The 9th
Mission Idea Contest:
to the Moon

# Taiwan India Lunar Dust Analysis

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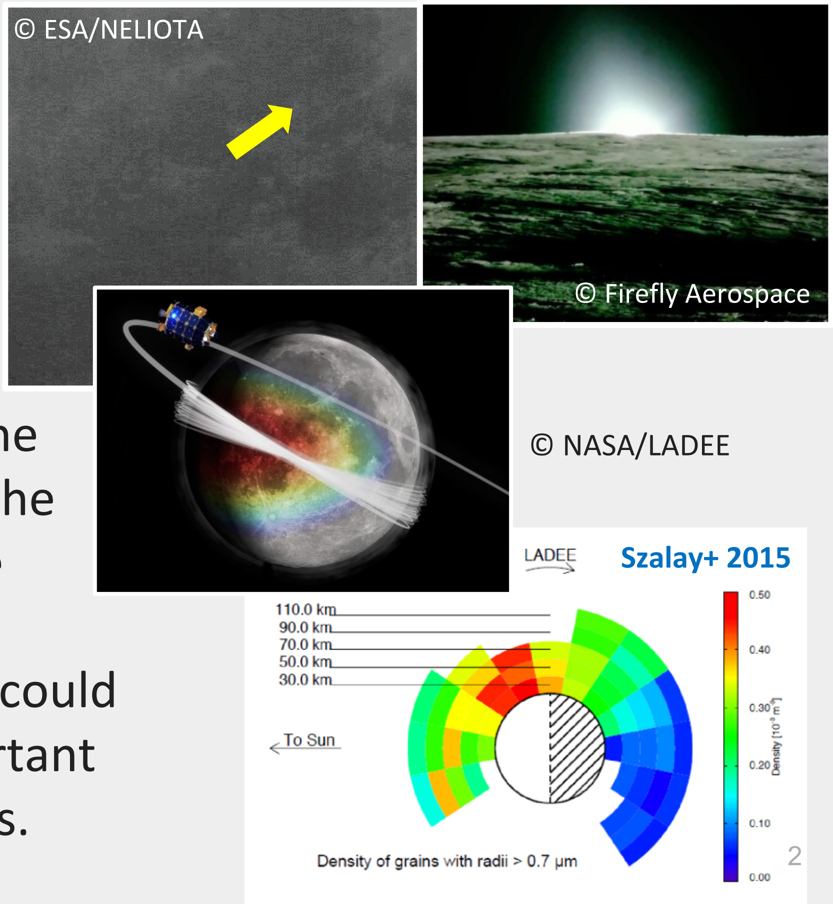
# Dust: Revealing the Secrets of Lunar Poles

- Meteoroid impacts are the key driver of:
  (1) volatile evolution in the PSRs;
  (2) Na exosphere.
- High-latitude meteoroid flux is crucial to understand the long-term evolution of volatiles in polar regions.
- NASA’s LADEE mission (2013): identified the dust ejecta cloud at ~20 - 100 km around the lunar equator ⇒ **dust in the vicinity of the polar region remains uninvestigated**!
- Levitation of surface particles within PSRs could affect in situ exploration and carries important implications for future lunar polar missions.



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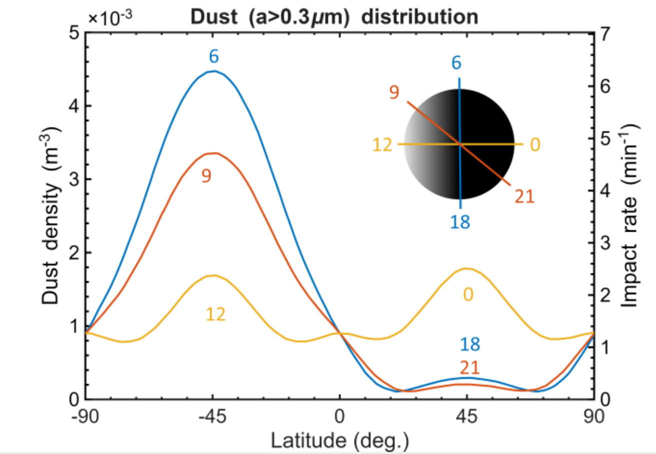
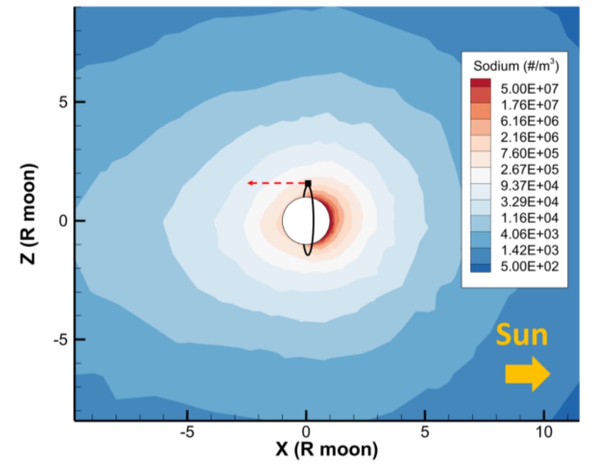
# TILDA Mission Objectives

- Identify the size, velocity and spatial distribution of the dust particles around the low-altitude polar region of the Moon.
- Monitor emissions from the lunar sodium exosphere, including its spatial variability and temporal changes under varying lunar phase conditions.
- Establish the possible correlation between lunar exosphere and dust environment.
- Demonstrate technology capabilities of CubeSats at low lunar orbits.





