POLARIS-CT, Polar Orientated Lunar Analysis of Radiated Infrared Surface & Cold Traps

Ahid Chenno Ruiz(1), **Oliver Becquart**(2), **Colin Tan**(3), **Matthew Enfield**(4), **Lyle Desborough**(5), **Stanford Burley**(6)

MSc Group 1
(1) *31470998, acr1n24@soton.ac.uk*
(2) *36507342, ob2u24@soton.ac.uk*
(3) *31990827, ckt1u24@soton.ac.uk*
(4) *36606782, mje1n24@soton.ac.uk*
(5) *31955584, led1g24@soton.ac.uk*
(6) *36643793, sb16g24@soton.ac.uk*

ABSTRACT(1)(4)

With new space expeditions on the horizon, such as the Artemis project, humanity is looking to the stars and is willing to return to the Lunar surface once again. Humans are destined to establish a permanent presence on the Moon and provide a sustainable habitat for all Astronauts willing to push humanity forward. To accomplish such an endeavour, missions determining favourable locations for a potential permanent presence are increasingly needed. POLARIS-CT is a mission developed at the University of Southampton, using a concurrent engineering design approach, that aims to provide thermal mapping of the South Pole. This mission aims to provide a comprehensive thermal mapping of regions of higher temperatures - areas of potential permanent presence. This report delivers the final design of the mission, whilst detailing the design process that has transpired between specialised subsystem teams, culminating in an effective collaborative design effort for POLARIS-CT.

1 INTRODUCTION(1)

1.1 Mission Statement

Utilising Infrared Radiation to assemble a thermal map of the Lunar South Pole, determining the temperature stability around cold traps in sun lit regions, and evaluating areas of interest for the potential of habitability on the Moon.

1.2 Mission Objectives

With the mission aims determined, the CubeSat Lunar mission objectives were set and split into a primary and a secondary group.
To conduct an analysis of the location and temperature stability of the south polar region, the primary set of objectives intends to generate thermal images of the Lunar South Pole surface, with the addition of potentially locating sub-surface ice deposits.

Once thermal mapping of the South Pole region is generated, POLARIS-CT will collect data on other locations, such as the North Pole or heavily cratered areas. The thermal mapping will further assist with the aim of locating cold traps, and in turn provide crucial data to determine the potential habitability of the Moon.

1.3 Mission Requirements & NRDD

Once the objectives were agreed upon, the mission requirements were determined considering the requirements and constraints set by the Nanoracks DoubleWide Deployer (NRDD) system [1].

The CubeSat must be located in a Low Lunar Orbit (LLO), with an inclination of 90° for maximum coverage of the South Pole. Additionally, the mission must be able to cover at least 90% of a circular area centred around the exact South Pole, up to 85°latitude S in every direction. Furthermore, the CubeSat will require a minimum mission lifetime of 2 years to gather all the data necessary and conduct a successful study. The CubeSat must be capable of conducting a spin manoeuvre as necessary to align the infrared sensor onto the desired points of interest.

Considering the end of life of the CubeSat, it will de-orbit onto the Lunar surface and impact at a velocity of no more than 2 km/s to mitigate the effect of Lunar regolith dispersing into orbit, and conform with current Space guidelines, and literature [2].

Finally, with consideration of the NRDD, the CubeSat is constrained to a 6U size to fit within the CubeSat deploy system, with a total mass of 12 kg. Furthermore, it must be capable of communicating effectively with the relay satellite located in the Lagrange point 2 (L2) to transmit the data collected throughout the mission from the far side of the Moon to the Deep Space Network (DSN) on Earth.

1.4 Concurrent Design Framework[4]

Following a structured waterfall methodology, Fig.1 demonstrates the approach used to create the POLARIS-CT CubeSat Mission.

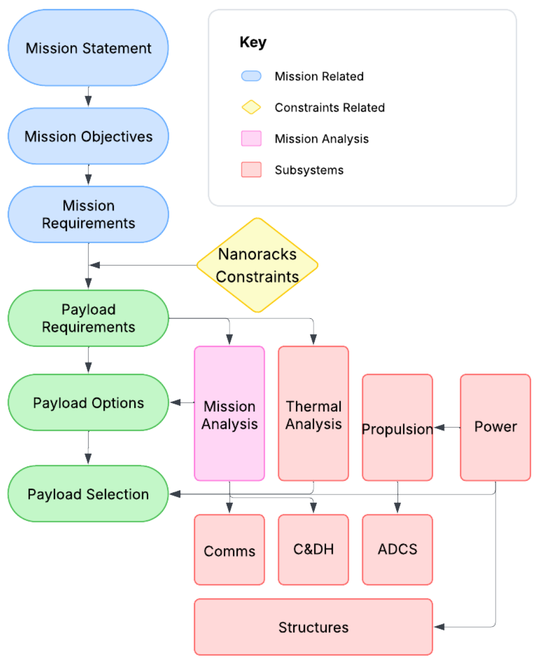

Figure 1: Diagram Reflecting Concurrent Engineering Design Process

2 PAYLOAD[4][1][2][3][5]

Key Payload Requirements:

1. Be able to detect infrared radiation with a wavelength range of 7 μm - 50 μm.
2. Should have a mass of less than 6 kg.
3. Should be able to achieve a spatial resolution of less than 1 km.
4. Must be able to withstand space environment.

