic Aperture Radar [19] which is a form of radar that is able to reconstruct 3-dimensional landscapes by sending radio waves and illuminating a target area, depending on the echo of these waves, data is calculated to reconstruct the landscape. Using SAR is critical to the mission as having an understanding of ice elevation at the poles will fulfill the mission requirement of observing and monitoring the ice at the poles. Using the SAR to calculate ice elevation, trends will be seen over time as how the ice elevation has changed or shifted in total or during varying months. In combination to the SAR technologies, a unique layout will be to have the solar panels fold out in order to maximize surface area as the PoleSat must have a specific orientation to ensure the SAR to be pointing towards Earth.

The expectation of the mission lifespan of the PoleSat is 3 years, as that is how long the batteries were estimated to last. The PoleSat has a relatively low level of autonomy due to a limitation on weight which will restrict onboard functions of data manipulation. Data will be transmitted to scientists who will use the data observed and produce the elevation of the ice at the poles.

The target audience of the PoleSat is NASA and Earth Scientists who would use this data to have a deeper understanding of the ice elevation at the poles, and understand the rate of climate change occurring in the world in order to take certain actions to prevent it from further worsening.

2 Relevance to NASA

The PoleSat proposal is in response to an important problem which NASA should be interested: the monitoring of ice at the Earth's poles. The PoleSat outlined in the mission concept design uses Synthetic Aperture Radar (SAR) to observe and collect data of ice at the poles. The SAR sends successive pulses of radio waves in order to "illuminate" a target area, where the echo of the pulse is recorded and processed to map the location. The motivation behind the PoleSat to analyze and monitor ice at the poles is due to the desire to help understand the Earth. Climate change is an effect which is being accelerated by humans, and the loss of ice at the poles is subsequently one aspect of climate change. The reduction in ice will inherently rise the levels of the oceans which will subsequentially cause the loss of land along the coastal regions. As such, the team felt that it is very important to help provide NASA scientists the data to monitor and analyze the reduction of ice at the poles. The PoleSat mission was chosen to be an orbital flight mission in comparison to other mission techniques such as balloon flight, or drones, etc which would struggle and be much more limited in the endeavor of measuring ice. Having an orbital mission is much simpler and allows for continuous tracking of the ice elevation.

The PoleSat proposed appeals to NASA's strategic goals in numerous ways. For Strategic Goal 1: Expand Human Knowledge Through New Scientific Discoveries sub-section Safeguarding and Improving Life on Earth [14]. Within this subsection, it is stated that:

"NASA's Earth science data helps to advance U.S. National interests in agriculture by providing food security for the Nation, economic growth, products to trade internationally, and jobs here at home. Through our partnerships with other agencies that maintain forecast and decision support systems, such as the National Oceanic and Atmospheric Administration (NOAA), United States Geological Survey, and Environmental Protection Agency, NASA improves National capabilities to predict climate, weather, and natural hazards, to manage resources, and to develop environmental policy."

Looking at this statement from NASA's strategic plan, the PoleSat directly appeals to NASA's ambitions as monitoring and observing ice elevation year-round will help Earth scientists to understand climate change more, which would then provide useful information to the agriculture industry as well as other industries. As it stands, 40% of the world's population is currently estimated to live within 100km of the coast [17]. The melting of the poles will increase the ocean level, which will then reduce land size on a global scale. Due to the melting at the poles, up to 40% of the world population will be directly affected by the rise in sea level and submergence of coastal regions, which will indirectly affect the rest of the world. Economically speaking, this would be a disaster as it will affect infrastructure directly in those areas, as well as the areas the people migrate to. Mitigating this environmental catastrophe is critical to NASA and the rest of the world, as resources will subsequently become scarcer if the poles melt. Thus, monitoring of the ice at the poles is a task that must be closely monitored.

Currently, there is a satellite being used to monitor ice at the poles of the earth: ICESat-2 [11], which was launched in 2018 and has a planned mission life of 3 years. As the end date of the ICESat-2 [11] is near, NASA will need some other mechanism to continue to monitor the poles for continued ice melt. This is why the PoleSat is key, as it will continue providing critical information to NASA in a different and dynamic way. The ICESat-2 [11] used an ATLAS laser to monitor ice elevation whereas the PoleSat aims to use SAR to monitor ice. The determination behind choosing SAR is due to the weight limitations on the mission, as the PoleSat is a CubeSat mission. The PoleSat mission is also conscious of end-of-life conditions as well, which appeals to sustainability. The PoleSat's end-of-life operation is to have a final burst of thrust to decrease altitude of the PoleSat, resulting in the PoleSat being burnt up in the upper atmosphere of Earth.

