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## Research Article

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# Experimental Surface Characterization and MLI-Based Thermal Control Enhancement for a Small Earth Observation Satellite

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## Abstract

Passive thermal control is critical for small satellites operating under strict power, mass, and volume constraints. This paper presents a measurement-driven passive thermal control methodology for the LEOS-50MR remote sensing satellite in low earth orbit (LEO). A comprehensive thermal mathematical model developed in Airbus Systema Thermica<sup>®</sup> is first verified against baseline operational data within **3.7%** accuracy. To eliminate material property assumptions, the thermo-optical properties of an ECSS compliant black-anodized aluminum panel are experimentally characterized using an ET10 reflectometer and integrated into the model, improving physical fidelity by 1–2 K. Subsequently, Multi-Layer Insulation (MLI) is applied to the bottom panel to minimize radiative heat losses. The enhanced configuration increases the cold case minimum temperatures by approximately 11–13 K relative to the baseline model without exceeding the maximum operational limits during high power imaging scenarios. Combining experimental property characterization with passive MLI design offers a highly effective, verified framework for enhancing small satellite thermal margins.

**Keywords:** LEOS-50MR, Thermal mathematical modeling, Systema Thermica<sup>®</sup>, MLI, Thermo-optical measurement, Earth observation satellite.

## 1 Introduction

Thermal control subsystem (TCS) is one of the essential subsystems in spacecraft design because it directly affects mission reliability, subsystem survivability, and operational lifetime. Its main function is to maintain all spacecraft components within their allowable temperature limits during different mission phases. A spacecraft operates efficiently and achieves a longer service life when its equipment remains within the specified qualification and operational temperature limits [1]. For satellites operating in low Earth orbit (LEO), the thermal environment is highly variable due to the combined effects of direct solar radiation, Earth albedo, Earth infrared radiation, internal heat dissipation, and repeated transitions between sunlight and eclipse. These effects are more critical in small satellites because of their low thermal mass, limited external surface area, restricted internal volume, limited electrical power, and high equipment power density. Therefore, accurate thermal modelling and proper thermal control design are required to ensure that all spacecraft subsystems remain within their acceptable thermal limits.

Passive thermal control is generally preferred for small satellite missions because it provides high reliability, low complexity, low mass, and no continuous power consumption. Passive techniques include thermal coatings, radiators, conductive interfaces, thermal tapes, thermal straps, phase change materials, and multi-layer insulation (MLI). However, the performance of passive thermal control depends strongly on the accuracy of the thermal mathematical model and on the thermo-physical and thermo-optical properties assigned to the spacecraft materials and surfaces. In particular, the solar absorptivity and infrared emissivity of external surfaces control the balance between absorbed environmental heat and radiated heat loss to space. If these properties are assumed from handbooks or previous missions without experimental confirmation, the thermal model may not accurately represent the actual manufactured spacecraft surfaces.

Several studies have investigated passive thermal control techniques for small satellites. Berggren (2015) [2] studied the thermal control design of the MIST spacecraft using passive elements such as coatings and MLI. Corpino et al. (2015) [3] analysed the thermal design of a nanosatellite in LEO using coatings, MLI, and thermal tapes, and showed that passive control can maintain operational temperatures, although heaters may still be required for the battery under cold conditions. Sakraker et al. (2015) [4] presented the QARMAN CubeSat atmospheric entry experiment, where passive coatings and variable emissivity thermal control concepts were considered. Escobar et al. (2016) [5] applied a genetic algorithm to optimize the distribution of passive thermal control coatings on CubeSat surfaces and showed that optimized coating layouts can reduce temperature levels compared with non optimized surfaces. Tuttle et al. (2017) [6] investigated advanced thermal control technologies for nanosatellites, including pyrolytic graphite thermal straps, thermal switches, and thermoelectric modules.

Garzón and Villanueva (2018) [7] developed a CubeSat thermal prediction model and showed that simplified early stage temperature estimates can support the selection and assessment of passive thermal control measures. Wallace et al. (2018) [8] performed thermal modelling of the EIRSAT-1 CubeSat and confirmed that passive coatings and MLI can significantly reduce component temperatures, while the battery remained one of the most critical subsystems. Athanasopoulos and Siakavellas (2018) [9] investigated variable emissivity multilayer patterned surfaces for passive thermal control of nanosatellites.

Further passive thermal control studies continued to confirm the importance of coatings, insulation, and structural heat transfer paths. Bonnici et al. (2019) [10] developed analytical and numerical thermal models for the UoMBSat-1 PocketQube and highlighted the influence of beta angle and surface coatings on thermal behaviour. Mesforoush et al. (2019) [11] experimentally and numerically investigated thin film MLI for satellite applications and showed that the actual MLI configuration has a significant effect on thermal resistance and effective emittance. Yang et al. (2019) [12] designed and validated a quasi-all-passive thermal control system for the LuoJia-1-01 satellite using MLI, isothermal-cabin methods, and limited active heaters. El Hefnawy (2019) [13] analysed the passive thermal control design of the ESEO satellite using Thermal Desktop, demonstrating the importance of numerical modelling in evaluating passive thermal solutions for small satellites. Athanasopoulos et al. (2019) [14] investigated smart variable emissivity surfaces for nanosatellites and showed their potential to improve thermal stability and reduce heater power demand. Sundu and Döner (2020) [15] modelled the thermal control of a LEO observation satellite using Systema Thermica<sup>®</sup> and showed that radiator placement is an important design variable in remote sensing spacecraft.

Recent studies have expanded passive thermal control toward more advanced materials and mission specific solutions. Bulut (2021) [16] presented the thermal design, analysis, and testing of the TURKSAT-3USAT CubeSat using passive materials such as Kapton and Mylar with limited active support. Elweteedy et al. (2021) [17] analysed the operational thermal behaviour of the ESEO satellite and showed that mission phase, internal heat dissipation, and orbital conditions strongly influence passive thermal control performance. Elhefnawy et al. (2021) [18] studied the post mission phase of ESEO and showed that passive thermal control improvement can be achieved through internal layout modification, coatings, and MLI. Farag and Elfarran (2021) [19] studied the passive thermal control design of a sun synchronous LEO micro satellite using Thermal Desktop/SINDA-Fluint, highlighting the role of numerical analysis in evaluating spacecraft thermal margins. Elhefnawy et al. (2022) [20] improved the ESEO passive thermal design by modifying the internal layout and applying suitable coatings and MLI. Dunwoody et al. (2022) [21] demonstrated the importance of thermal vacuum testing for validating CubeSat thermal models and identifying system level issues that may not appear during ambient testing. Tachikawa et al. (2022) [22] reviewed advanced passive thermal control materials and devices, including variable emissivity surfaces, MEMS-based systems, and deployable radiators. Zhou et al. (2023) [23] validated the thermal design of the JZJ-5 CubeSat using finite-element modelling and on orbit data, demonstrating the importance of correlating numerical predictions with

operational measurements. Elshaer et al. (2023) [24] investigated a phase change material thermal energy storage panel for small satellites in LEO and showed that thermal storage can reduce extreme temperature variations. Chen et al. (2023) [25] designed and analysed a micro/nano satellite thermal control system using MLI, white paint coatings, and insulation. Sirca et al. (2025) [28] reviewed spacecraft thermal control methods and confirmed that passive methods remain the main solution for small satellites constrained by mass, volume, and power. Barghandan and Mahmoodi (2026) [29] investigated solid–solid phase change material heat sinks with optimized fins for satellite avionics, showing that passive thermal storage can reduce peak electronic temperatures. Zhou et al. (2026) [30] proposed a rapid thermal radiation analysis method for a micro hyperspectral imager in inclined LEO and linked the thermal design to payload thermo-optical performance.