The FLIR Boson+ 320 [3] is a commercially available Long Wavelength Infrared (LWIR) thermal camera module, manufactured by Teledyne FLIR. This payload was selected in combination with a ruggedised casing to protect it from launch vibrations and solar radiation. This is a relatively simple custom casing made from a 3D printed material; Acrylonitrile Styrene Acrylate (ASA), known for excellent solar radiation resistant properties. Altogether, the thermal camera and casing fit within a 1U volume and have exceedingly low mass, whilst being able to deliver a moderate resolution at low orbits. Furthermore, estimates for the expected thermal environment were received from the thermal analysis team and translated into a quantified required operational temperature range.

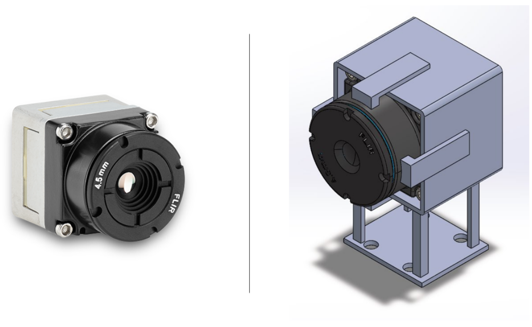

Figure 2: Left: FLIR Boson+ 320 4.5 mm, Right: CAD Model of protective casing

2.1 Trade-offs

All commercially available IR detectors have a spectral response range of approximately (7 μm - 15 μm) as they are mostly designed for terrestrial use. For extra-terrestrial missions, the IR wavelength range becomes larger as temperatures drop to near 0 K, where only highly specialised instruments are capable of detecting. Based on the expected temperature of cold traps, a range of 7 - 50 μm was determined as a useful range. Designing a specialised payload capable of detecting these longer wavelengths, whilst simultaneously meeting the tight constraints of the mission, was determined to be unrealistic. Therefore, a trade-off of spectral response was made in favour of simplicity, volume, mass and availability. Consequently, no temperature data will be available inside the cold traps as the expected temperatures are constantly below 100 K, although the locations may be discernible from the gaps in data. Neither will there be data for the dark side of the Lunar surface, leading to a carefully designed operational plan for the payload.

Table 1: Payload Characteristics

Parameter	Value
Mass	12.5 grams
Power	Up to 1 Watt
Volume	1U
Data rate	12.2 kbps
Frequency of data production	Once per orbit

3 MISSION ANALYSIS[4]

The orbit altitude was chosen to provide a balance between spatial resolution and orbit stability.

Table 2: Orbital Parameters Determined

Orbital Parameter	Value
Radius (r)	1937 km
Inclination (i)	90°
Eccentricity (e)	0
RAAN (Ω)	45°
AOP (ω)	0°
Period (τ)	127.5 mins

At an altitude of 200 km, this orbit is stable over a three-year period, which allows operating beyond the current planned lifetime of two years. A polar orbit was necessary for this mission to fulfil the objective of thermal imaging the south pole area of the Moon. This requires the centre of the resulting compiled image to be at the south pole - only possible with a polar orbit.

3.1 Concept of Operation

A circular area centred around the south pole with a radius of 5°of latitude is the desired area of coverage from which the detector will receive data. However, due to the nature of the moon's orbit, half of this area will always be in darkness. The Boson+ 320 is not able to detect significant emission radiation from the Lunar surface in shadow because expected temperatures do not exceed 100 K without sunlight. So, in order to use the payload only when necessary, three modes of operation are to be used over the solar year period.

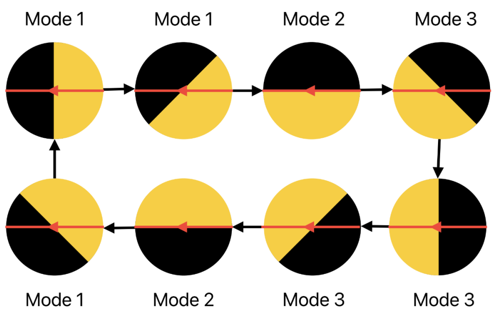

Figure 3: Illustration of payload modes of operation during one solar year. Yellow = Light Side, Black = Dark Side, Red = Spacecraft Trajectory

The modes of operation indicating the section of the path that the payload will be active for:

- Mode 1: Southbound 85°to 90°
- Mode 2: No active zone
- Mode 3: Northbound 90°to 85°

The modes of operation are implemented to make use of the payload as efficiently as possible by limiting data collection to discrete areas of interest. Due to the termination line across the south pole making a 360°rotation once per solar year, the modes of operation will proceed as shown in Fig. 4.

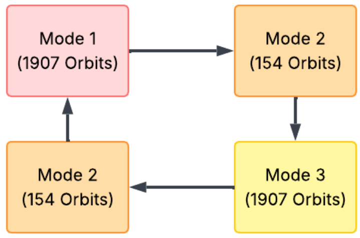

Figure 4: Modes of operation cycle over a solar year period

The payload will collect data once per orbit and will take one 360°rotation of the moon to capture a full map of the South Pole. During an active period of one orbit, 1.31 Mb of data will be collected and stored on a hard drive. After leaving the area of operation, the shutter will close, and the payload will be put on standby. Additionally, to confirm nominal operation before data collection, the payload will be taken off standby at an inclination of 5°prior to entering the active zone, and system checks will be run. This allows for 106 seconds of buffer time to check systems and prepare for payload activation.