3 Feasibility

3.1 Mission Statement, Requirements, Constraints and Mission Success Criteria

Mission Statement: The mission of the PoleSat is to accurately measure and track the ice elevation on Earth at the North Pole and South Pole. The PoleSat will use Synthetic Aperture Radar [19] techniques to collect data which will be sent to NASA Earth scientists to have a better understanding of the change in ice elevation.

Requirements: The requirements for the PoleSat is that the mission lifespan should be no less than 36 months from the day the PoleSat is launched. The PoleSat's orbit should cover the North and South Pole. The ground resolution of the SAR should be 2 - 75 meters.

Constraints: The top level constraint to the CubeSat comes from the size for the PoleSat (i.e. 12 U). The max mass of the CubeSat is limited to 20 ± 2 kg, the max cuboid size is limited to $0.2m \times 0.2m \times 0.3m$. The launch site Kennedy Space Center $(28.5729N, 80.6490W)$ is also considered as constraint due to the chosen target orbit.

Mission Success Criteria: To measure the success of the PoleSat will be based on the ability to accurately measure the polar regions with the SAR while keeping proper orbit. Success also comes from being able to transmit data within a day to the ground station so the NASA scientist can collect data in a timely fashion. The PoleSat will also be evaluated by the ability to accommodate the SAR and to ensure that the CubeSat is able to operate for 3 years.

3.2 Review of the Design at a System Level

3.2.1 Subsystems Required

The subsystems that will be taken into account to ensure the PoleSat is able to accomplish the desired mission include: Structure & Configuration, Electrical Power, Data Handling, Communications, Attitude Determination & Control, Orbit determination & Control and Thermal Control. These distinct subsystems were used in an iterative process as the results for one subsystem directly affected the design of the other subsystems and the rationale behind why certain decisions for subsystems were made. The name of each subsystem is self explanatory and these subsystems are critical to the success of the PoleSat.

3.3 Performance Characteristics for the System and Subsystems

The performance characteristics for the PoleSat are the ability to measure ice elevation at the North and South Poles. The way to measure the accuracy of this elevation measurement would be to manually go and measure or measure against legacy data of a certain spot and check if the elevations are the same then confidence can be given to the instrumentation used. However, if the measurements are not the same then it would be necessary to compute an offset by measuring several areas. The subsystems can be evaluated by their ability to operate the spacecraft which is done by using Orbit Determination & Control subsystem to maintain the proper orbit. The metrics that will be used to determine if the Polesat is performing well or not is that, "is the PoleSat able to full fill different mission requirements and are the subsystems able to perform their specific functions when needed?" In order to accurately gage the performance of the PoleSat, the PoleSat must satisfy these conditions.

3.4 Elements of the Mission Architecture

Various elements were used in order to come up with a solution to the PoleSat that is satisfactory both logically and numerically. The following sections show the values found and used in the design calculations, which ensure confidence in the PoleSat mission.

3.4.1 Preliminary Payload

The SAR payload [19] has the following instrument information, which will be critical in the mission.

L Band operating altitude	717 km
Frequency	1.25 Ghz
Range Coverage	121 km - 538 km
Incidence Angle	9.6°-36.9°
Resolution	2 - 75 m
Antenna	0.925 m * 1.37 m

Table 3.1: SAR Data from legacy mission [19]

3.4.2 Orbits

Inspiration for classical orbital elements were originally drawn from the ICESat-2 [11] mission. FreeFlyer Software [9] was used to calculate a more specific orbit that exactly fits the scope of the mission.

Height	717 km
Semi-Major Axis	6,859 km
Eccentricity	0.0002684
Inclination	92°
Argument of Perigee	0°
RAAN	90°
True Anomaly	0°
Orbital Period	94.22 minutes

Table 3.2: Classical Orbital Elements

3.4.3 Launch Vehicle Assessment

A table was created of several launch vehicles that may be considered for the mission and they were compared based on the payload mass and total launch cost. The results showed that the best launch vehicle is the Falcon 9 rocket as it has the lowest, Cost to Payload mass ratio, as well as it is able to allow the PoleSat to get into LEO.

Launch Vehicle	Cost $/kg	Orbit Altitude (km)	Total cost ($)	Payload LEO (kg)
Electron	26,666	500 SSO	6 mln	150 - 225 SSO
SPARK	N/A	400 SSO	N/A	250 SSO
Minotaur 1	69,000	500	40 mln	580
Minotaur C	34,200	500	50 mln	1,458
Delta 2	8,500	LEO	137 mln	2,700 - 6,100
Antares	12,308	LEO	80 - 85 mln	8,000
Falcon 9	**2,684**	**LEO**	**62 mln**	**22**
Atlas 5	5,782 - 8,117	LEO	109 - 53 mln	8.250 - 20,520

Table 3.3: Table of Launch Vehicle Data

3.4.4 Size, Mass and Power Budgets

The size of the PoleSat was chosen to be 12U [equivalent to 20 kg] due to the mass constraint placed on the mission. The payload itself consumes 8 kg of the total CubeSat's weight, therefore the team felt the need to have the maximum mass allowed to consider other subsystem's weight.