# TILDA CONOPS

**Science Mode**

Dust & Sodium Observation (nightside)

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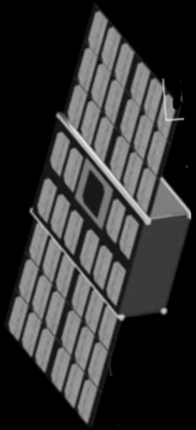
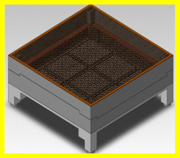
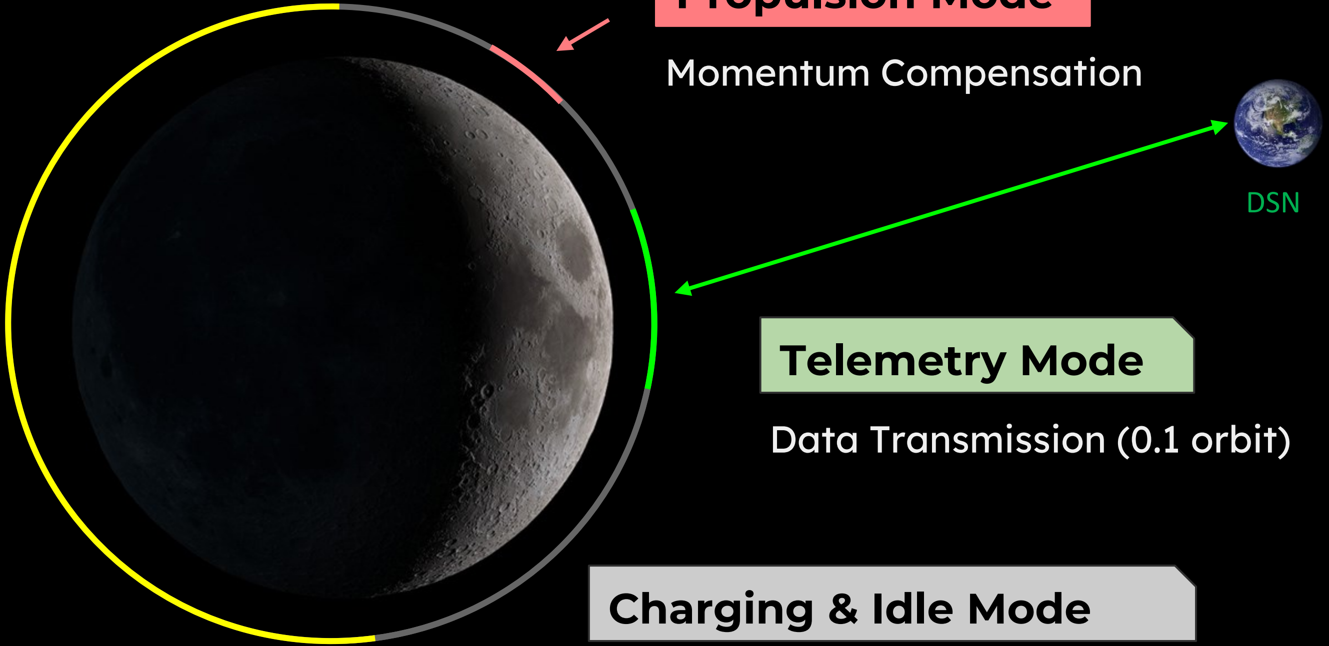




Piezo Dust Detector
(Wolf et al., 2012)



**Propulsion Mode**

Momentum Compensation

**Telemetry Mode**

Data Transmission (0.1 orbit)

**Charging & Idle Mode**

Battery Recharge & Health Monitoring (dayside)

# Key Performance Parameters

## I. The Dust Detector:

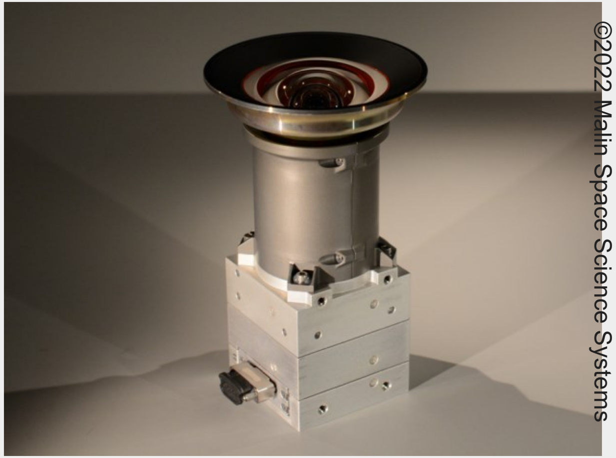
- Resolve particle sizes from micron to sub-millimeter (~1.0 μm – 0.5 mm)
- Achieve sensitivity sufficient to detect impact energies on the order of ~10 nJ.
- Be capable of high temporal resolution such as 0.1 ct/min.