4 SUB-SYSTEMS

4.1 Propulsion (4)

Propulsion is solely required to successfully de-orbit POLARIS-CT into the Lunar surface. At the end of life, the Busek BGT-X5 [4] propulsion system will fire retrograde in order to deliver sufficient ΔV to the CubeSat, decreasing it's orbital velocity and ultimately intersecting with the Moon.

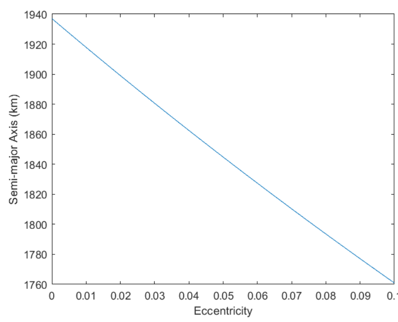

Figure 5: Plot showing eccentricity and semi-major axis of de-orbit transfers, providing values for STK simulation

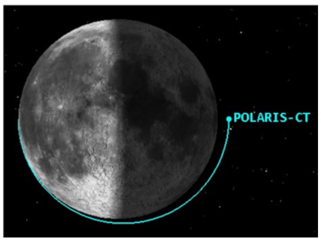

Figure 6: De-orbit trajectory

It was determined via MATLAB and STK 12; the minimum ΔV needed to de-orbit the spacecraft is 44 m/s. At this impulse, the velocity on impact would be 1.73 km/s; a reasonable impact velocity based on previous missions. To support the validity of this method, NASA executed a planned de-orbit of the LADEE mission into the Lunar surface in 2016, with an impact velocity of 1.6 km/s [5]. Additionally, the LCROSS mission impacted the Moon's surface in October 2009 with an impact velocity of 2.5 km/s [2]. Without a definitive guideline for de-orbit impacts on the Moon, it has been concluded that the reasonable maximum impact velocity of POLARIS-CT will not cause significant damage to the surface or excessive debris.

$$\Delta p = m\Delta V \tag{1}$$

From communicating with other subsystem teams to get an estimate for overall mass, it was determined that the mass would likely be less than 6 kg. Using Eq. 1, the minimum required impulse to successfully de-orbit POLARIS-CT is 264 Ns. From considering propulsion options, it was clear that the Busek BGT-X5 would be the most suitable choice and would be able to deliver the required impulse whilst satisfying tight constraints.

$$m_p = m_0 \left(1 - exp\left(\frac{\Delta V}{I_{sp} g_0}\right)\right) \tag{2}$$

The minimum specific impulse of the 'ASCENT' green monopropellant fueled system is 220 s, leading to a minimum required propellant mass of 121 grams. The final dry mass of POLARIS-CT after preliminary design is 5.885 kg. The propulsion system will be kept for the 6 kg configuration as the extra impulse is beneficial for minimising de-orbit impact velocity and providing redundancy in the event of the propulsion system not functioning at maximum thrust level.

Table 3: Performance Index of Busek BGT-X5.

Performance	**Value**
Max. Impulse	565 Ns
Specific Impulse	220-225 secs
System Mass	1.5 kg BOL
Power Draw	20 Watts

4.2 Attitude Control(1)

Within the CubeSat, there is a designated attitude determination and control system (ADCS) specifically selected for this mission. The ADCS selected was the RWA05 [6] due to its 3-axis high pointing accuracy system, and its extensive heritage. The RWA05 contains four reaction wheels, a star tracker and an integrated inertial measurement unit (IMU). To manage the pointing requirements of the payload, solar panels, and communication uplink, the reaction wheels are intermittently activated during Modes 1 & 3. Intermittent activation ensures that the reaction wheels do not become oversaturated, lubricant degradation is prolonged, and that they do not create any extra vibrations that could affect the systems onboard.

To determine the reaction wheel saturation occurring over the mission lifetime, firstly, the angle change θ, required for the CubeSat to always be nadir-pointing, was obtained using Eq. 4, from the arc length formula.

$$s = 2 * Sat_{altitude} * \tan(FOV/2) \tag{3}$$

$$\Delta\theta = s/r \tag{4}$$

Where s was the ground swath width of the orbit at 186.5 km obtained from Eq. 3 with the FoV being 50°, and r was the radius of the orbit found in Tab. 2, providing a value of 5.5° for $\Delta\theta$.
To obtain the total angular momentum, the moment of inertia was calculated with the use of Eq. 5, where the mass of the CubeSat, m, was 5.885 kg, and the perpendicular width and height of the spin axis were 0.2263 m and 0.34 m respectively.

$$I = \frac{1}{12} m(w^2 + h^2) \tag{5}$$

Providing a value of 0.082 kgm^2, which was then implemented onto Eq. 6, alongside ω_{orbit} which determined the angular velocity along half of the orbit, this being 1.4e-10 deg/s.

$$H = I * \omega_{orbit} \tag{6}$$

A total angular momentum of 1.18e-4 Nms per orbit was achieved, signifying that for the lifetime of the mission, considering the rate of angular change when pointing nadir only occurring at Modes 1 & 3, the CubeSat will accumulate a reaction wheel saturation of 0.4504 Nms, which is just below the maximum 0.5 Nms the RWA05 system can sustain.