In parallel with thermal control hardware development, accurate thermal modelling has become a major research direction. Thermal modelling allows the prediction of spacecraft temperature response under hot and cold orbital conditions, different attitude modes, eclipse periods, and time dependent internal heat dissipations. Berggren (2015) [2] combined analytical analysis, numerical simulation, conductance measurement, and MLI modelling to support the thermal design of the MIST spacecraft. Jacques (2016) [31] proposed reduced finite element modelling techniques for spacecraft thermal analysis to reduce computational cost while preserving conductive radiative behaviour. Garzón and Villanueva (2018) [7] developed a thermal prediction model for the Libertad 2 CubeSat using ESATAN-TMS. Boushon (2018) [32] investigated how increasing model complexity affects temperature prediction in small satellites, demonstrating the importance of iterative modelling and validation of thermal assumptions. Wallace et al. (2018) [8] used Thermal Desktop for steady state thermal analysis of EIRSAT-1 and demonstrated the importance of modelling passive coatings and MLI. Kanumuru (2019) [33] developed the PyTherm tool as a low cost and rapid thermal management design system for CubeSat applications and compared it with Thermal Desktop. Bonnici et al. (2019) [10] combined analytical and numerical methods to predict PocketQube thermal behaviour. Gadalla et al. (2019) [34] developed a computational thermal model for a rotating spacecraft exposed to solar radiation and solved it using a meshless method. El Hefnawy (2019) [13] designed and analysed the passive thermal control system of the ESEO satellite using Thermal Desktop and compared the results with ESATAN-TMS. Sundu and Döner (2020) [15] used Systema Thermica<sup>®</sup> to analyse a LEO observation satellite, confirming the suitability of high fidelity commercial tools for remote sensing spacecraft thermal modelling.

Recent modelling studies have emphasized the importance of validation, testing, and correlation with measured or operational data. Park et al. (2021) [35], Dunwoody et al. (2022) [21], Zhou et al. (2023) [23], Buitrago-Leiva et al. (2024) [26], Song et al. (2025) [36], and Huang and Bu (2025) [27] demonstrated the use of on orbit data, thermal vacuum testing, or thermal balance testing to validate spacecraft thermal models. Elweteedy et al. (2021) [17] and Elhefnawy et al. (2022) [20] showed that detailed thermal modelling supports operational analysis, redesign, and performance improvement. Bai et al. (2024) [37], Liu et al. (2025) [38], and Ali et al. (2025) [39] proposed equivalent or experimentally verified thermal testing and modelling approaches

for small satellites. More recently, Bai et al. (2026) [40] and Zhou et al. (2026) [30] linked thermal modelling with mission verification and payload thermal performance assessment.

From the reviewed studies, three main conclusions can be identified. First, passive thermal control remains the preferred baseline solution for small satellites because it satisfies the strict limitations of mass, volume, power, and reliability. Second, active thermal control is introduced selectively when passive techniques cannot provide sufficient protection for thermally sensitive or high power components. Third, accurate thermal modelling and experimental validation are essential for reliable temperature prediction and thermal margin assessment. However, many published studies still rely on assumed thermo-optical properties taken from literature, databases, or previous missions. This introduces uncertainty because the actual absorptivity and emissivity of treated spacecraft surfaces depend on the material, coating process, surface preparation, anodizing quality, contamination level, and measurement conditions. This limitation is particularly important for small Earth observation satellites, where external surfaces, solar exposure, eclipse cooling, and mission dependent power dissipation interact strongly. The reviewed literature also shows that although thermal control technologies are becoming more diverse, including adaptive radiators, two phase systems, mission level thermal verification tools, and specialized payload thermal strategies, the quality of the final thermal prediction still depends strongly on the realism of the physical inputs and the degree of experimental or operational validation [40–42].

Accordingly, there is a need for tighter integration between experimental surface property characterization and detailed spacecraft thermal modelling. The present work addresses this gap by developing and verifying a measurement informed passive thermal control methodology for a LEO Earth observation small satellite. The LEOS-50MR platform is selected as the case study. A detailed thermal mathematical model is developed using Airbus Systema Thermica<sup>®</sup>, including the spacecraft geometry, material properties, surface thermo-optical properties, conductive and radiative couplings, environmental heat fluxes, orbit definition, attitude modes, and time dependent internal heat dissipation. The model is first verified against a reference case study under the imaging and post processing operational scenario. Then, the thermo-optical properties of a representative black anodized aluminium panel are experimentally measured and incorporated into the model. Finally, MLI is applied to the spacecraft bottom panel using experimentally measured emissivity to reduce radiative heat losses and improve thermal stability, especially under cold eclipse conditions.

## 2 THERMAL ENVIRONMENTAL EFFECTS ON A SPACE SATELLITE

Low Earth Orbit (LEO) satellites operate at altitudes ranging from approximately 500 to 1500 km and are exposed to multiple external thermal loads. These include heat flux incident from the Sun, the flux from Earth, and the reflection of solar rays from Earth, commonly known as albedo.

The Earth's elliptical orbit causes seasonal variations in solar flux. The maximum solar flux, approximately  $1414 \pm 5 \text{ W/m}^2$ , occurs during the northern hemisphere's winter solstice when the Earth is closest to the Sun. Conversely, the minimum value, around  $1317 \pm 5 \text{ W/m}^2$ , is observed during the summer solstice when the Earth is at its farthest point from the Sun. Albedo flux refers to the portion of solar radiation that is reflected by the Earth's surface back into space. Its nominal value is approximately  $30\% \pm 10\%$  of the incident solar flux, corresponding to an average of  $237 \pm 21 \text{ W/m}^2$ . However, this value is not fixed and varies with the Earth's surface characteristics. A portion of the solar flux absorbed by the Earth is re emitted as longwave infrared (IR) radiation, resulting from the Earth's thermal emission. The Earth is commonly approximated as a black body with an effective blackbody temperature of approximately  $255 \text{ K}$  [43].

## 2.1 Mathematical Modelling of a Satellite

The thermal response of a spacecraft can be formulated from the energy conservation equation, which is represented mathematically by a second order partial differential equation (PDE) [44]. Under orbital vacuum conditions, thermal energy is redistributed within the spacecraft primarily through solid conduction across physical interfaces and radiative exchange among its surfaces. For numerical analysis, the spacecraft is consequently divided into a network of discrete thermal nodes. The temperature evolution of each node is determined by its thermal interactions with neighboring nodes together with the external and internal heat loads acting on it.