## II. The Optical Camera:

- Be equipped with a narrow-band filter with a center wavelength of 589.3 nm and bandwidth 3.5 nm to detect Na emission D1 and D2 lines (589.6/589.0 nm).

©NASA/Goddard/Conceptual Image Lab

# Payload



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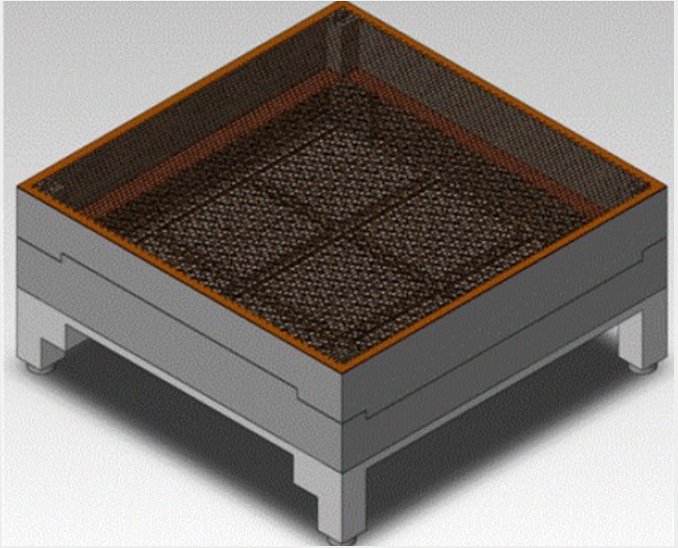
## MSSS ECAM-N50

- 350 – 850 nm (Mono)
- FOV: 44 x 56
- Mass: 700 - 950 g
- With a sodium narrow band filter (~3 nm)

PDD © Wolf et al., 2012.



Baylor U.
(USA)

## Piezo Dust Detector

- Sensor area covers ~ 2,600 $mm^2$
- Particle size: 1 µm to 1 mm (@10 km/s)
- Max measurable impact energy: 1 J

(Note: The related mission failed in LEO and no updated information of PDD. The team are searching for alternatives.)

# Spacecraft Design

- Orbit maintenance and deorbit mauver
- Diagonal thrust balancing
- External disturbance elimination capability
- Attitude control in 3-axis
- Clearest field of view for direct to earth communication
- Avoid direct sun exposure for sensitive image sensor