The mass of the PoleSat was determined based on a top-down approach, the given mass of the SAR allowed the team to work around the budget of a 12U CubeSat. The final mass determined based on the design calculations follows the constraints for the CubeSat size and ensures confidence in the design choices made.

Subsystems	Mass Value (kg)
Payload	8
Structure	5.78
Thermal	0.4
Power	1.56
TT&C	1.5
ADCS	2.04
Propulsion	0.9
Total	20.18

Table 3.4: Table of Mass Budget Breakdown

The power budget was first determined by calculating the power required for each subsystem by using an iterative process. Working through a process to determine the power required allowed the team to determine the size of the solar panel needed. A 30% power budget was included to ensure safety for the PoleSat and to account for power needed in future at the manufacturing stage.

Subsystems	Power Available (W)
Payload	50
C&DH	45
ADCS	9.36
Total	104.3
Average Power with 30% margin	135.6

Table 3.5: Table of Power Budget Breakdown

3.4.5 Dimensional Drawings

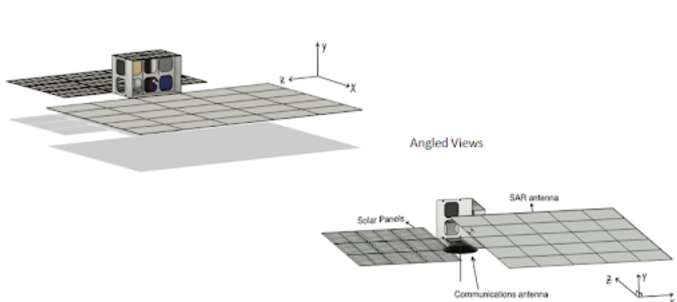

Figure 3.1: PoleSat's Deployed Angled View

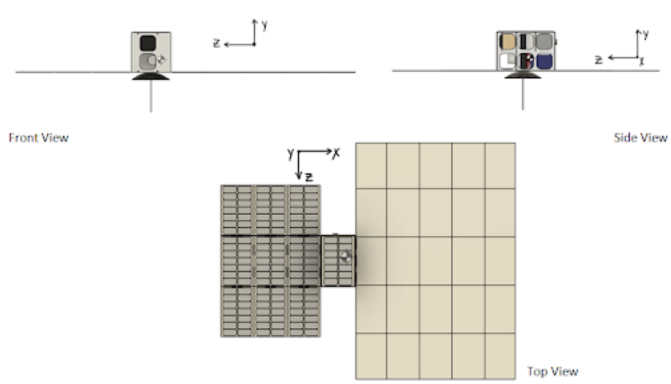

Figure 3.2: Front, Top and Side View of PoleSat

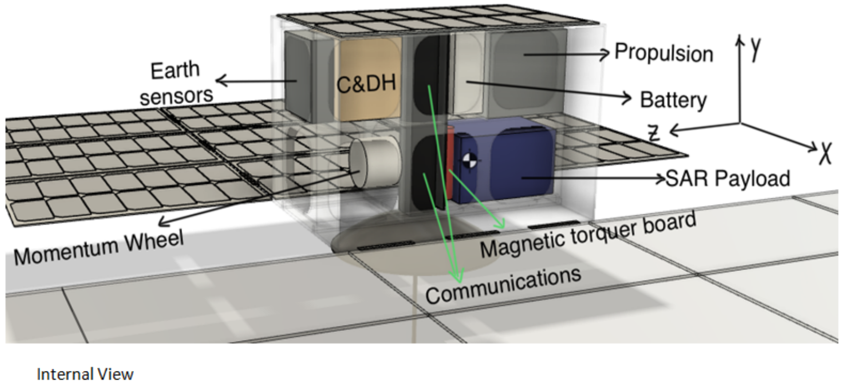

Figure 3.3: Internal Configuration of the PoleSat

CAD calculations of the deployed state are important to demonstrate the center of mass and the moment of inertia.The origin point of the spacecraft is at the bottom, front, left corner of the main spacecraft body.
- Center of Mass, unit : meters

$$CoM{=}0.1i + 0.145j + 0.102k$$

Moments of inertia are provided in Table 3.6.

$I_{xx} = 0.924$	$I_{xy} = 0.08$	$I_{xz} = -0.031$
$I_{yx} = 0.08$	$I_{yy} = 2.748$	$I_{yz} = 0.003$
$I_{zx} = -0.031$	$I_{zy} = 0.003$	$I_{zz} = 2.006$

Table 3.6: Moments of Inertia of Deployed System

3.5 Subsystem Design

The following subsections describe in detail the each subsystem and the rationale and design process that went through each subsystem.