4.3 Communications, Command & Data Handling(2)

For communications, the pairing of an ISISpace VHF Uplink/UHF Downlink full duplex transceiver [7], with the

ISISpace Antenna System for 6U CubeSats [8] was chosen due to their effective compatibility with each other, as well as both being low mass and having a low power consumption. It has a reliable deployment system as well as a proven flight heritage since 2016.

Table 4: Technical Parameters for CubeSat Communication System

Parameter	Value/Description
Transceiver Type	VHF Uplink/UHF Downlink Full Duplex
Frequency Range (Uplink)	VHF: 140–150 MHz
Frequency Range (Downlink)	UHF: 400–450 MHz (435 MHz used)
Data Rate (Downlink)	Up to 9,600 bps
Data Rate (Uplink)	Up to 9,600 bps
Modulation	FM (supports BPSK)
Transceiver Power Consumption	Low power ($<$ 2 W typical)
Transmit Power (CubeSat)	2.3 dBW (nominal), -0.7 dBW (eclipse)
Antenna Configurations	dual dipole
Antenna Frequency Range	100–500 MHz (covers 435 MHz)
Antenna Deployment	Automated, sequential
Supply Voltage (Antenna System)	4 V to 17 V
Mass (Transceiver)	$\sim$100 g (typical)

To correctly model the communication system, every driving parameter has to be considered. The orbital period is approximately 7214 seconds (120 minutes), which influences the communication windows and eclipse periods. During an eclipse period, in the shadow of the moon, the transmit power will drop by 3 dB (from 2.3 dBW to -0.7 dBW, meaning the wattage is less than 1 milliwatt) to conserve power when the solar array is not active and due to reliance on the battery pack. This lowers the signal-to-noise ratio (SNR) and increases the bit error rate (BER), which reduces the communication system's reliability.

The system operates at a frequency of 435 MHz in the ultra-high frequency band (UHF), a bandwidth of 10 MHz supports a data rate of 9600 bps.

The distance between the CubeSat and the Lagrange point (L2) relay satellite, being the QUEQIAO supplied by the China National Space Administration [9], is 61500 km from the moon's centre, which varies between 59,563 km and 63,437 km, causing the free-space-path-loss (FSPL) to vary, thus causing the SNR between the CubeSat and the L2 relay satellite to fluctuate. This can be seen within Fig. 7.

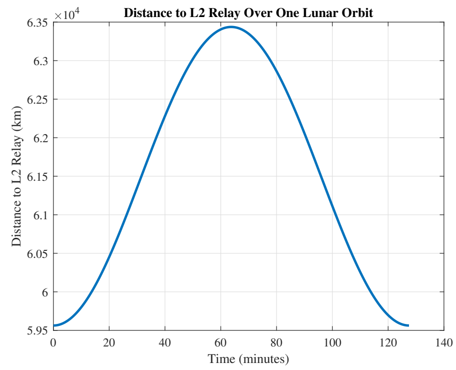

Figure 7: Distance to L2 over one orbit

This curve reflects on the SNR and BER rate as can be seen in the following Fig. 8 and 9.

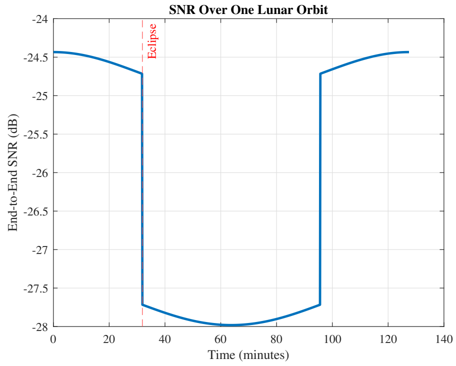

Figure 8: Signal to Noise Ratio Along One Orbit

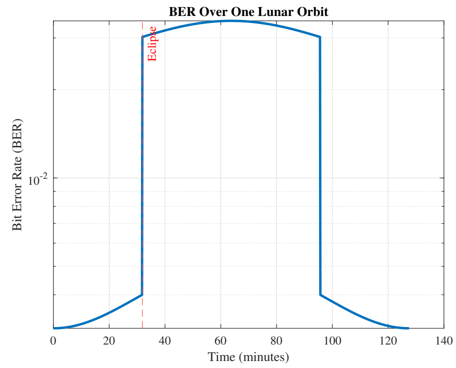

Figure 9: Bit Error Rate Along One Orbit

Thanks to the Lagrange point satellite, the CubeSat would be capable of maintaining near-continuous communication with the deep-space network at all times; however, to conserve power and reduce noise and errors, the satellite would transmit collected data back to Earth when the moon does not eclipse it.

As the primary payload collects data, it is uploaded to the ISISPACE On-Board Computer (IOBC) for processing and storage [10]. The FLIR Boson+ 320 has a data rate of 12.2 kbps with a data collection frequency suitable for the selected orbital height and area of interest. As one image is collected per track, 1.31072 Mb is collected, processed and stored. The time it takes to process the image with the clock speed of the onboard computer is 0.0033 seconds, and the time it takes to transmit the data back to Earth is 156.1 seconds. The following Tab. 5 shows the specifications of the onboard computer.

Table 5: Specifications of ISIS On Board Computer (iOBC)

Specification	Value
Processor	ARM9
Processor Speed	400 MHz
Storage	2×2 GB SD Cards
Flight Heritage	Yes
Power consumption	400 mW average

The onboard computer system will also be responsible for receiving, handling and performing commands for controlling attitude, orientation, propulsion, solar array deployment, etc.