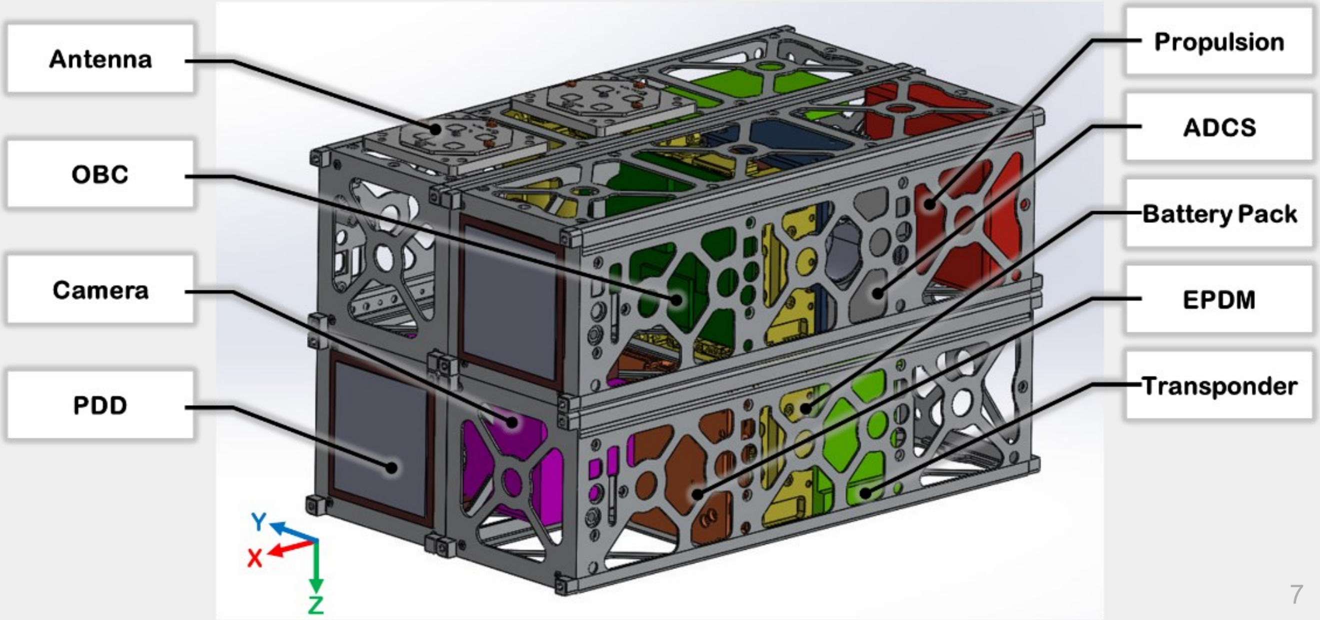
**Flight dir. (+X)**
- 2x PDD

**Anti- flight dir. (-X)**
- 2x Thruster

**Anti- nadir dir. (+Z)**
- 2x Antenna
- 1x Fine sun sensor
- 4x Solar panels (Hide)

**Cross track dir. (+Y)**
- 1x Camera

**Anti- cross track dir. (-Y)**
- 1x Star tracker



# Spacecraft Design

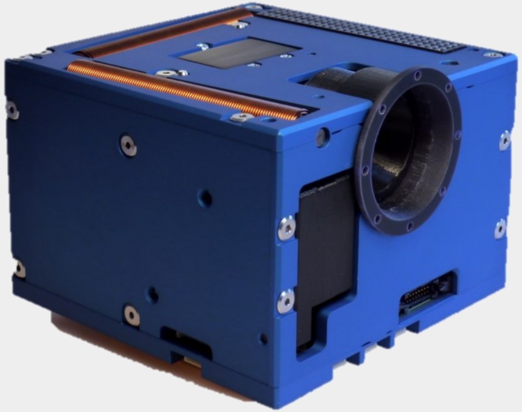
- TILDA is a 12U CubeSat with mass ~14 kg in total.

| Category | Product | QTY | Mass (kg) | Size (mm) | Total Mass (kg) |
|---|---|---|---|---|---|
| 12U Frame | 12U CubeSat Structure - SM12 | 1 | 1.750 | 340.5x226.3x226.3 | 1.750 |
| OBC | Proton400K | 1 | 0.165 | 91.44x127 | 0.165 |
| Transponder | Iris V2.2 SmallSat Deep Space Transponder | 1 | 0.875 | 100.5x101.0x56.0 | 0.875 |
| Antenna | S-band Antenna Wideband | 1 | 0.500 | 80x80x5 | 0.500 |
| LNA | Iris V2.2 SmallSat Deep Space Transponder | 1 | 0.080 | 114.3x46.0x15.5 | 0.080 |
| SSPA | | 1 | 0.150 | 102.9x55.7x24.4 | 0.150 |
| EPS | High-power Multi-channel EPSM1 | 1 | 0.230 | 96x90x15.24 | 0.230 |
| Solar Panels | 6U solar panel | 5 | 0.390 | - | 1.950 |
| Batteries | Battery Module 2 (BM2) | 4 | 0.710 | 100x100x48.4 | 2.840 |
| ADCS | iADCS400 | 1 | 1.700 | 95.4x95.9x67.3 | 1.700 |
| Propulsion | VACCO's Standard MiPS | 2 | 0.848 | 100x100x36 | 1.696 |
| PDD | Piezo Dust Detector | 2 | 0.500 | 80x80x40 | 1.000 |
| Camera | ECAM-N50 | 1 | 0.950 | 67x57x137 | 0.950 |
| | | | | **Total Mass (kg)** | **13.886** |

# C&DH & ADCS

**PROTON400K**

- Dual core microprocessor for redundancy
- High radiation tolerance (>100krad) for server lunar orbit



- 3x Reaction Wheels
- Star Tracker
- Fine Sun Sensor
- Gyro
- Integrated ADCS

**iADCS400**

- Three axes reaction wheel attitude control for various pointing requirement.
- Star tracker for accurate attitude determination.
- Fine sun sensor for attitude determination support.
- Gyro for quick attitude acquisition.
- Leverage satellite's thruster for wheel desaturation

# Propulsion

**Micro Propulsion System**

- Each propulsion system provides four thrusters to compensate undesired disturbance at any direction.

- Modular design to have a sufficient total impulse up to 515 N-sec per propulsion system, enough for deorbit operation.

- Low power requirement < 1W for communication and health monitoring, friendly for non-propulsion mode.

- Radiation shielding embedded for radiation resistance.

- Flight proven control software included.

- Flight proven (Mars mission) component and manufacturing method are used to increase the reliability.

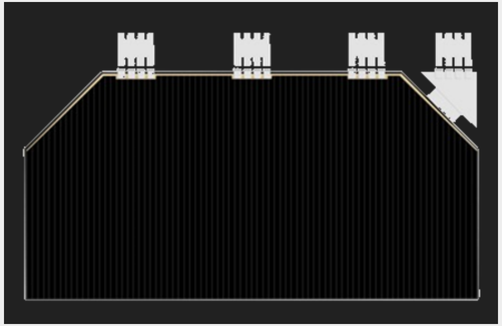
# EPS

**EPSM1**

- Radiation tolerance enhance for lunar orbit operation
- High energy transfer efficiency

**BM2**

- Embedded battery heater for dramatic temperature change.
- 168 W output (peak) for thruster.