3.5.1 Structure and Configuration

The spacecraft structure and configuration consists of two types of structures: the primary structure and the secondary structure. The primary structure will be the main 12U body and the platforms on the body which will hold all the components. The entire primary structure will be made out of Aluminum 6061 due to its high stiffness to weight ratio. Having the density of 2.70 $\frac{g}{cm^3}$, the final mass of the primary structure is 1.9 kg.

The spacecraft also consists of secondary structures. They are the solar panels, communications antenna dish, and the support trusses connecting them to the main body. The support trusses are also made out of Aluminum 6061. The solar panels are made out of Magnesium, since it's much less dense than Aluminum 6061 and doesn't need to handle the stresses of launch. With a density of 1.74 $\frac{g}{cm^3}$, Magnesium was the perfect choice for the panels which the solar arrays were going to be installed. The spacecraft comprised 136 solar cells in total, with each cell weighing 10g and powering an average of 1W. 126 solar cells will be placed on nine panels with 14 solar cells on each panel. 10 solar cells will be placed on top of the spacecraft. This will provide the spacecraft with 136W of power, which will be sufficient for the operation of our spacecraft. In total, the solar panels will weigh 2.62 kg and the support trusses will weigh 0.1 kg.

Finally, the PoleSat will have the communications antenna dish which will be a parabolic dish facing towards the earth from the bottom of the spacecraft upon its deployment. Weighing at 1.8 kg, it will total the entire spacecraft structure to a mass of 5.78 kg.

3.5.2 Attitude Determination & Control

The PoleSat's ADCS will need to be able to stabilize the spacecraft with the ability to meet the control performance requirements of the SAR payload. The PoleSat will be nadir-pointing its entire mission life. It will be orbiting at an inclination of 92° and the sensors and antennas will all be facing towards Earth. Because of this, the PoleSat will not be needing to do any slewing, since the SAR sensor will already be facing where it needs to be due to the nature of its orbit and mission. The PoleSat will be launched using the Falcon 9 and its orbital injection will be accomplished using orbital maneuvering rather than ADCS, which will be discussed in further detail later in the report. Before the actuators and sensors are determined, requirements for the sensors must be discussed. The determination and control accuracy of the SAR will be 0.5°. Since the PoleSat is handling such a large surface area of the poles, there is a more lenient accuracy requirement. The range of angular motion must be within 0.5° of nadir pointing and the drift rate must be less than 5° per hour. The ADCS control system was chosen based on these requirements explained above.

The PoleSat will experience a total external disturbance of 8.931E^{-5} N · m. The 10% margin assigned to the propellant mass will compensate for the minimal external and unexpected disturbances. Below are the equations explaining the total external disturbance torque the spacecraft will experience:

i. Gravity Gradient:

$$T_g = \frac{3 \cdot \mu}{2 \cdot R^3} \cdot [I_z - I_y] \cdot sin(2 \cdot \theta) = \frac{3 \cdot (3.99 \times 10^{14})}{2 \cdot (717 \times 10^3 + 6378 \times 10^3)^3} \cdot [0.274 - 0.182] = 1.54 \times 10^{-7} N \cdot m \quad (3.1)$$

ii. Magnetic Field:

T_m = D.B

D = residual dipole = 2 A $\cdot \text{m}^2$

B = magnetic field at the poles = $\frac{2 \cdot M}{R^3} = \frac{2 \cdot (7.96 \times 10^{15})}{(717 \times 10^3 + 6378 \times 10^3)^3}$

B = 4.46 $\times 10^{-5}$ Tesla

$T_m = 2 \cdot 4.46 \times 10^{-5} = 8.91 \times 10^{-5}$ N $\cdot m$

iii. Aerodynamic Pressure:

$$T_a = \frac{1}{2} . \rho . C_D . A . V^2 . (c_{pa} - c_g) = \frac{1}{2} (1.47 \times 10^{-13})(2.1)(0.06)(0.05)(7.49^2) = 2.6 \times 10^{-14} N.m \quad (3.2)$$

iv. Solar Radiation Pressure:

$$T_{sp} = \frac{F_s}{c} \cdot A_s \cdot (1+q) \cdot cos(i) \cdot (c_{ps} - c_g) = \frac{1000}{3 \times 10^8} \cdot 0.06 \cdot (1+0.6) \cdot 1 \cdot 0.05 = 1.6 \times 10^{-8} N \cdot m \tag{3.3}$$

The PoleSat will be controlled via passive magnetic and momentum bias methods. This is the best way to control the spacecraft to get the most torque for the least power for the PoleSat. For the PoleSat, the payload will need to be pointing towards Earth for the entirety of the mission, meaning it is nadir pointing. Due to this, there will be need to have 3-axis stabilization. The PoleSat will be utilizing passive magnetic method for two of the three axes of the spacecraft and momentum bias method for the third axis.