The absorbed direct solar heat load is expressed as:

$$\dot{Q}_S = \alpha_s A_s F q_{\text{solar}} \quad (1)$$

The absorbed albedo heat load is given by:

$$\dot{Q}_A = \alpha_s A_s F q_{\text{albedo}} \quad (2)$$

The absorbed Earth infrared heat load is expressed as:

$$\dot{Q}_{E,\text{IR}} = \sigma T_E^4 F A_s \varepsilon_{\text{IR}} \quad (3)$$

In Eqs. (1)–(3),  $\alpha_s$  denotes the solar absorptivity [-] and  $A_s$  is the exposed surface area [ $\text{m}^2$ ]. The geometric relationship associated with the incident radiation is represented by the view factor  $F$  [-]. The quantities  $q_{\text{solar}}$  and  $q_{\text{albedo}}$  denote the direct solar and Earth reflected solar fluxes, respectively [ $\text{W/m}^2$ ]. The parameter  $\varepsilon_{\text{IR}}$  represents the infrared absorptivity [-],  $T_E$  is the effective Earth temperature [ $\text{K}$ ], and  $\sigma$  denotes the Stefan–Boltzmann constant [ $\text{W m}^{-2} \text{K}^{-4}$ ].

For node  $i$ , the transient thermal energy balance can be written as:

$$C_i \frac{dT_i}{dt} = \sigma \sum_{j=1}^N GR_{ij} (T_j^4 - T_i^4) + \sum_{j=1}^N GL_{ij} (T_j - T_i) + \dot{Q}_S + \dot{Q}_A + \dot{Q}_{E,\text{IR}} + \dot{Q}_{\text{dissipated}} \quad (4)$$

In Eq. (4),  $C_i$  denotes the thermal capacitance of node  $i$  [J/K], while  $T_i$  and  $T_j$  represent the temperatures of nodes  $i$  and  $j$  [K], respectively. The total number of nodes in the thermal network is denoted by  $N$ . Radiative interaction between nodes  $i$  and  $j$  is characterized by  $GR_{ij}$  [m<sup>2</sup>], whereas  $GL_{ij}$  [W/K] describes their conductive coupling. The remaining terms account for the absorbed direct solar load  $\dot{Q}_S$ , the absorbed albedo load  $\dot{Q}_A$ , the absorbed Earth infrared load  $\dot{Q}_{E,IR}$ , and the internally generated heat  $\dot{Q}_{\text{dissipated}}$ , all expressed in watts.

### 3 Satellite's Thermal Control Subsystem Modelling

The LEOS-50MR platform satellite is adopted as the reference spacecraft for the present investigation [45]. It is a small Earth observation satellite designed for operation in low Earth orbit (LEO), with a passive thermal control subsystem originally developed by Berlin Space Technologies (BST).

The numerical thermal representation is constructed progressively, beginning with the spacecraft geometry and followed by the definition of the thermo-physical and thermo-optical characteristics of its constituent materials and surfaces. Structural panels, solar arrays, and onboard units are included together with their thermal interfaces and equipment heat dissipation. The completed spacecraft model is then subjected to the specified orbital and attitude conditions so that the thermal loads associated with the relevant mission scenarios can be reproduced.

Airbus Systema Thermica<sup>®</sup> is employed to perform the thermal analysis. The modelling procedure combines the geometrical and thermal representations of the spacecraft, nodal discretization, conductive connections, radiative interactions, orbital trajectory, spacecraft attitude, and mission dependent operating conditions. The predicted temperature histories and extreme values are subsequently examined against the applicable design limits to evaluate the thermal behaviour of the spacecraft. For the LEOS-50MR platform, the overall allowable operating temperature interval considered in this study is 258–308 K.

#### 3.1 Geometry Modeling and Methodology

The LEOS-50MR platform has a compact cuboid configuration in which six primary aluminium alloy panels form the spacecraft bus. These consist of a bottom plate, a top plate, and four orthogonal sidewalls, while an enclosed payload compartment is mounted on the +Z face. The spacecraft incorporates six solar panels. Three panels are body mounted on the  $\pm X$  and  $-Z$  faces, and another three are installed on the upper payload hat structure. This configuration enables full surface solar energy collection without requiring deployable mechanisms. The internal equipment, including the power system, command data handling (CDH), attitude determination control system (ADCS), communications units, and dual camera payload, is accommodated in dedicated enclosures with thermal connections established through their mounting interfaces.

The thermal geometry is established from the detailed spacecraft CAD representation and implemented in Airbus Systema Thermica<sup>®</sup> as a Geometrical Mathematical

Model (GMM). The principal structural panels, solar arrays, and equipment housings are represented by shell elements, with equivalent thicknesses assigned to retain the appropriate thermal mass and radiative characteristics of the corresponding hardware. Surface representations are used for thin walled components, whereas the ribbed retaining structures are explicitly maintained so that their influence on the radiative view factors is captured.

The spacecraft fixed coordinate frame follows the right-handed convention shown in Figure 1, where the positive  $X$ ,  $Y$ , and  $Z$  directions are defined according to the LEOS-50MR reference system. The GMM is subsequently developed into the Thermal Mathematical Model (TMM) through nodal discretization and definition of the required heat transfer paths. The complete spacecraft representation contains 774 thermal nodes, distributed among the individual subsystems as summarized in Table 1. Conductive heat transfer between components is represented through their mechanical interfaces, while radiative interactions among model surfaces are determined using the Monte Carlo ray tracing capability of Systema Thermica<sup>®</sup>. This thermal network is used to predict the spacecraft response under the considered sun tracking and nadir pointing operational modes.

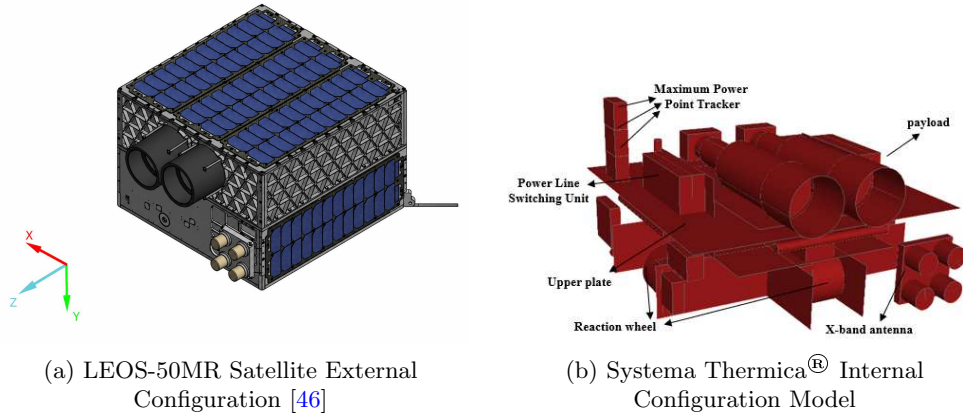


**Fig. 1:** LEOS-50MR satellite external and internal configuration

### 3.2 Thermo-Physical/Optical properties

Reliable prediction of the spacecraft thermal response requires representative material and surface properties to be assigned throughout the numerical model. The principal thermo-physical parameters adopted for the LEOS-50MR components are presented in Table 2. Thermal conductivity ( $k$ ) determines the rate at which heat is transported through a material, whereas specific heat capacity ( $C_p$ ) and density ( $\rho$ ) influence its thermal inertia and, consequently, its transient temperature response. Surface radiative behaviour is characterized separately using the thermo-optical properties summarized in Table 3. Solar absorptivity ( $\alpha$ ) specifies the fraction of incident