Having communicated with other subsystem teams, an estimate for SNR and BER was acquired thanks to eclipse and orbit parameters, as well as power, which is important for transmitting data back to Earth.

4.4 Thermal Control(5)

4.4.1 Operational and Survivable Ranges

The thermal control of POLARIS-CT begins with identifying the operational and survivable temperature limits of its components. These temperature ranges define the bounds in which each component can reliably function and remain undamaged. Tab. 6 summarises these temperature specifications for the main components of the CubeSat. These temperature ranges will be the main constraints, guiding the thermal model and design of any passive and active thermal control systems.

Table 6: Operational and Survivable Temperature Ranges

Component	Operating [°C]	Survival [°C]
Solar Panels [11]	-40 to 125	-100 to 150
Batteries [12]	-20 to 70	-40 to 85
IR Camera [3]	-40 to 80	-50 to 85
Propellant [13]	0 to 50	-20 to 60
Sun Sensor [14]	-30 to 85	-40 to 95
Reaction Wheels [6]	-20 to 60	-30 to 70
On-Board Computer [15]	-25 to 65	-30 to 70

4.4.2 Steady States

To assess how POLARIS-CT behaves in its most extreme orbital conditions, two scenarios were modelled using the worst-case thermal environments: the 'Hot Steady State' and the 'Cold Steady State'. These environments will be encountered during full solar exposure and complete eclipse, respectively.

Table 7: Hot & Cold Steady States

Hot Steady State [°C]	Cold Steady State [°C]
58.85	-60.15

4.4.3 In-Orbit Temperature (Lumped Model)

While steady-state analysis provides worst-case temperatures, it is more important to evaluate the dynamic temperature of POLARIS-CT throughout a full orbit. A lumped thermal model was developed for this purpose, assuming a uniform internal temperature distribution across the spacecraft. The resulting temperature profile can be seen in Fig. 10, which shows the temperature of the CubeSat over multiple orbits. The graph reveals how quickly the CubeSat heats and cools, and serves as a useful estimator of the overall temperature of the CubeSat, despite the lack of an internal temperature gradient.

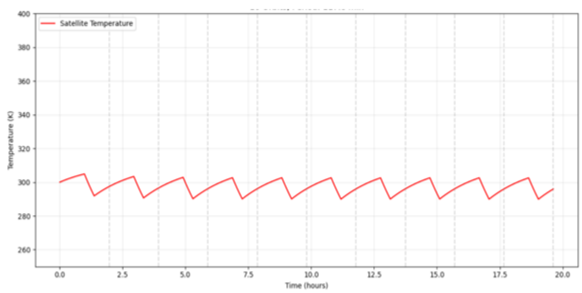

Figure 10: In-Orbit Temperature (Lumped Model)

4.4.4 Simulation

To supplement the lumped thermal model and gain insight into the internal heat distribution, a thermal simulation was performed using SolidWorks. The simulation includes thermal inputs such as solar radiation, albedo, internal heat generation, and radiative heat loss, and uses the minimum and maximum temperature values from Fig. 10 as initial temperatures.

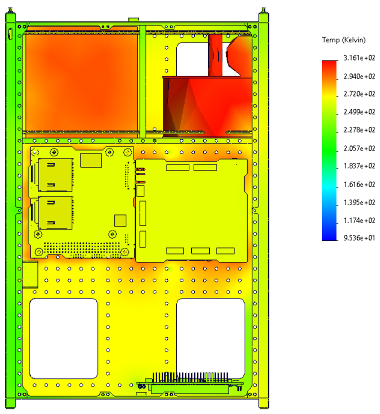

Figure 11: SolidWorks Thermal Simulation, Eclipse

The simulation concludes that, when exposed to the sun, no active heating or cooling is required. However, during eclipse, an active heater placed onto the CubeSat's base frame could be required to generate up to 10 W of thermal power to keep these components within operational ranges listed in Tab. 6.

4.5 Power Systems[3]

4.5.1 Estimated Power Consumption

The POLARIS-CT's 2-year mission will feature two main operating modes per orbit, an imaging/data collection phase and a data transfer (X-link) phase. Additionally, there will be a planned end-of-life (EoL) phase, which will involve de-orbiting the satellite and disposing of it on the lunar surface. During the imaging phase, the satellite will be in illumination for 127.5 minutes with an eclipse period of 44.8 minutes (approx. 35.2% of orbital period). The battery will be charged throughout the imaging phase, during which the imager and associated systems will be run off solar-generated power. During the X-link phase, select components will be switched to standby mode, namely the payload, sun-sensor and propulsion to reduce load on the battery. Components associated with the data uplink to the L2 satellite [9] will run solely off battery power.

With these considerations, an estimate for the power consumption for the main operational phases was completed to size up the key components of the power system: the battery and solar panels, detailed in Tab. 8.

Table 8: Estimated Power Consumptions

Subsystem	**Idle [W]**	**Active [W]**	**Eclipse [W]**	**Safe [W]**
Payload	0.0500	1.0000	0.0500	0.0
Sun Sensor	0.0693	0.0759	0.0693	0.07
ADCS	0.7800	3.0000	3.0000	0.78
Comms	0.0300	2.3000	2.3000	2.3
Computer	0.4000	0.4000	0.4000	0.4
Thermal	5.0000	10.0000	10.0000	5.0
Propulsion	0.0000	20.000	0.0000	0.0
EPS	0.1620	2.5000	2.5000	2.5
Total	**6.4913**	**39.2759**	**16.8193**	**13.3**

Note that, while the total active draw includes the propulsion system, it is planned to be activated only during the planned deorbit phase. Thus, the operational power draw, throughout the majority of the mission, would be 19.2759 W.