For choosing actuators, the decision was made that magnetic torquers were the best options because they are relatively light, reliable, and easy to operate since there are no moving parts. However, magnetic torquers can only operate in 2 axes since the third axes will be parallel with the Earth's magnetic field. This means only pitch and yaw moments can be controlled on the spacecraft. There is a need to control the 3rd roll axis, this is solved by adding a momentum wheel oriented for the third roll axis. The momentum wheel was the best way to solve this since it spin-stabilized the axis constantly compared to reaction wheels, which initially starts at zero momentum. However, the downside was that the reaction/-momentum wheels had to be installed with thrusters because they were needed for desaturation of the wheel when it became saturated.

For sensors, three types of sensors will be used to determine the attitude of the PoleSat. A sun sensor and an earth sensor was needed to be used together since Earth sensors could only provide vertical reference. The sun sensor provided the 3rd axis reference and completed the attitude determination of the spacecraft. Finally, a third type of sensor, a magnetometer, was needed for the operation of the magnetic torquers. It was used for the detection of the magnetic field relative to the spacecraft. Each sensor was placed in unobstructed locations to view its needs. Four sun sensors were placed on each side of the spacecraft for the best unobstructed view of the sun during its orbit. Two horizon/Earth sensors were placed orthogonally to one another, one placed in front and once placed on the side of the spacecraft facing angled towards earth.

Actuators	Properties
2x Magnetic Torquers (1 board) [12]	Weight: 0.196 kg (Board) Power: 1.2 W (Board) Performance: 0.2 A*m^2
1x Momentum Wheel [21]	Weightt: 0.185 kg Power: 1 W Performance: 0.04 N$\cdot m \cdot s$
1x Bi-Propellant Liquid Thruster [18]	Weight: 1.1 kg Power: 6 W Performance: 0.035 N$\cdot s - 5N \cdot s$
Sensors	**Properties**
4x Sun Sensors [15]	Weight: 0.035 kg (Each) Power: 0.037 W (Each) Accuracy: 0.1°
2x Earth Sensors [13]	Weight: 0.033 kg (Each) Power: 0.132 W (Each) Accuracy: 0.1° - 0.25°
1x Magnetometer [16]	Weight: 0.085 kg Power: 0.750 W Accuracy: 0.5° - 3°

Table 3.7: ADCS & Sensor Properties

Below is the calculation for determining the amount of propellant needed for ADCS thrusters for momentum dumping:

$\mathrm{F} = \frac{h}{L \cdot t} = \frac{0.04}{0.2 \cdot 1} = 0.2$ N

Total Pulses = 1 pulse $\cdot 1 wheel \cdot 365 \ \frac{days}{yr} \cdot$ years = 1095 pulses

F = 1095 pulses $\cdot 1 \frac{sec}{pulse} \cdot 0.2N = 219 \ N \cdot s$

$\mathrm{M}_p = \frac{Ft}{I_{sp} \cdot g} = \frac{219}{430 \cdot 9.8} = 0.0558 \ kg$

In conclusion, our ADCS subsystem will weigh a total of 2.035 kg including the needed propellant and an additional 11.2% of the propellant mass for compensation of external and random unexpected disturbances. It will also require a total of 9.362 W of power to operate the entire subsystem.

3.5.3 Electrical Power

Working through each specific subsystem, the total power used by each subsystem was identified and summed in order to have a specific power requirement for the PoleSat. A 30% power budget was included in order to ensure that all the power demands would be met in the case of an emergency and future power requirement during manufacturing. A table 3.5 which was included in section 3.4.4 illustrates that.

Using the power budget in combination with simulated values from FreeFlyer [9] an iterative process was used to derive values of different parameter involved in the design calculation of electrical power subsystem. A table below shows the calculated values:

The findings of the iterative process used for EPS found the mass of the battery was to be 1.54

Desired Variable	Calculated Value
Eclipse time (T_e)	1828.138 s
Sunlight time (T_s)	3881.138 s
Power Output of the Solar Array Needed in Sunlight Time (P_{SA})	257.923 W
Solar Flux	1339.98 W
Power Density Beginning of Life (P_{BOL})	946.2 W/m^2
Power Density End of Life (P_{EOL})	863.571 W/m^2
Desired Solar Array Size (A_{SA})	0.29867 m^2
Number of Cycles	16570.92
Depth of Discharge (DoD)	33%
Capacity of one Battery (C_r)	231.967 W-hr
Battery Mass	1.546 kg

Table 3.8: Summary of design process for Electrical Power Subsytem

Kg when using a Lithium Ion battery. A Lithium Ion battery was chosen due to that specific type of battery being very energy dense and allowing the team to minimize the PoleSat's battery mass. A diagram was found from reference [4] which correlates the depth of discharge to the number of cycles for a lithium ion battery, figure 3.4.