- Consider End of life (EOL) degradation as 0.81.
- 60 solar cells are used for sufficient power generation.
- 1.21W power generation per cell

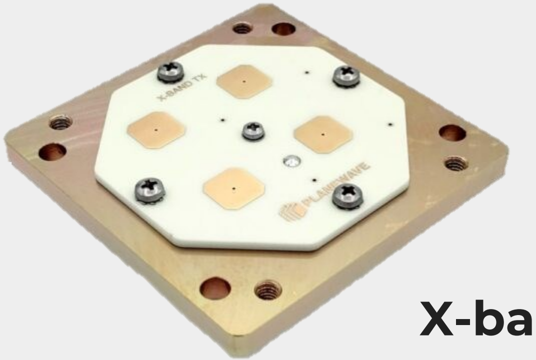
# EPS Budget

- The EPS can supply power to meet demands at all times.

| Mode | OBC | EPS | TTC | ADCS | PDD | Camera | Propulsion | Consumption (W) | Duty Cycle |
|---|---|---|---|---|---|---|---|---|---|
| Sys. Avg power (w) | 12 | 2 | 25 @Tx,<br>9 @Rx,<br>29 @all | 2 | 6 | 3.25 @obs, 1.25 @idle | 24 @full,<br>2 @idle | - | - |
| Propulsion | On | On | Off | On | Off | Off | On @full | 46 | 0.05/ 0.075 |
| Science (Camera) | On | On | On @Rx | On | On | On @all | On @idle | 41.25 | 0.15 |
| Telemetry | On | On | On @all | On | Off | On @idle | On @idle | 48.25 | 0.1 |
| Idle | On | On | On @Rx | On | Off | On @idle | On @idle | 28.25 | 0.15/ 0.275 |
| Safe | On | On | On @Rx | Off | Off | Off | Off | 23 | 1 |
| Charging | On | On | On @Rx | On | Off | On @idle | On @idle | 28.25 | 0.55 |
| **Power Generation/Orbit (W)** | | | | | | | **36.4** | | |
| **Power Consumption/Orbit (W)** | | | | | | | **33.09/31.58** | | |
| **Total Power /Orbit (W)** | | | | | | | **3.31/4.82** | | |

# TT&C

**IRIS V2.2**



**X-band Antenna**

- Had been use for lunar mission already.
- Design to be compatible with Deep Space Network (DSN).
- Radiation tolerance parts are use for deep space, muti-year mission.
- External Low Noise Amplifier (LNA) to amplify the uplink signal to reduce noise figure.
- High gain antenna to provide a reliable link.

# TT&C Budget: Uplink

| Ground Station: | | Uplink Path: | |
|---|---|---|---|
| Ground Station Transmitter Power Output: | 200.0 W | Ground Station Antenna Pointing Loss: | 0.4 dB |
| | | Gnd-to-S/C Antenna Polarization Losses: | 2.7 dB |
| Ground Stn. Total Transmission Line Losses: | 2.6 dB | Path Loss: | 221.8 dB |
| | | Atmospheric Losses: | 2.1 dB |
| Antenna Gain: | 72.8 dBi | Ionospheric Losses: | 0.4 dB |
| | | Rain Losses: | 0.0 dB |
| Ground Station EIRP: | 93.3 dBW | Isotropic Signal Level at Spacecraft: | -134.1 dBW |

| ------- Eb/No Method ------- | | |
|---|---|---|
| Spacecraft Antenna Pointing Loss: | 0.9 | dB |
| Spacecraft Antenna Gain: | 10.0 | dBi |
| Spacecraft Total Transmission Line Losses: | 0.8 | dB |
| Spacecraft Effective Noise Temperature: | 682 | K |
| Spacecraft Figure of Merit (G/T): | -19.1 | dB/K |
| S/C Signal-to-Noise Power Density (S/No): | 74.4 | dBHz |
| System Desired Data Rate: | 10000 | bps |
| Command System Eb/No: | 34.4 | dB |
| Demodulation Method Selected: | QPSK | |
| Forward Error Correction Coding Used: | None | |
| System Allowed or Specified Bit-Error-Rate: | 1.0E-05 | |
| Demodulator Implementation Loss: | 1.0 | dB |
| Telemetry System Required Eb/No: | 9.6 | dB |
| Eb/No Threshold: | 10.6 | dB |
| **System Link Margin:** | **23.8** | dB |