4.5.2 Power generation

A preliminary estimate of the solar array size was defined analytically using the total active power draw at a time, resulting in a required array area of 0.1708 m^2. Two 6U Deployable Solar Arrays, produced by Endurosat [16], were selected for power generation, featuring redundant burnwire mechanisms for deployment. The panels utilise triple junction GaAs cells with an estimated packing and EoL efficiencies of 90% & 29%, respectively. Throughout the mission, only 3 out of 6 panels will be exposed to direct sunlight, resulting in a total array area of 0.18 m^2; that being the case, the minimum power generated by the array configuration, assuming passive orientation, was estimated to be 50.2358 W.
To substantiate this figure, the panel output was modelled within MATLAB over two orbits (Fig. 12), assuming a maximum solar incidence angle of 45°, varying sinusoidally.

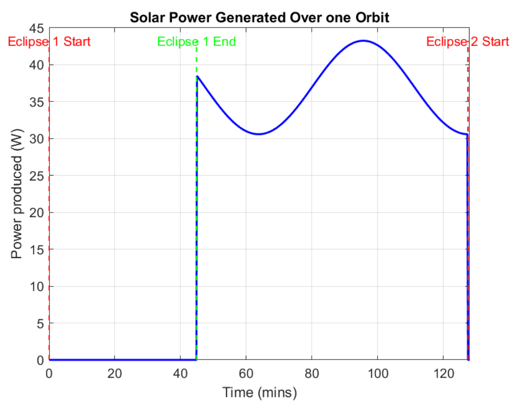

Figure 12: Power generated by three Endurosat 6U arrays over two orbits

The model estimated a peak power production of 43.26 W, with an average power generated in sunlight approximating to 30.10 W. While the satellite is within sunlight, the solar arrays are required to provide power to the subsystems as well as replenish the battery. Thus, the energy remaining to charge the batteries was determined to be:

$$E_{charge} = E_{array} - E_{operational} = 10.8241W \tag{7}$$

where E_{array} denotes the average energy produced by the panels and $E_{operational}$ indicates the power draw required for the imaging phase (excluding propulsion). Over the sunlit period, the energy produced by the arrays available for charging will be 14.9192 Wh.

4.5.3 Power storage

The required battery capacity is determined by the power required when the satellite is in eclipse, estimated to be a maximum of $\sim 16.8W$. This was computed through:

$$Wh_b = \frac{E_{eclipse} \cdot MOS}{DoD \cdot \eta_{charge}} = 85.93Wh \tag{8}$$

Where $E_{eclipse}$ indicates the energy required over the eclipse duration, MOS represents the margin of safety (30%), DoD is the depth of discharge (20%), and η_{charge} designates the charging efficiency (95%). This requirement led to the selection of the ISISPACE Modular EPS 2 [17], a modular power management system (PMS) configured with two 45 Wh battery packs (4s2p), resulting in a 90 Wh power storage unit. This selection allows for an additional $\sim 4.3\ Wh$ of capacity on top of the margin of safety, mitigating against unexpected transient spikes. The PMS only allows for an operating voltage range of $V_{bat} = 8\ V - 16\ V$. Nevertheless, the PMS features a fully integrated power distribution unit that will accommodate subsystems that operate outside of this range, namely the payload, which requires a step-down to 3.3 V. The selected operating voltage for the battery unit will be 16 V to minimise losses due to the circuitry.
The battery depletion during the eclipse must be determined to compute the energy required to recharge it during the sunlit period. Given an eclipse period of $\sim 0.747\ hrs$, the battery will be depleted to 77.4416 Wh (real $DoD =$ 13.95%), requiring an energy input of 12.5584 Wh to fully recharge. Accordingly, this is within the means of the power generation unit. Thus, the satellite will be able to operate solely off battery power throughout the eclipse periods. For the 2-year estimated mission duration, a key concern is capacity fade. The battery unit was predicted to undergo $\sim$ 3860 equivalent full cycles, adjusted for a typical charging efficiency and average degradation factor, leading to an estimated final capacity of 87.03 Wh at EoL.
The power subsystem was driven by all other subsystems within the mission, in particular high power draw components such as the propulsion, thermal control, attitude determination/control and communication systems, requiring substantial collaboration with the relevant subsystem technical leads to ensure a synergetic design.