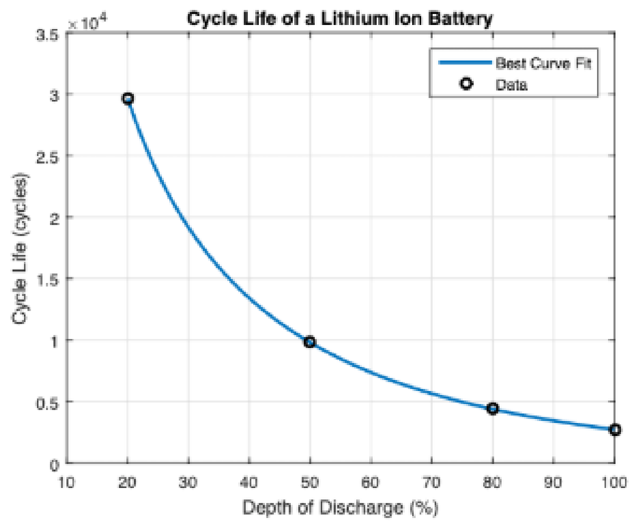

Figure 3.4: Number of cycles vs Depth of Discharge (DoD) graph for Lithium Ion Battery [4]

3.5.4 Communication

One of the most important subsystems of the spacecraft is communications, which ensures a proper transmission of data between the ground station and the satellite. It will also be used to control spacecraft from ground based operations. A top down analysis has been carried out in order to make sure that the communication requirements are met.

The design process of the communication subsystem was started by choosing the bit error rate (i.e. 10^{-5}), which dictates the required link budget for the spacecraft. Communication will be done employing L band frequency band which has frequency value of 1.6 GHz in uplink and 1.5 GHz in downlink. Binary Phase Shift Keying (BPSK) was selected to be a modulation scheme for the communications subsystem of our mission. After determining the required frequency band, the antenna diameter and mass values were calculated. The spacecraft's antenna configuration was chosen to be parabolic, with a size of 0.4x0.4x0.47m^3. The gain that the antenna is capable of having is 25 dB with the corresponding beamwidth of 32.8$^{\text{o}}$. The estimated mass of the antenna was determined to equal 1.8 kg.

Once the antenna configuration was determined, the required transmitter power value was calculated from the Friis Transmission Formula to be 45W, the corresponding weight of transmitter is 1.5 kg. The receiver's noise was calculated to be 18 dB and 24.6 dB respectively for each uplink and downlink mode. Table 3.9 shows the summary of the design variables and their calculated values.

Design Variables	**Calculated Values**
Frequency Band	L - Band
Modulation Scheme	BPSK
Bit Error Rate	10^{-5}
Antenna Configuration	Parabolic
Antenna Size	0.4m $\times$ 0.4m $\times$ 0.47m
Beamwidth	32.8°
Antenna Gains	25 dB
Antenna Mass	1.8 kg
Transmitter Mass	1.5 kg
Transmitter Power	45 W

Table 3.9: Summary of design process for Communication Subsytem

3.5.5 Data Handling

The onboard processor which is planned to be used is called Block Adaptive Quantization [3], [23], which is an algorithm that encodes the raw data with the compression range of 2-6 bits. This feature allows a selection of compression image quality trade-offs.

After determining the transmitter and receiver gains (22 dB and 37 dB respectively), the link budget calculations were made. There are also some losses associated with the data handling, which include atmospheric loss [23], antenna pointing loss [26] and rain loss [6]. The values for those losses were found from the corresponding references. The transmitter line loss value was determined from the Friis' Transmission Formula. The rate at which the information is transferred, also known as the Baud Rate, is equal to 160 Mbps.

Link budget and Link margin were calculated considering the worst case slant height of 3065 km, and their value equals to 25.3 dB and 20.3 dB respectively. The required value of link budget (i.e. 5 dB) comes from the chosen probability of bit error which is 10^{-5}. It can be noted that the PoleSat has relatively high link margin, which gives confidence for better transmission of data and communication with the PoleSat.

The corresponding values for each are indicated in table 3.10.