**Table 1:** Summary of thermal nodes per subsystem in the LEOS-50MR platform satellite TMM [46]

| Subsystem                                                                                                     | Number of nodes |
|---------------------------------------------------------------------------------------------------------------|-----------------|
| Structure                                                                                                     | 307             |
| Power                                                                                                         | 208             |
| Command/Data Handling (CDH)                                                                                   | 26              |
| Attitude Determination and Control Subsystem (ADCS)                                                           | 72              |
| Communications (COMS)                                                                                         | 36              |
| Payload (Dual Camera Payload (DCP), Payload Processing and Data Handling (PPDH), Payload Support Unit (PLSU)) | 125             |
| <b>Total</b>                                                                                                  | <b>774</b>      |

solar and Earth-reflected radiation absorbed by a surface, while infrared emissivity ( $\varepsilon$ ) determines its ability to exchange thermal radiation with the surrounding environment. These properties directly influence the balance between environmental heat absorption and radiative heat rejection to space. Their appropriate representation is therefore important when determining component temperatures and available thermal margins, particularly when evaluating passive thermal control treatments such as surface coatings and multi-layer insulation.

**Table 2:** Thermo-physical properties of main element groups in the thermal model [46]

| Element group                                       | Material        | $k$ [W/m·K] | $C_p$ [J/kg·K] | $\rho$ [kg/m <sup>3</sup> ] |
|-----------------------------------------------------|-----------------|-------------|----------------|-----------------------------|
| Primary structure panels                            | AL-6082 T6      | 180         | 896            | 2700                        |
| Retaining walls, mounts, equipment baseplates       | AL-6082 T6      | 180         | 896            | 2700                        |
| Electronics housings (PCU, CDH, COMS, payload base) | Aluminium       | 236         | 888            | 2700                        |
| Antenna bases and brackets                          | AL-6061         | 180         | 897            | 2700                        |
| GPS box and selected housings                       | AL-7075         | 145         | 862            | 2800                        |
| Solar panel substrates                              | CFK laminate    | 17 / 1.5    | 710            | 1550                        |
| Mechanical fasteners and stiffeners                 | Stainless steel | 15          | 500            | 7900                        |
| Thermal isolation inserts / spacers                 | PEEK            | 0.25        | 1300           | 1320                        |

### 3.3 Internal Heat Dissipation

The heat generated by the onboard equipment varies with the operating mode assigned to each thermal scenario. Scenario S01B represents the cold case, in which the spacecraft is subjected to minimum power operation with empty batteries and consequently

**Table 3:** Thermo-optical properties of main coatings used in the LEOS-50MR platform satellite thermal model [46]

| Material in BOM       | Material in model  | $\varepsilon$ | $\alpha$ | Remarks                                          |
|-----------------------|--------------------|---------------|----------|--------------------------------------------------|
| Aluminium, blank      | Al Sandblasted     | 0.20          | 0.40     | Bare/sandblasted-aluminium                       |
| CFK blank-Si cells    | Solar Panels       | 0.86          | 0.4795   | Solar cell surface properties                    |
| Kepla coat black      | Kepla Black        | 0.65          | 0.95     | Black coating on selected parts                  |
| Black anodized        | Black Anodize      | 0.88          | 0.88     | Default-black-anodized aluminium surfaces        |
| Partially-black anod. | Black Anod Partial | 0.70          | 0.70     | 80% coverage of black anodize                    |
| Black painted         | Black Paint        | 0.88          | 0.96     | High absorptivity black paint                    |
| White Z93             | White paint        | 0.92          | 0.09     | High- $\varepsilon$ /low- $\alpha$ thermal paint |
| Alodine 1200          | Alodine 1200       | 0.068         | 0.39     | Light conversion / passivation coating           |

low internal heat generation. As summarized in Table 4, the combined equipment dissipation is approximately 10.8 W during the sunlight period. During eclipse, this value falls to about 2.0 W because several units associated with power generation are no longer active.

Scenario S03, on the other hand, represents the imaging and post processing mission sequence and therefore imposes considerably higher internal thermal loads. Activation of the payload together with its supporting subsystems increases the total dissipated power to approximately 94.2 W during the active sun phase. As reported in Table 5, the corresponding totals are 93.2 W during imaging and 94.2 W during post processing. During the active eclipse period, the dissipation decreases substantially to approximately 8.9 W. Including these mission dependent power profiles in the thermal model allows the transient spacecraft temperatures to be evaluated under both the low dissipation cold condition and the high power operational condition.

**Table 4:** Component heat dissipations for Scenario S01B (Cold – Empty Batteries) [46]

| Component                              | Sunlight [W] | Eclipse [W] |
|----------------------------------------|--------------|-------------|
| Battery                                | 0.075        | 0.075       |
| Power Control Unit (PCU)–DigMod        | 0.62         | 0.62        |
| Maximum Power Point Tracker (MPPT) 1   | 0.80         | 0           |
| MPPT 3                                 | 0.80         | 0           |
| MPPT 4                                 | 2.60         | 0           |
| MPPT 5                                 | 2.60         | 0           |
| MPPT 6                                 | 2.60         | 0           |
| CDH–OBC                                | 0.32         | 0.32        |
| Ultra-High Frequency (UHF) Transceiver | 0.35         | 1.00        |
| <b>Worst case summation</b>            | <b>10.8</b>  | <b>2.0</b>  |