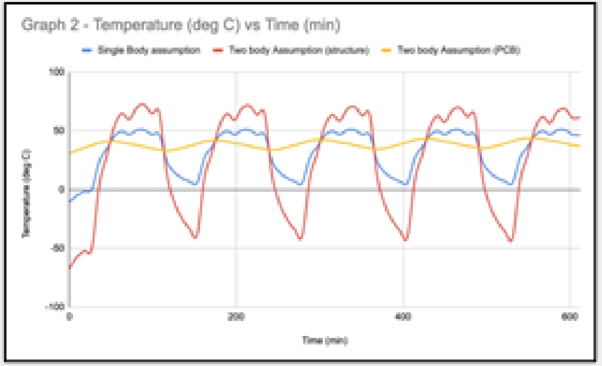
# TT&C Budget: Downlink

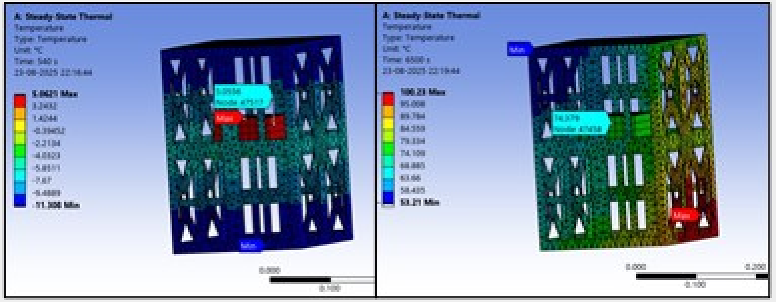
| Spacecraft: | | Downlink Path: | |
|---|---|---|---|
| Spacecraft Transmitter Power Output: | 3.8 W | Spacecraft Antenna Pointing Loss: | 0.9 dB |
| In dBW: | 5.8 dBW | S/C-to-Ground Antenna Polarization Loss: | 0.0 dB |
| In dBm: | 35.8 dBm | Path Loss: | 223.1 dB |
| Spacecraft Total Transmission Line Losses: | 0.4 dB | Atmospheric Loss: | 2.1 dB |
| Spacecraft Antenna Gain: | 10.0 dBi | Ionospheric Loss: | 0.8 dB |
| Spacecraft EIRP: | 15.4 dBW | Rain Loss: | 0.0 dB |
| | | Isotropic Signal Level at Ground Station: | -211.5 dBW |

| ------- Eb/No Method ------- | | |
|---|---|---|
| Ground Station Antenna Pointing Loss: | 0.6 | dB |
| Ground Station Antenna Gain: | 74.2 | dBi |
| Ground Station Total Transmission Line Losses: | 2.2 | dB |
| Ground Station Effective Noise Temperature: | 422 | K |
| Ground Station Figure of Merit (G/T): | 60.8 | dB/K |
| G.S. Signal-to-Noise Power Density (S/No): | 77.3 | dBHz |
| System Desired Data Rate: | 400000 | bps |
| Telemetry System Eb/No for the Downlink: | 21.3 | dB |
| Demodulation Method Selected: | QPSK | |
| Forward Error Correction Coding Used: | None | |
| System Allowed or Specified Bit-Error-Rate: | 1.0E-05 | |
| Demodulator Implementation Loss: | 0.5 | dB |
| Telemetry System Required Eb/No: | 9.6 | dB |
| Eb/No Threshold: | 10.1 | dB |
| **System Link Margin:** | **11.2** | **dB** |

# Thermal Analysis

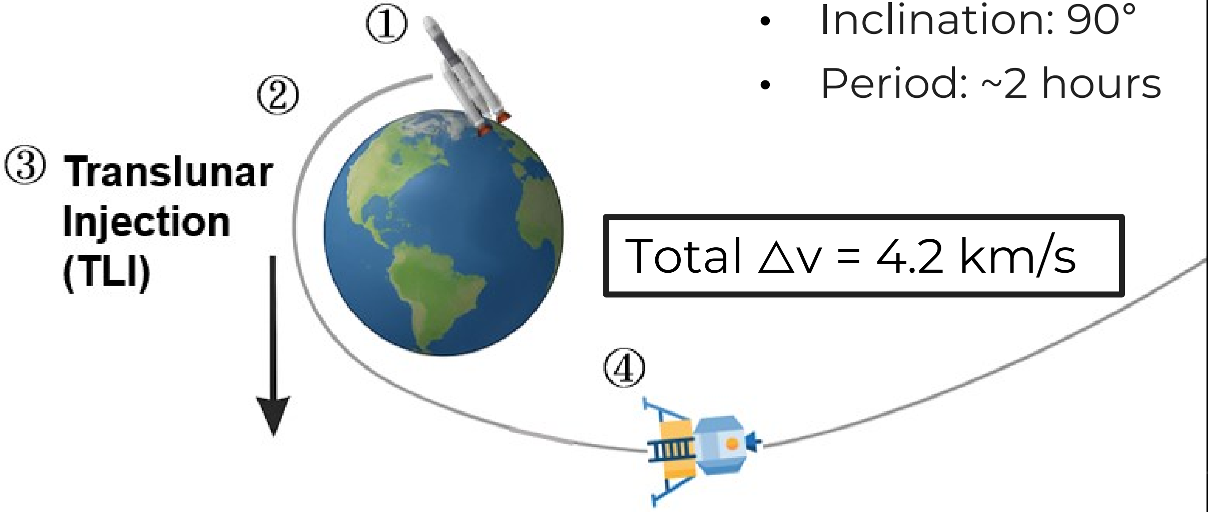
- Both simplified and detailed modeling show the s/c stays within survivable limits in sunlight and eclipse, with electronics in the 30 - 50 °C zone.
- Good conduction through the aluminum chassis and continuous avionics power smooth out temperature swings, protecting PCB assemblies from harsh external extremes.
- Passive thermal control is enough:
  Simple measures such as MLI patches and improved board-chassis interfaces can keep structural temperatures tidy without adding mass or complexity.