3.5.6 Thermal Control

The design process of the thermal subsystem starts by knowing the operating temperature limit of all the onboard subsystems, payload and other structural elements. Then the spacecraft attitude and orientation relative to the sun is predicted to set the boundary conditions for the mission. The max and min worst case temperature limits were calculated using the spherical satellite equations. It appeared

Transmission Gain	22 dB
Receiver Gain	37 dB
Transmitter Line Loss	-1.25 dB
Free Space Path Loss	-164.1 dB
Atmospheric Loss	-0.99
Antenna Pointing Loss	-4.3 dB
Rain Loss	-2 dB
System Temperature	260 K
Baud Rate	160 Mbps (i.e. 82.04 dB)

Table 3.10: Parameters involved in Link Budget Calculation

from the calculation that worst case max and min temperature falls within operating temperature range of all other subsystems, including Li-Ion Battery, on board propellant and on-board processor. But these formulas can give us first-order estimates of spacecraft temperature. However, an actual spacecraft will likely exhibit lower maximum temperatures and higher minimum temperatures than those predicted by the equations, due to transient effect ([25] p.448).

Therefore the team thought of having a backup to control the max and min temperature. A temperature sensor will operate like a switch to turn on or off the heaters. There will be Silicone Rubber Thermofoil Heaters, which are a kind of resistive heater and will control the spacecraft from reaching its lower temperature limit. It typically weighs only 0.04 g/ cm^2. Actual power required depends on the desired temperature change. To have a rough estimate of the power required by these silicone heaters, it was found that 50 g of silicone rubber heaters (which covers an estimated area of 1200 cm^2) needs 11.83 W to increase the temperature of spacecraft by 20 deg C in 60 seconds. Our power budget has a 30% margin which can account for this power requirement, if needed in future. Trade can also be made here between power required by these heaters and warm-up time. The spacecraft is less vulnerable to reach the max temperature limit, but again as a backup the team thought of using thermal paint called AZ-93 [2], which will reflect the solar rays and will help to maintain the inner temperature. The paint will be applied on the main body of the spacecraft.

The following table shows the max and min operating temperature range of various subsystems.

Subsystem	**Temperature range (°C)**
Telecommunications	-10 to +50
Payload	-10 to +40
OnBoard Processor	-10 to +50
Telemetry & Command Units	-10 to +50
Electrical Power, Lithium Ion Battery	-20 to +40
Attitude Control, Momentum and Reaction Wheels	-5 to +45
Propulsion, Thrusters	-5 to +35
Harness, Spacecraft internal	-15 to +55
Structures, Nonalignment Critical	-45 to +65
Parabolic Antennas	-160 to +95

Table 3.11: Temperature ranges

To find the max and min operating range of the spacecraft the calculations were made according to the SMAD text, where firstly the spacecraft is approximated as a cylinder and its total surface area is found, then the radius of sphere is determined which has equivalent surface area as cylinder. From the calculation the worst case minimum temperature and maximum temperature values were found to be -1.24°C and 34.3°C respectively.

Design Variables	Calculated Values
Absorptivity (α)	0.3
Emissivity (ϵ)	0.8
Surface Area of cylinder (A)	2513 cm^2
Cross section area of spherical satellite (A_c)	616 cm^2
Angular radius of Earth (ρ)	1.116 rad
Albedo correction (K_a)	0.9926
Electrical Dissipation (Q_w)	10 W - 50 W
View factor (F)	0.2805
Worst case (T_{max})	34.3°C
Worst case (T_{min})	-1.24°C

Table 3.12: Design Variables Values

3.5.7 Orbit Determination & Control / Propulsion

The selected launch vehicle from section 3.4.3, Falcon 9 is efficient enough to launch spacecraft in LEO. But the spacecraft will need propellant for orbit maintenance and correction. The team thought of using Liquid Bipropellant as it can successfully provide functions such as orbit maintenance, attitude control and also large ΔV maneuvers for orbit insertion in case it is required in future use. Nitrous Oxide and Propylene liquid bipropellant [18] is planned to be used for the PoleSat mission, the Isp value for this propellant is 430 sec.

The required ΔV for orbit maintenance, correction and station keeping was assumed to be 30 m/s - 40 m/s per year [24]. There will be the same propellant for orbit correction and attitude control. The total mass of required propellant was determined to be 0.61 kg using rocket equation. The total delta V it can produce is 130 m/s. The team also accounted for tank weight and weight of propellant management devices for the worst case scenarios based on references. The total estimated weight of the entire propulsion system was found to be 0.9 kg.

Design Variables	Values	References
Required ΔV	30 $\frac{m}{s}$ - 40 $\frac{m}{s}$ per year	[24]
Propellant type	Nitrous Oxide and Propylene	[18]
Propellant Mass	0.61 kg	
Tank Weight	0.0915 kg	[25]
Propellant management devices	0.183 kg	[25]
Total Weight	0.9 kg	

Table 3.13: Propulsion Data

4 Risk Assessment

Every successful space mission takes into account the risks involved in the mission to try and mitigate these dilemmas which would otherwise compromise and have a negative impact. Identifying risks and being able to provide reasonable responses to these problems increase confidence in an individual mission. In order to identify risks for the PoleSat, a table was made to first describe the rating that each individual risk may possess. The figure below of the likelihood vs. impact illustrates the ratings that will be used to measure the risks identified. For the likelihood metric, it will range from very unlikely to very likely, and for the impact metric the range will be negligible to severe. Each metric has an associated number with it, and these numbers will be used as a tool to measure the average risk for each respective subsystem to have an understanding of how at risk each subsystem is.