**Table 5:** Component heat dissipations for Scenario S03 (Imaging and Post Processing) [46]

| Component                   | Active Sun [W] | Active Eclipse [W] | Imaging [W]         | Post-Processing [W] |
|-----------------------------|----------------|--------------------|---------------------|---------------------|
| Battery                     | 76.075         | 0.075              | 6.975               | 40.675              |
| PCU-DCDC                    | 0.80           | 0.82               | 2.02                | 0.82                |
| PCU-InMod                   | 0.00           | 0.00               | 1.20                | 0.50                |
| PCU-DigMod                  | 0.30           | 0.30               | 0.30                | 0.30                |
| MPPT 1                      | 0.8            | 0                  | *T                  | 0.8                 |
| MPPT 2                      | 0              | 0                  | 0                   | 0                   |
| MPPT 3                      | 0.8            | 0                  | *T                  | 0.8                 |
| MPPT 4                      | 2.6            | 0                  | *T                  | 2.6                 |
| MPPT 5                      | 2.6            | 0                  | *T                  | 2.6                 |
| MPPT 6                      | 2.6            | 0                  | *T                  | 2.6                 |
| CDH-Acc Dig                 | 0.38           | 0.38               | 0.38                | 0.38                |
| CDH-OBC                     | 0.32           | 0.32               | 0.32                | 0.32                |
| CDH-HP OBC                  | 0.73           | 0.73               | 0.73                | 0.73                |
| Reaction Wheels (1-3)       | 1.80           | 1.767              | 5.18                | 1.77                |
| GPS Box                     | 0.00           | 0.00               | 0.038               | 0.00                |
| FSS E-Box                   | 0.26           | 0.00               | 0.26                | 0.26                |
| Magnetometer                | 0.00           | 0.00               | 0.69                | 0.00                |
| Star Tracker                | 0.00           | 0.00               | 0.76                | 0.00                |
| UHF Transceiver             | 0.35           | 1.00               | 0.35                | 0.35                |
| DCP Camera                  | 0.00           | 0.00               | 12.00               | 0.00                |
| PPDH<br>(Low/Upper/Dig)     | 0.00           | 0.00               | 5.15 / 5.15 / 1.048 | 2.95 / 2.95 / 1.048 |
| PLSU FoCM                   | 0.00           | 0.00               | 14.00               | 14.00               |
| <b>Worst case summation</b> | <b>94.2</b>    | <b>8.9</b>         | <b>93.2</b>         | <b>94.2</b>         |

\*T Transient dissipation over the simulation duration.

### 3.4 Boundary Conditions

The boundary conditions adopted for the orbital thermal simulations are specified through the orbit parameters, spacecraft attitude, and external radiative environment summarized in Tables 6, 7, and 8, respectively. Together, these inputs establish the spacecraft exposure to the surrounding thermal environment throughout the investigated mission scenarios.

The analysis considers two spacecraft orientations: SunTrack and Nadir. The applicable orientation is selected according to the corresponding mission phase, thereby changing the geometric exposure of the external surfaces to direct solar radiation, Earth reflected solar radiation (albedo), and terrestrial infrared emission. The orbital parameters also determine the periods of solar illumination and eclipse encountered by the spacecraft. To represent the limiting environmental conditions, separate sets of external flux inputs are prescribed for the hot and cold cases.

**Table 6:** Orbit definition parameters used in the LEOS-50MR platform satellite thermal model [46]

| Parameter                           | S01 Orbit         | S03 Orbit         |
|-------------------------------------|-------------------|-------------------|
| Orbit type                          | General Keplerian | General Keplerian |
| Altitude / Semi major axis [km]     | 500               | 500               |
| Local Time of Ascending Node (LTAN) | 09:30             | 09:30             |
| Inclination $i$ [deg]               | 97.406            | 97.406            |
| RAAN [deg]                          | 187.707           | 187.707           |
| Mean anomaly [deg]                  | 180               | 180               |
| Usage in analysis                   | S01B              | S03               |

**Table 7:** Orientation parameters [46]

| Parameter        | SunTrack                   | Nadir                                |
|------------------|----------------------------|--------------------------------------|
| Orientation mode | SunTrack                   | Nadir                                |
| Target direction | Sun                        | Earth                                |
| Reference vector | (0.3015, -0.9045, -0.3015) | (0, 0, 1)                            |
| Velocity vector  | (-1, 0, 0)                 | (1, 0, 0)                            |
| Used in analysis | S01B, Sun phases of S03    | Imaging / Post processing / Downlink |

**Table 8:** Environmental flux inputs [46]

| Parameter                          | Hot Case | Cold Case |
|------------------------------------|----------|-----------|
| Solar flux [W/m <sup>2</sup> ]     | 1414     | 1322      |
| Albedo coefficient                 | 0.35     | 0.25      |
| Planet IR flux [W/m <sup>2</sup> ] | 214      | 231       |
| Planet temperature [°C]            | -25      | -20.5     |

### 3.5 Reference Case Study Verification

After completing the construction of the thermal mathematical model, a verification process is performed to ensure that the reconstructed model correctly represents the reference spacecraft configuration and thermal behaviour. The verification is carried out using the imaging and post processing operational scenario, since this case includes significant internal heat dissipation, attitude transition, and transient thermal response. In this scenario, the spacecraft is initially stabilized in SunTrack orientation and then operated in Nadir orientation during the imaging and post processing phases. Therefore, it provides a suitable case for evaluating the accuracy of the implemented geometry, material properties, thermo-optical properties, environmental fluxes, conductive couplings, radiative exchange, and time dependent heat dissipation profiles.

The verification is performed by comparing the temperature results obtained from the reconstructed Systema Thermica<sup>®</sup> model with the corresponding reference case study results. For each subsystem, the minimum and maximum temperatures are extracted and compared to quantify the agreement between both models.

The verification results are summarized in Table 9. The results show good agreement between the reconstructed (baseline) model and the reference case study data,

with small deviations in both minimum and maximum temperatures. This confirms that the thermal model is correctly implemented and that the selected modelling assumptions are suitable for representing the spacecraft thermal response during the operational mode. In addition to the numerical comparison, the temperature time histories shown in Figs. 2–3 provide a direct visual comparison between the reconstructed (baseline) model and the reference results. The close agreement in the curve trends, temperature extrema, and transient behaviour confirms the validity of the model and provides confidence for using it in the subsequent experimental property update and passive thermal enhancement phases.

**Table 9:** Scenario S03 subsystem maximum and minimum temperatures verification

| Subsystem                          | Reference case study |         | Baseline model |         | Error   |         |
|------------------------------------|----------------------|---------|----------------|---------|---------|---------|
|                                    | Min [K]              | Max [K] | Min [K]        | Max [K] | Min [%] | Max [%] |
| Solar Panel                        | 262.20               | 297.05  | 269.89         | 290.39  | 2.93    | -2.24   |
| PPDH-PLSU                          | 262.23               | 300.81  | 271.92         | 309.75  | 3.70    | 2.97    |
| Power Sys.                         | 263.04               | 301.02  | 269.89         | 300.87  | 2.60    | -0.05   |
| Outer Str.                         | 261.57               | 295.93  | 270.02         | 290.10  | 3.23    | -1.97   |
| Inner Str.                         | 263.26               | 285.94  | 271.25         | 287.61  | 3.04    | 0.58    |
| COMS                               | 262.15               | 283.85  | 265.77         | 288.03  | 1.38    | 1.47    |
| CDH                                | 266.52               | 285.36  | 273.49         | 287.96  | 2.62    | 0.91    |
| ADCS                               | 263.35               | 289.03  | 270.39         | 288.16  | 2.67    | -0.30   |
| Battery                            | 264.91               | 301.12  | 271.91         | 300.87  | 2.64    | -0.08   |
| <b>Operating Temperature Range</b> |                      |         | 258 K : 308 K  |         |         |         |