# Orbit

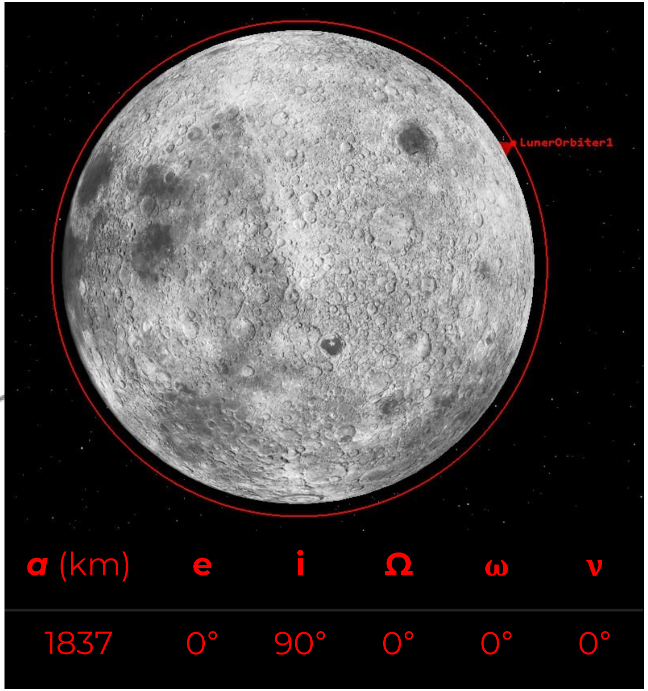
## I. E-M Transfer



Total Δv = 4.2 km/s

- Altitude: 100 km
- Inclination: 90°
- Period: ~2 hours

## II. Low Lunar Orbit

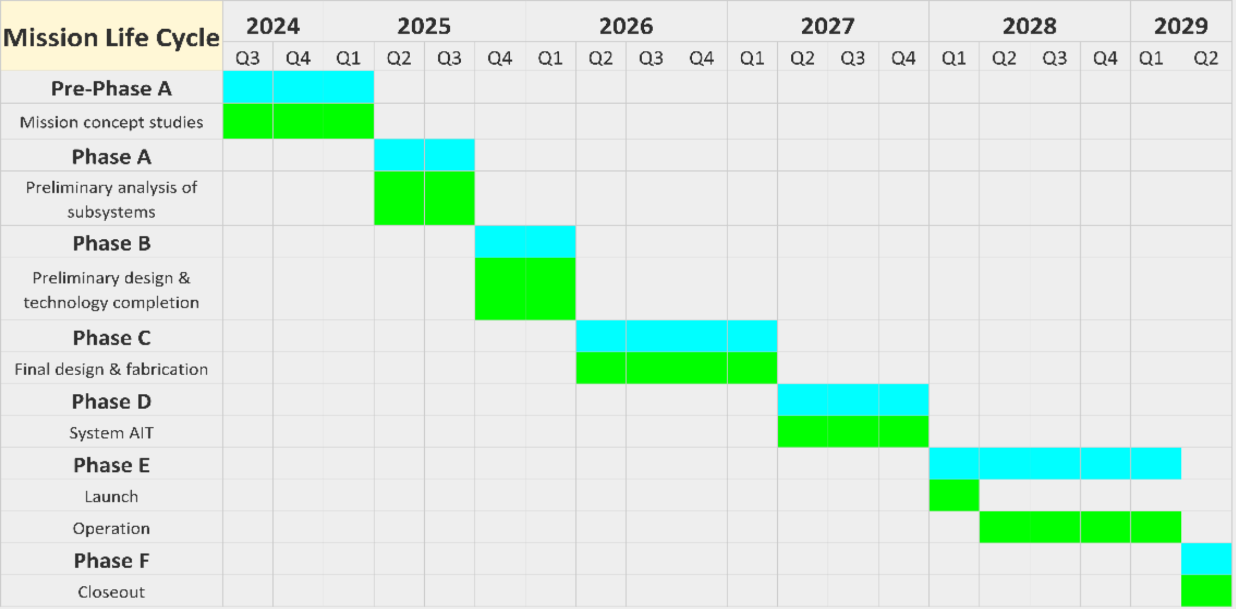


| $a$ (km) | $e$ | $i$ | $\Omega$ | $\omega$ | $\nu$ |
|---|---|---|---|---|---|
| 1837 | 0° | 90° | 0° | 0° | 0° |

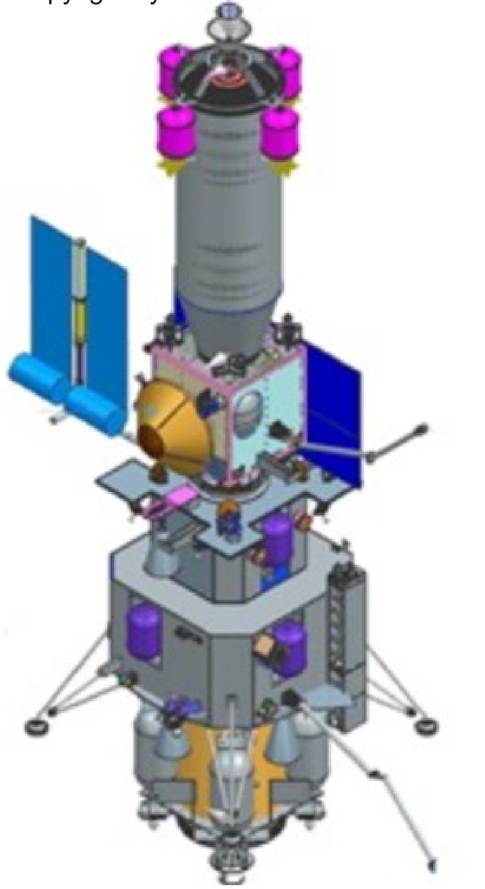
① **Launch Vehicle Takeoff**
② **Launch Vehicle & Spacecraft Separation**
③ **Translunar Injection (TLI) Maneuver**
④ **Translunar Cruise**
⑤ **Lunar Orbit Insertion Maneuver**
⑥ **Lunar Satellite Deployment**