4.6 Structures[2]

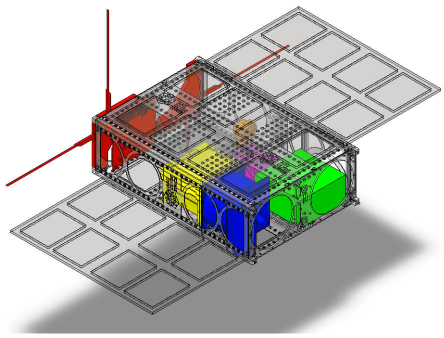

Figure 13: Isometric view of POLARIS-CT

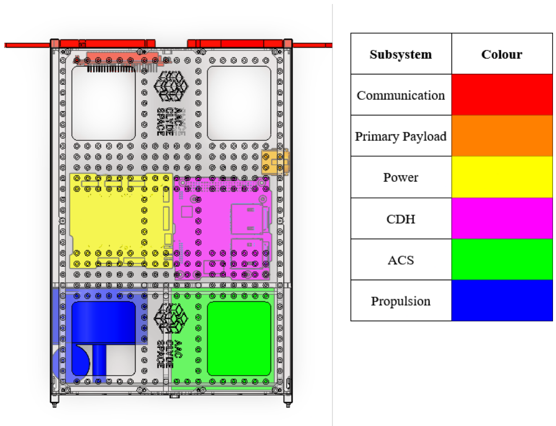

Figure 14: Component Layout

The CAD model of the POLARIS-CT is shown in Fig. 13 and 14. The frame is an AAC ClydeSpace 6U CubeSat frame [18], made entirely of Aluminium 7075 & 6082, with four deployment switches. It is also lightweight and allows for a high level of modularity and compatibility with the Nanoracks deployer, thanks to its rails. The entire satellite weighs 5.885 kg and is contained within 6U volume when all the deployable mechanisms are retracted. The 6U frame is slightly modified to allow the sun-sensor to be unobstructed, an area is cut out to allow the sensor to retain its full field of view and the areas of the frame around that are reinforced.

The communication antennas have a deployment system that relies on the antennas being under strain when retracted, which roll out to be straight on command. Similarly, the solar arrays, which are flush with the front and back faces of the CubeSat, rely on burn-wire mechanisms for hinge deployment to fold out.

The primary payload camera's casing also serves as a mount for the camera to be able to be bolted to the 6U frame. All components are bolted to the frame except for the attitude and sun-sensor system, which is fixed in place by rails at the bottom of the frame.

Coordination had to be made with all other subsystems and mission design teams for selection, placement and design of the final CubeSat structure.

5 SYSTEM BUDGETS(1)

5.1 Mass Budget

Specified within Section 1.3, the maximum allowed mass of the CubeSat dictated by the NRDD had to be 12 kg. POLARIS-CT has a mass of 5.885 kg split into seven subsystem categories as demonstrated within Tab. 10, with a centre of mass located at the following locations: X-axis (52.7 cm), Y-axis (89.7 cm) & Z-axis (128 cm).

Table 9: Mass Budget Specifications

POLARIS-CT Subsystem	Mass (kg)
Structure	0.674
Payload	0.0125
Propulsion	1.5
ADCS	1.1
CCDH	0.275
Thermal	0.1416
Power	2.182
Total Mass	**5.885**

5.2 Power Budget

Within Section 4.5, Tab. 8, the estimated power consumption was obtained for each subsystem, where three independent states (idle, active and eclipse) were studied to demonstrate what the CubeSat will experience during a single orbit. To obtain the total Wattage the power system will experience during the lifetime of the mission, it was assumed that the idle and active states would only attain energy during Modes 1 & 3, whilst the eclipse state would attain energy from all modes. With this in mind, a total estimated power consumption of 24,758 W, 73,518 W & 69,329 W, was obtained for the idle, active & eclipse mode over the 2 year lifetime of the mission.

5.3 Volume Budget

Considering the NRDD requirements and POLARIS-CT being a 6U CubeSat, the BGT-X5 required a form factor of 1U with the nozzle facing outwards, which meant it had to be located on the left side of the CubeSat to avoid interference with the rails. Furthermore, the internal components, such as the avionics and power systems, were constrained to a rail envelope of 366 mm by 245 mm in length and width, respectively. Additionally, the payload had to be directly nadir of the Lunar surface to ensure maximum coverage, which required a hinge setup and enough room for the camera to rotate into position. Finally, the antennas & solar arrays could not protrude during launch and deployment, thus they are retracted until release from the NRDD. With all measurements in mind, POLARIS-CT has a total volume with the components discussed of 0.002282 m^3.

5.4 Cost Budget

Due to high-level restrictions for each of the subsystem components, not every cost was obtained, as either it was not publicly available, or companies did not respond with a quote*. It can still be estimated that the cost of building a fully functional CubeSat is quite significant, even without considering launch costs, demonstrating how rigorous considerations have to take place during the design, manufacturing and testing of the CubeSat to ensure it functions as expected upon deployment.

Table 10: Cost of Parts

POLARIS-CT Subsystem	Cost (£)
Structure	5,700 - 6,900 [18]
Payload	1,800 - 2,000 [3]
Power	76,783 [16]
Total Cost*	**84283 - 85693**

6 DISCUSSION(2)

To ensure the success of the mission, the spacecraft must be capable of meeting or exceeding the requirements and mission objectives set.

The mission's required lifetime is two years. Through simulation in STK of an orbit at 200 km of altitude around the moon with an inclination of 90°, it is estimated that the orbit will remain stable for a minimum of three years. The primary payload selected exceeds spatial resolution requirements, attaining 600 m resolution on ground at nadir. The CubeSat meets the mission objectives by being able to generate thermal images of the sunlit south pole between 85°-90°latitude. However, the unavailability of compact, off-the-shelf thermal imaging sensors means the limited spectral range of the Boson+ 320 (7-15 μm) renders the sensor unable to detect temperatures below 100 K, when areas of the surface are shaded. Due to unavailability of data within shadowed areas, the focus of the mission adapted to focus on thermal stability of the surface, especially around cold traps. These locations are of significance because of their adjacency to potential invaluable resources - ice water, and so remain to serve to overall habitability focus of the mission. During eclipse times, the transmit power drops to save energy, reducing the SNR and increasing BER. This trade-off prioritises power conservation over constant communication, this is done by transmitting data only when not in eclipse. The thermal control system can require up to 10 W during eclipse to keep components within operational ranges, which is a significant power draw. This was resolved by ensuring the solar arrays (43.26 W peak) and battery (90 Wh) can support this load. The issue of being unable to detect temperatures below 100 K remains unsolved, as there simply aren't any off-the-shelf components capable of detecting these temperatures in such a small and lightweight package. In future, with further development on making miniature infrared imagers, much lower temperatures could be sensed.