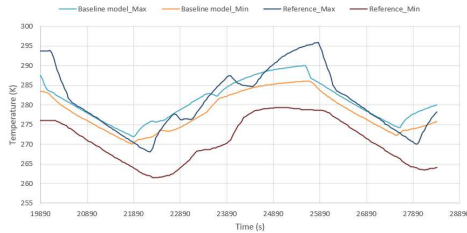


**Fig. 2:** Outer Structure temperature field

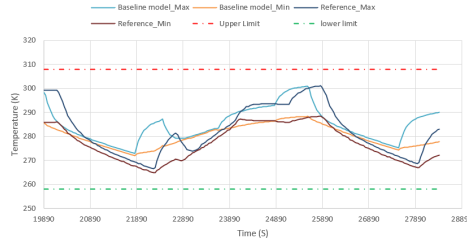


**Fig. 3:** Battery temperature field

The cold-case results are summarized in Table 10 that the spacecraft exhibits stable and repeatable thermal behaviour under minimum environmental and internal heat-load conditions. The temperature profiles as shown in Figs. 4–5 follow the expected orbital cyclic trend, where subsystem temperatures decrease during eclipse and recover during sunlight exposure. However, the overall results confirm that most subsystem temperatures remain close to or within the allowable operational limits, and the cold

case scenario provides an important reference for identifying components that may require additional passive insulation or thermal margin improvement in subsequent design phases.

**Table 10:** Maximum and minimum subsystem temperatures in Kelvin for Scenario S01B (Cold Case).

| Subsystem        | Max [K] | Min [K] |
|------------------|---------|---------|
| Structure (Str)  | 277     | 260     |
| Power            | 277     | 260     |
| Solar Panel (SP) | 277     | 260     |
| Battery          | 271     | 261     |
| ADCS             | 272     | 260     |
| COMS             | 273     | 256     |

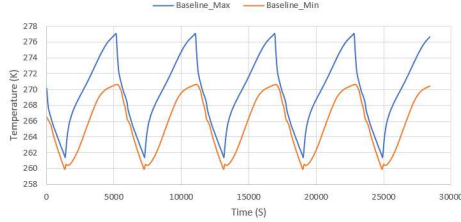


**Fig. 4:** Structure temperature field.

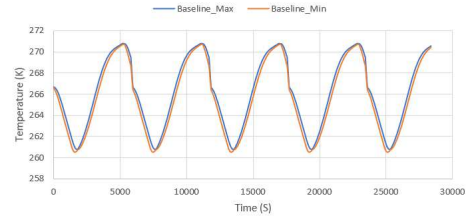


**Fig. 5:** Battery temperature field.

## 4 Experimental Characterization and Model Update

The experimental characterization and model update procedure is performed to improve the physical representativeness of the spacecraft thermal mathematical model. A representative aluminium structural panel is manufactured and prepared according to European Cooperation for Space Standardization (ECSS) oriented requirements for spacecraft structural materials, workmanship, surface preparation, contamination control, and quality assurance [47–53]. After manufacturing, the panel is subjected to a controlled black anodizing process in accordance with ECSS-Q-ST-70-03C, which defines black anodizing procedures [50].

The black anodized aluminium surface is an important case in passive spacecraft thermal control because it is characterized by high solar absorptivity and high infrared emissivity. Experimental studies on cobalt sulphide black anodic films report typical values of  $\alpha_s \approx 0.90\text{--}0.93$  and  $\varepsilon \approx 0.90\text{--}0.95$ , resulting in an  $\alpha_s/\varepsilon$  ratio close to unity [54, 55]. This near unity ratio indicates that black anodized aluminium behaves as a nearly non selective radiative surface, where the coating absorbs a high fraction of incident solar radiation while also re emitting thermal energy efficiently in the infrared

range. Consistent with this behaviour, ECSS recommendations for black anodizing specify high values of both solar absorptance and infrared emittance for space applications [56]. Therefore, using assumed handbook values for this coating may introduce uncertainty in the predicted spacecraft radiative balance, especially for external panels exposed to solar radiation, Earth albedo, Earth infrared radiation, and deep space cooling. To reduce this uncertainty, the thermo-optical properties of the manufactured black anodized aluminium panel are experimentally measured using the ET10 reflectometer [57], as shown in Figure 6 the average measured values of absorptivity and emissivity listed in Table 11 are incorporated into the thermal model.

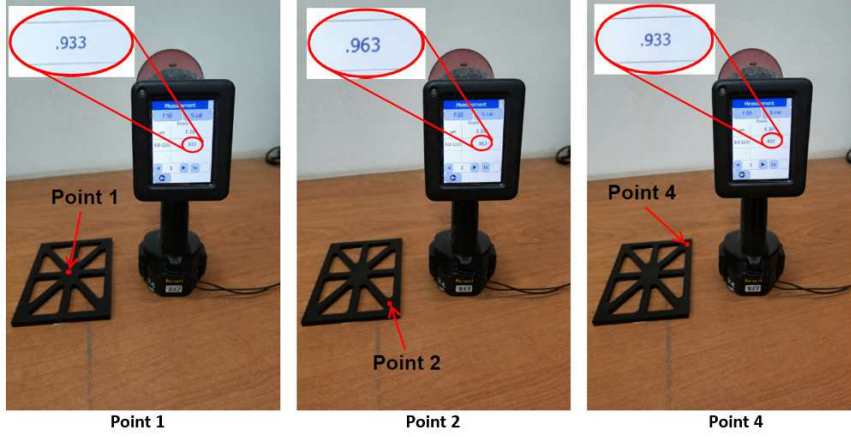


**Fig. 6:** Black anodized thermo-optical properties measurement

Before applying the measured values to the full thermal analysis, a sensitivity analysis is conducted to evaluate the influence of the ET10 measurement uncertainty on the predicted temperature response. The manufacturer stated accuracy of approximately  $\pm 0.03$  is considered, and two bounding cases are introduced into the model:  $\alpha = \varepsilon = 0.91$  and  $\alpha = \varepsilon = 0.97$ . The battery subsystem is selected as the reference component because it is one of the most thermally critical units and has relatively narrow allowable temperature margins as shown in Fig. 7. The sensitivity results showed that the maximum battery temperature changed from 300.39 K to 299.49 K, corresponding to a variation of approximately 0.30%. The minimum battery temperature changed from 266.17 K to 265.24 K, corresponding to a variation of approximately 0.35%. These small variations indicate that the measurement uncertainty has a limited effect on the predicted battery temperature and that the experimentally measured thermo-optical properties are reliable for model updating.