# Implementation

- Taiwan-India Collaboration
- Estimated cost: $19 Million USD
- Chances of piggyback of Chandrayaan-4 (~2028)



| Mission Life Cycle | 2024 | | 2025 | | | | 2026 | | | | 2027 | | | | 2028 | | | | 2029 | |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | Q3 | Q4 | Q1 | Q2 | Q3 | Q4 | Q1 | Q2 | Q3 | Q4 | Q1 | Q2 | Q3 | Q4 | Q1 | Q2 | Q3 | Q4 | Q1 | Q2 |
| **Pre-Phase A** | 🟦 | 🟦 | 🟦 | | | | | | | | | | | | | | | | | |
| Mission concept studies | 🟩 | 🟩 | 🟩 | | | | | | | | | | | | | | | | | |
| **Phase A** | | | | 🟦 | 🟦 | | | | | | | | | | | | | | | |
| Preliminary analysis of subsystems | | | | 🟩 | 🟩 | | | | | | | | | | | | | | | |
| **Phase B** | | | | | | 🟦 | 🟦 | | | | | | | | | | | | | |
| Preliminary design & technology completion | | | | | | 🟩 | 🟩 | | | | | | | | | | | | | |
| **Phase C** | | | | | | | | 🟦 | 🟦 | 🟦 | 🟦 | | | | | | | | | |
| Final design & fabrication | | | | | | | | 🟩 | 🟩 | 🟩 | 🟩 | | | | | | | | | |
| **Phase D** | | | | | | | | | | | | 🟦 | 🟦 | 🟦 | | | | | | |
| System AIT | | | | | | | | | | | | 🟩 | 🟩 | 🟩 | | | | | | |
| **Phase E** | | | | | | | | | | | | | | | 🟦 | 🟦 | 🟦 | 🟦 | 🟦 | |
| Launch | | | | | | | | | | | | | | | 🟩 | | | | | |
| Operation | | | | | | | | | | | | | | | | 🟩 | 🟩 | 🟩 | 🟩 | |
| **Phase F** | | | | | | | | | | | | | | | | | | | | 🟦 |
| Closeout | | | | | | | | | | | | | | | | | | | | 🟩 |



**Chandrayaan-4**

# Risk Analysis

| Risk Item | Risk Level | Mitigation Action |
|---|---|---|
| Potential communication delays or data loss during lunar operations | Low | Develop robust communication protocols and redundant data storage systems |
| Dependence on governmental launch providers may lead to schedule delays | Low | Secure backup launch options and maintain flexibility in scheduling |
| Insufficient sensitivity and detection range of piezoelectric dust detector | Medium | Extensive testing and calibration during the development phase |
| Incapability of spacecraft subsystems withstanding the lunar environment | Medium | Rigorous environmental testing and use of proven, space-qualified components |
| Insufficient funding affecting project timelines and scope | High | Diversify funding sources and establish contingency budgets |

# Summary

## Polar Dust Science

TILDA maps dust size, speed and motion in low-altitude lunar polar orbits to decode meteoroid impacts and volatile evolution.

## Dust–Exosphere Connection

Dual sensing of dust impacts and sodium emissions uncovers how surface activity shapes the lunar exosphere.

## Miniaturized Tech Demonstrator

A 12U CubeSat with a piezoelectric dust detector and compact CMOS camera proves small spacecraft can deliver big science.

## Taiwan–India Partnership for Artemis Era

An international collaboration advancing lunar knowledge and technology that strengthens future exploration on the Moon.

# Reviewer's comments in pre-MIC9

Q: The section "orbit description" mentions that ionised lunar dust clouds are expected to cause attitude disturbances for the spacecraft. Is it possible to roughly quantify the expected perturbation? The attitude disturbance, in itself, could provide information on the dust clouds, if it is large enough.

A: Our simulation shows that the CubeSat will have -34 V surface potential with plasma conditions. ⇒ the total charge $Q = CV \sim 4.5\times10^{-10}$ C. Assume the magnetic field $B = 100$ nT ⇒ Lorentz force $F = QvB \sim 7\times10^{-14}$ N. CubeSat acceleration $a \sim 3.6\times10^{-15}$ m/s². If we consider the ambient electric field $E = 1$ mV/m, then $a \sim 2.3\times10^{-14}$ m/s². Compared to lunar gravity (1.3 m/s²) or solar radiation pressure perturbations ($10^{-8}$ m/s²), this is still 6 – 14 orders of magnitude smaller.