Upon completing the way in which each risk will be analyzed a risk table 4.1 was created to identify the risk category as well as the risk type with their respective impact and likelihood rating. From table 4.1, a trend appears that the risks may be severe, however the likelihood rating appears to be low. Having a low likelihood rating comes from the design process which allows confidence to be instilled into the solution of these problems Another table was made to showcase how the design of the PoleSat solves these risks.

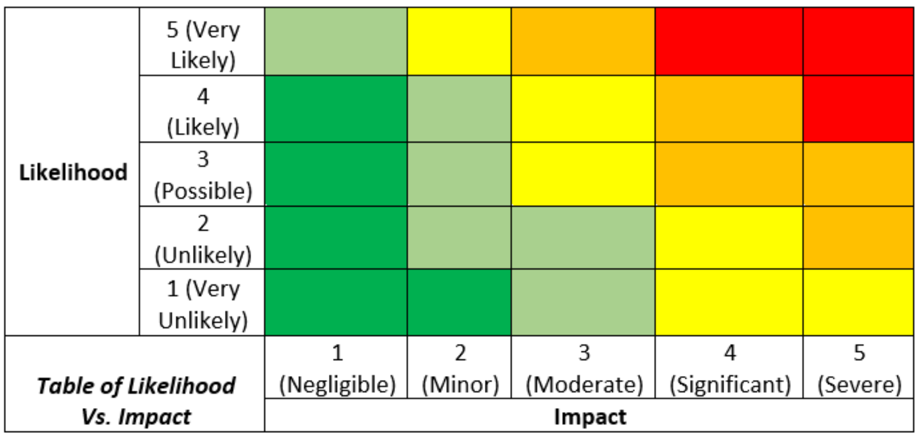

Figure 4.1: Impact vs. Likelihood

Risk Category	Risk Type	Impact	Likelihood
Structure & Configuration	Vibration during takeoff disturbing performance	3	3
Structure & Configuration	Deformation which causes improper configuration	4	1
Average		3.5	2
EPS	Battery degradation and failure	4	1
EPS	Not enough power to meet demands	3	1
EPS	Loss of power permanently	5	1
Average		4	1
C&DH	Loss of communication with satellite	4	2
C&DH	Data Transmission Failure	3	1
Average		3.5	1.5
ADCS	Spacecraft gets out of orbit	4	1
ADCS	Impact by space debris no longer able to control	5	1
Average		4.5	1
Thermal Control	Exceeds temperature limits	3	1
Average		3	1
Scheduling	Failure to meet deadlines for mission success	2	1
Average		2	1

Table 4.1: Risk Assessment Table

Mitigating the risks identified instills confidence into the proposal for the PoleSat. Whether this was done by choosing the worst case scenario as the limit or choosing a material which has specific characteristics that are valuable. A specific example of this is the worry of not having enough power to last the mission, a 30% power budget is included in the mission to ensure that the power available to the PoleSat is not limited. The overall evaluation of the risks posed on the PoleSat have been determined to be manageable and the PoleSat seems like it will be a successful mission.

The results of the derivations and risk assessment demonstrate the effectiveness of the PoleSat, and show the team's hard work in producing a thoughtful and meaningful solution to NASA's CubeSat launch initiative. The PoleSat is satisfactory and provides an interesting proposal.

Risk Category	**Risk Type**	**Mitigation**
Structure & Configuration	Vibration during takeoff Deformation	Material selection of Al6061 has a high young's modulus which will limit deformation and damage caused by vibrations.
EPS	Battery degradation Not enough power Loss of power	Battery calculation considered a higher battery degradation rate Power calculation included a 30% budget Battery has a life expectancy of 3 years
C&DH	Loss of communication Data Transmission Failure	Link budget & margin calculated for worst case scenario Low probability of bit error
ADCS	Spacecraft gets out of orbit Space Debris	Orbit maintenance ensured by a propulsion system with a ΔV large enough to ensure proper ability to maintain orbit. Proper selection of actuators allows for system redundancy.
Thermal Control	CubeSat exceeds temp limits	Design process has a backup of active thermal heaters and passive temp control system.
Scheduling	Failure to meet deadlines	Team communicated and worked effectively to accomplish the mission.

Table 4.2: Risk Mitigation Table

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