After completing the sensitivity assessment, the original thermal model is updated by replacing the previously assumed black anodized surface properties with the experimentally measured values. The purpose of this phase is to improve the accuracy of the radiative boundary conditions and to make the numerical model more representative of the actual treated spacecraft surface. The updated model is then analysed under

**Table 11:** Thermo-optical properties of the anodized aluminium panel.

| Measurement Point | Absorptivity $\alpha$ | Emissivity $\varepsilon$ | Accuracy        |
|-------------------|-----------------------|--------------------------|-----------------|
| Point 1           | 0.933                 | 0.933                    | $\pm 0.03$      |
| Point 2           | 0.963                 | 0.963                    | $\pm 0.03$      |
| Point 3           | 0.932                 | 0.932                    | $\pm 0.03$      |
| Point 4           | 0.933                 | 0.933                    | $\pm 0.03$      |
| <b>Average</b>    | <b>0.940</b>          | <b>0.940</b>             | $\pm 0.03$ [57] |

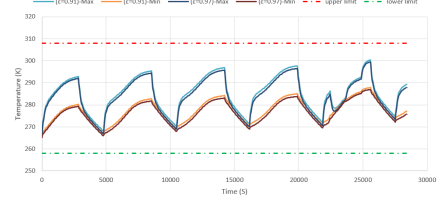


**Fig. 7:** Battery temperature sensitivity analysis.

the selected cold and imaging/post processing scenarios, and the resulting subsystem temperatures are compared with the baseline model to evaluate the influence of experimentally measured surface properties on the predicted spacecraft thermal behaviour as listed in Table 12–13 and shown in Figs. 8–11.

**Table 12:** Comparison of minimum and maximum subsystem temperatures for Scenario S01B Cold (Baseline Model vs. Updated Model).

| Subsystem  | Baseline model |         | Updated model |         |
|------------|----------------|---------|---------------|---------|
|            | Max [K]        | Min [K] | Max [K]       | Min [K] |
| Outer Str. | 277            | 260     | 276           | 259     |
| Battery    | 271            | 261     | 270           | 259     |
| Power      | 278            | 260     | 277           | 259     |
| COMS       | 273            | 256     | 272           | 255     |
| ADCS       | 272            | 260     | 271           | 255     |

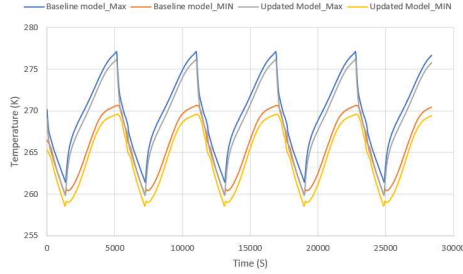


**Fig. 8:** Outer structure's temperatures for cold case S01B (Baseline Model vs. Updated Model)

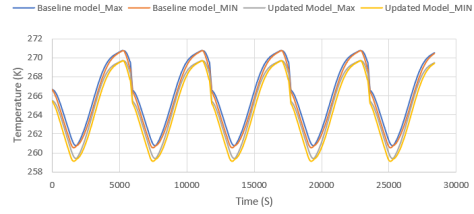


**Fig. 9:** Battery's temperatures for cold case S01B (Baseline Model vs. Updated Model)

**Table 13:** Comparison of minimum and maximum subsystem temperatures for Scenario S03 (Baseline Model vs. Updated Model).

| Subsystem                          | Baseline model |                      | Updated model |         |
|------------------------------------|----------------|----------------------|---------------|---------|
|                                    | Min [K]        | Max [K]              | Min [K]       | Max [K] |
| Outer Str.                         | 266            | 290                  | 265           | 289     |
| Inner Str.                         | 267            | 288                  | 266           | 287     |
| PPDH-PLSU                          | 267            | 310                  | 266           | 308     |
| Power                              | 266            | 301                  | 265           | 300     |
| Solar Panel                        | 266            | 290                  | 265           | 289     |
| COMS                               | 262            | 288                  | 261           | 287     |
| CDH                                | 268            | 288                  | 267           | 287     |
| ADCS                               | 266            | 288                  | 265           | 287     |
| Battery                            | 266            | 301                  | 265           | 300     |
| <b>Operating Temperature Range</b> |                | <b>258 K : 308 K</b> |               |         |

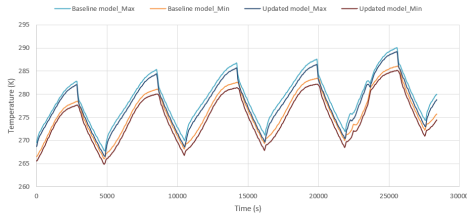


**Fig. 10:** Outer structure's temperatures for Imaging/post-processing Scenario S03 (Baseline Model vs. Updated Model)

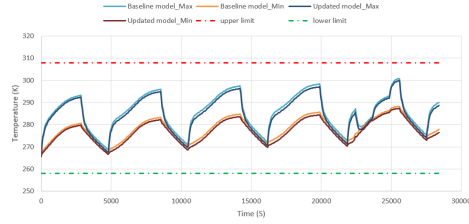


**Fig. 11:** Battery's temperatures for Imaging/post-processing Scenario S03 (Baseline Model vs. Updated Model)

## 5 Passive Thermal Control System Performance Enhancement

The passive thermal control enhancement phase, referred to as enhanced thermal subsystem model, is developed to improve the spacecraft thermal margin using a simple, and reliable thermal control approach. The updated model is used as the reference configuration because it represents the spacecraft more realistically than the baseline model. The previous results showed that updating the black anodized thermo-optical properties produced limited temperature changes, but some subsystems still operated close to the lower allowable temperature limit in the cold case. This behaviour is mainly caused by the low internal heat dissipation during survival conditions and the continuous radiative heat loss from the spacecraft external surfaces during eclipse periods. Therefore, an additional passive thermal control solution is required to reduce excessive heat loss and improve the cold case temperature margins.

Multi-layer insulation (MLI) is selected as the passive enhancement method in enhanced thermal subsystem model because it is widely used in spacecraft thermal control systems to reduce radiative heat exchange between external surfaces and the

space environment. In this work, the MLI is applied to the bottom panel of the spacecraft, which has a significant influence on the overall heat balance of the satellite. This panel contributes to heat rejection during both sunlight and eclipse phases; therefore, reducing its effective radiative heat loss can increase the minimum subsystem temperatures, especially during the cold scenario. However, the use of MLI must also be evaluated under the hot imaging and post processing scenario to ensure that the added insulation does not cause overheating or exceed the upper allowable temperature limit.

The enhanced thermal subsystem model performance is evaluated by comparing three model states: the baseline model, the updated model, and the enhanced passive thermal control model with MLI. The baseline model represents the initial thermal design, the updated model represents the effect of incorporating the experimentally measured black anodized properties, and enhanced thermal subsystem model represents the final passive thermal control configuration after adding MLI with its properties listed in Table 14. The assessment is performed for both the S01B cold case and the S03 imaging/post processing scenario by comparing the minimum and maximum temperatures of the main spacecraft subsystems.

The main objective of enhanced thermal subsystem model is to demonstrate that the proposed passive enhancement can increase the cold case thermal margin while keeping the spacecraft within the allowable operational range during the hot operational scenario. The following results present the temperature response before and after applying MLI and discuss the effect of the passive enhancement on the satellite subsystems.