7 SUMMARY & CONCLUSION

In summary, POLARIS-CT aims to gain insight into the thermal characteristics of the Lunar South Pole, with a focus on generating comprehensive thermal maps and assessing temperature stability of the surface, especially around cold traps. The internal characteristics of the cold traps are undetectable by this mission, however their location may be inferred indirectly via the gaps in data, paired with existing topographic data. In regard to the payload, while it has met the mission requirements, it was not capable of directly characterising cold traps. However, considering the majority of COTS IR detectors possess limited spectral ranges, a detector with the ability to read below 100 Kelvin would likely come with significant trade-offs such as increased complexity, cost and mass, that would not be suitable given the NRDD constraints. The resolution could have been improved to develop a more comprehensive thermal mapping, which could be resolved with a lower orbit. This itself comes with inherent issues such as increased orbit perturbations, due to variations in the lunar gravitational field.

To conclude, the establishment of this mission and of POLARIS-CT has been an accomplished group effort in which all members cooperated and stayed in constant communication to ensure POLARIS-CT is a practical and achievable CubeSat mission. To further expand this project, computer simulations and studies should be conducted for the structural effect of the solar array deployment, alongside the effects of impact upon de-orbiting, such as regolith dispersion.

References

[1] "nanoracks.com," https://nanoracks.com/wp-content/uploads/Nanoracks-DoubleWide-Deployer-NRCSD-IDD.pdf, [Accessed 04-05-2025].

[2] "LCROSS - NASA Science — science.nasa.gov," https://science.nasa.gov/mission/lcross/, [Accessed 04-05-2025].

[3] FLIR Systems, "Boson+ Longwave Infrared Thermal Camera Core," 2025, accessed: 2025-04-29. [Online]. Available: https://www.flir.co.uk/products/boson-plus/?model=22320AS50-6IARXvertical=lwirsegment=oem

[4] Busek Co. Inc., "BGT-X5: Green Monopropellant Thruster," 2025, accessed: 2025-04-29. [Online]. Available: https://www.busek.com/bgtx5

[5] "Ladee in depth - nasa science — science.nasa.gov," https://science.nasa.gov/mission/ladee/in-depth/, [Accessed 04-05-2025].

[6] "RWA05 - Berlin Space Technologies — Reaction Wheel — satnow.com," https://www.satnow.com/products/reaction-wheels/berlin-space-technologies/38-1185-rwa05, [Accessed 08-05-2025].

[7] ISISPACE, "ISIS UHF Downlink / VHF Uplink Full Duplex Transceiver," 2025, accessed: 2025-04-29. [Online]. Available: https://isispace.nl/product/isis-uhf-downlink-vhf-uplink-full-duplex-transceiver

[8] ——, "Antenna System for 6U/12U CubeSats," 2025, accessed: 2025-04-29. [Online]. Available: https://isispace.nl/product/antenna-system-for-6u-12u-cubesats

[9] NASA, "Nasa - nssdca - spacecraft details," https://nssdc.gsfc.nasa.gov/nmc/spacecraft/display.action?id=QUEQIAO, [Accessed 08-05-2025].

[10] ISISPACE, "On-Board Computer for CubeSats," 2025, accessed: 2025-04-29. [Online]. Available: https://isispace.nl/product/on-board-computer

[11] "Small Satellite Solar Panels — isispace.nl," https://www.isispace.nl/product/isis-cubesat-solar-panels/, [Accessed 06-05-2025].

[12] ISISPACE, "Modular electric power system 2," https://www.isispace.nl/product/modular-electrical-power-system/, [Accessed 06-05-2025].

[13] MIT, "Temperature-dependent performance of electrospray thrusters with ascent monopropellant," https://dspace.mit.edu/bitstream/handle/1721.1/155453/IEPC-2024-782_Bruno.pdf?sequence=1isAllowed=y, [Accessed 06-05-2025].

[14] S. Mems, "nanossoc a60," https://solar-mems.com/space-equipment/nanossoc-a60/, [Accessed 06-05-2025].

[15] ISISPACE, "Isispace on-board computer (iobc)," https://www.isispace.nl/product/isispace-iobc/, [Accessed 06-05-2025].

[16] EnduroSat, "6U Deployable Solar Array," 2025, accessed: 2025-04-29. [Online]. Available: https://endurosat.com/products/6u-deployable-solar-array

[17] "Modular Electrical Power System 2 — isispace.nl," https://www.isispace.nl/product/modular-electrical-power-system/, [Accessed 05-05-2025].

[18] "Cubesat structures - aac clyde space — aac-clyde.space," https://www.aac-clyde.space/what-we-do/space-products-components/cubesat-structures, [Accessed 05-05-2025].