The emissivity of the selected MLI surface is experimentally measured as shown in Figure 12 to define a more representative radiative property in the thermal model. Five measurement points listed in Table 15 are taken on the MLI sample using the ET10 reflectometer to account for possible local variation over the surface. The average emissivity is calculated as  $\varepsilon = 0.4232$ , with an instrument accuracy of approximately  $\pm 0.03$  [57]. This measured value is then assigned to the MLI covered bottom panel in the enhanced thermal model. The use of the experimentally measured emissivity improves the reliability of the enhanced thermal subsystem model thermal analysis because the radiative heat exchange of the insulated panel is directly controlled by its effective infrared emissivity. Therefore, the MLI implementation in the model reflects the measured surface behaviour rather than relying only on generic database values. The enhanced thermal model is then analysed under the selected cold and imaging/post processing scenarios, and the resulting subsystem temperatures are compared with the baseline model to evaluate the influence of MLI implementation as listed in Tables 16–17. The plotted temperature histories show that the enhanced MLI model increases the minimum temperature levels while preserving the same cyclic orbital trend observed in the baseline model as shown in Figs. 13–16.

## 6 Conclusion

This paper presents a measurement-driven passive thermal control methodology for a LEO Earth observation small satellite. A detailed thermal model was developed in Systema Thermica® and verified against baseline reference data with most deviations



**Fig. 12:** MLI's emissivity measurement using the ET10 reflectometer.

**Table 14:** MLI's properties [58]

| Property                     | Value |
|------------------------------|-------|
| Solar absorptivity, $\alpha$ | 0.35  |
| Thickness, $t$ (m)           | 0.001 |

**Table 15:** Measured emissivity of MLI.

| Point          | $\epsilon$    | Accuracy                          |
|----------------|---------------|-----------------------------------|
| Point 1        | 0.415         | —                                 |
| Point 2        | 0.419         | —                                 |
| Point 3        | 0.421         | —                                 |
| Point 4        | 0.428         | —                                 |
| Point 5        | 0.433         | —                                 |
| <b>Average</b> | <b>0.4232</b> | <b><math>\pm 0.03</math></b> [57] |

**Table 16:** Subsystem temperature limits comparison for the hot case (S03): Baseline Model, Updated Model, and Enhanced thermal subsystem Model

| Subsystem                          | Baseline Model |         | Updated Model        |         | Enhanced thermal Model |         |
|------------------------------------|----------------|---------|----------------------|---------|------------------------|---------|
|                                    | Min (K)        | Max (K) | Min (K)              | Max (K) | Min (K)                | Max (K) |
| Outer Structure                    | 266            | 290     | 265                  | 289     | 276                    | 298     |
| Inner Structure                    | 267            | 288     | 266                  | 287     | 276                    | 296     |
| Battery                            | 267            | 301     | 266                  | 300     | 276                    | 300     |
| Solar Panel                        | 266            | 290     | 265                  | 290     | 276                    | 298     |
| PPDH-PLSU                          | 267            | 310     | 266                  | 309     | 276                    | 308     |
| Power                              | 266            | 301     | 265                  | 300     | 276                    | 300     |
| COMMS                              | 262            | 288     | 261                  | 287     | 270                    | 297     |
| CDH                                | 268            | 288     | 267                  | 287     | 277                    | 296     |
| ADCS                               | 266            | 288     | 265                  | 287     | 276                    | 298     |
| <b>Operating Temperature Range</b> |                |         | <b>258 K : 308 K</b> |         |                        |         |

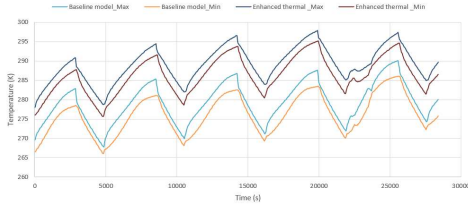


**Fig. 13:** Outer structure's temperatures for Imaging/post-processing Scenario S03 (Baseline Model vs. Enhanced MLI Model)

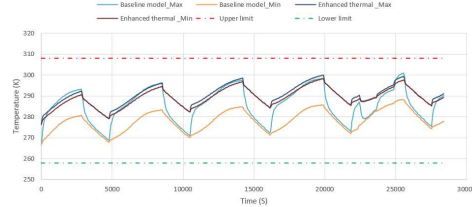


**Fig. 14:** Battery's temperatures for Imaging/post-processing Scenario S03 (Baseline Model vs. Enhanced MLI Model)

close to  $\pm 3\%$  and a maximum absolute deviation of approximately 3.7% across two representative orbital scenarios (S01B cold case and S03 operational case). Incorporating measured thermo-optical properties of black-anodized aluminum improved the

**Table 17:** Subsystem temperature limits comparison for the cold case (S01B): Baseline Model, Updated Model, and Enhanced thermal subsystem Model)

| Subsystem                                        | Baseline Model |         | Updated Model |         | Enhanced thermal Model |         |
|--------------------------------------------------|----------------|---------|---------------|---------|------------------------|---------|
|                                                  | Min (K)        | Max (K) | Min (K)       | Max (K) | Min (K)                | Max (K) |
| Structure                                        | 260            | 277     | 259           | 276     | 271                    | 287     |
| Power                                            | 260            | 277     | 259           | 277     | 271                    | 288     |
| Solar Panel                                      | 260            | 277     | 263           | 288     | 271                    | 288     |
| Battery                                          | 261            | 271     | 259           | 270     | 273                    | 283     |
| ADCS                                             | 260            | 272     | 258           | 271     | 273                    | 284     |
| COMMS                                            | 256            | 273     | 255           | 272     | 267                    | 285     |
| <b>Operating Temperature Range</b> 258 K : 308 K |                |         |               |         |                        |         |

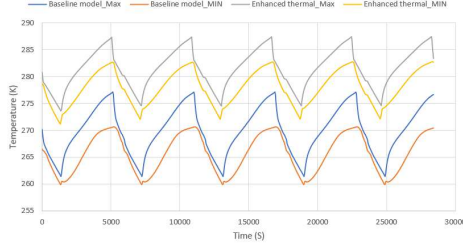


**Fig. 15:** Outer structure's temperatures for cold Scenario S01B (Baseline Model vs. Enhanced MLI Model)

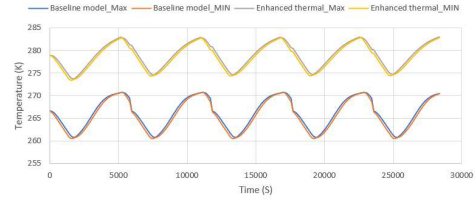


**Fig. 16:** Battery's temperatures for cold Scenario S01B (Baseline Model vs. Enhanced MLI Model)

model's physical realism, shifting subsystem temperatures by 1–2 K. Finally, introducing Multi-Layer Insulation (MLI) boosted cold case minimum temperatures by 11–13 K without exceeding the allowable operational range (258–308 K) during high power scenarios. Therefore, the combination of experimental surface property characterization and passive thermal design enhancement provides a practical and reliable approach for improving the thermal performance of small LEO Earth observation satellites.

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