

Article

Thermal Design of a Microsatellite in Sun-Synchronous Orbit Based on Bayesian Optimization

Yigithan Mehmet Kose * and Murat Celik 

Department of Mechanical Engineering, Bogazici University, Istanbul 34342, Türkiye; murat.celik@bogazici.edu.tr

* Correspondence: mehmet.kose.2022@alumni.bogazici.edu.tr

Abstract

The thermal control subsystem is one of the critical subsystems of a satellite. Optimization of the design of this subsystem could benefit a satellite's power and mass budgets. In previous optimization studies regarding the thermal management of low Earth orbit satellites, solstices are typically assumed to be the thermally extreme cases. However, this could not always be accurate. In addition, previous studies often evaluated specific parts of the thermal system, while overlooking the dynamics of energy conversion within the satellite. In this study, thermally extreme cases are identified using a lumped analysis approach. Moreover, heat dissipation of certain components is modeled based on the relationship between the generation and the consumption of power. The process begins with an orbital analysis, which provides inputs for the lumped analysis. According to the results of the lumped analysis, finite element analyses are performed for the worst hot and the worst cold cases. The aim is to minimize the need for heater energy consumption by adjusting the coordinates of the corners of the radiators, where constraints are set by the safe temperature intervals of specific components. This computationally demanding optimization problem is solved by Bayesian optimization, also a first for a satellite design study. The optimized system requires a reduced amount of heater energy while satisfying the temperature constraints.

Keywords: spacecraft thermal control; microsatellite; thermal system design; thermal analysis; sun-synchronous orbit; Bayesian optimization

1. Introduction

Earth-orbiting satellites are being employed in ever greater numbers for applications ranging from Earth observation, communication, intelligence gathering, and navigation to early warning systems. In order to function properly, satellites have communication and power instruments, sensors, actuators, and processors, all of which dissipate heat. Additionally, satellites receive thermal radiation from the sun and Earth, which varies significantly during operation. Considering that all equipment inside the satellite have a safe operating temperature interval, thermal management using a thermal control subsystem (TCS) is necessary to ensure that the satellite remains functional. Ultimately, heat supplied, which is the addition of the amount of the internally generated heat and the amount of the externally absorbed heat, will be balanced by the heat removed, which is the amount of heat radiated out to space. Therefore, TCSs are utilized to manipulate the thermo-optical properties of the outer surfaces, and the conduction/storage paths in the interior, so as to control the transient temperature field of the satellite. TCSs include, but are not limited to, multilayer insulation blankets (MLI), radiators, heaters, thermal straps, heat pipes, and surface finishes. Depending



Academic Editor: Fanghua Jiang

Received: 2 June 2026

Revised: 16 July 2026

Accepted: 3 August 2026

Published: 7 August 2026

Copyright: © 2026 by the authors.

Licensee MDPI, Basel, Switzerland.

This article is an open access article distributed under the terms and conditions of the [Creative Commons Attribution \(CC BY\)](https://creativecommons.org/licenses/by/4.0/) license.

on the mission and the size of the satellite, the TCS design should be as passive and simple as possible. For small satellites, TCS can be composed almost entirely of passive methods [1–4].

Satellites orbiting Earth are exposed to three main external thermal loads: direct solar thermal radiation, reflection of solar thermal radiation from the Earth (Albedo), and planetary thermal radiation. In order to manipulate the amount of the thermal load absorbed and emitted by a particular exterior satellite surface, special thermo-optical properties are often utilized in TCS design. Nevertheless, environmental conditions in space, such as the presence of atomic oxygen, charged particles, and ultraviolet radiation, degrade exterior spacecraft surfaces throughout the mission duration. This may cause a change in the thermo-optical properties of these surfaces, especially towards an increase in solar absorptivity. This gives rise to different emission and absorption characteristics in the beginning-of-life (BOL) and the end-of-life (EOL) of the mission. In addition, both the amount of external heat exposure and the internal heat generation during the mission are variables. When designing a spacecraft's TCS, two extreme cases, the worst hot case (WHC) and worst cold case (WCC), are considered. In WHC, EOL conditions are assumed, and all external as well as internal thermal loads are assumed to be at their maximum expected values. In WCC, BOL conditions are assumed, along with the minimum expected external and internal heat supply values [5]. If the spacecraft is designed to thermally survive both of these cases, ideally it is expected to survive all other cases as well. Typically, the radiator area is determined first, considering the WHC, and then the heater power is determined, considering the WCC. If the radiator area is more than sufficient, it will result in a greater heater power requirement and eventually cause TCS to occupy a greater share in the mass and power budgets, which are scarce resources for a spacecraft. Thermal design optimization is thus desirable for the efficient operation of the satellite, in addition to the fact that it reduces the trial-and-error in the design process [6].

Several studies have evaluated the optimization of satellite TCS design in previous decades [7–13]. In recent studies, Xiong et al. [14] proposed an intelligent density-based global sensitivity analysis framework for spacecraft thermal design, combining machine learning, multi-fidelity modeling, and a neural network surrogate to achieve high accuracy with significantly reduced computational cost. Wang et al. [15] introduced an intelligent optimization strategy using blind Kriging and many-objective evolutionary algorithms to minimize thermal power consumption and temperature deviation in a high-resolution spaceborne optical sensor, with results validated by thermal vacuum testing. Bai et al. [16] presented an optimized thermal design for a 310 kg X-band synthetic aperture radar (SAR) satellite in low Earth orbit (LEO), achieving zero-power thermal control for the antenna and reducing heater usage by 30% while maintaining temperature stability, verified through thermal vacuum testing. Cannon et al. [17] proposed a passive thermal control system based on deployable radiator panels for CubeSats. Using thermal simulations and experimental validation, they optimized the radiator deployment concept to achieve up to a 60 °C reduction in spacecraft body temperature while maintaining a compact stowed configuration. Chen et al. [18] proposed an adaptive thermal radiation design based on smart radiator devices with temperature-dependent emissivity. Their simulations demonstrated substantial reductions in spacecraft temperature fluctuations and thermal control power consumption, while incorporating deep learning to accelerate the optimization of radiator radiative properties. Finally, Tribak et al. [19] presented a multi-objective thermal optimization framework for a 1U CubeSat, demonstrating the significant influence of the orbital Beta angle on temperature distributions and reliability while simultaneously optimizing thermal performance and mission constraints. Nevertheless, winter and summer solstices are selected as the thermally extreme cases in most of these studies. Moreover, the satellite is often evaluated in a segmented manner.

In this study, a novel approach for the determination and analyses of the extreme cases is presented. The extreme cases are detected more accurately than the conventional approach by the annual screening of the lumped temperature, which are shown to be different from the solstices, and a finite element analysis (FEA) is carried out for those identified cases. Obviously, this requires evaluating a significantly more expensive objective function, which is a challenge for conventional global optimization algorithms. However, Bayesian optimization, a machine learning-based method, is much more efficient in terms of function evaluations compared to those exploitation-only methods, as it balances exploration and exploitation [20]. Bayesian optimization relies on a surrogate model, which is usually a Gaussian process, and an acquisition function to optimize any black-box function. Hence, this study utilized Bayesian optimization in the TCS design of a generic microsatellite for the first time. The objective will be to minimize the total energy consumed by the heaters under the constraints set by the operating temperature intervals of several components. This objective function indirectly involves the minimization of the radiator area as well; thus, it considers both the mass and the power budgets. In this process, a systems engineering approach is adopted in the identification of heat-generating components and the evaluation of the satellite as a whole.

2. Method and Formulation

In this study, a satellite deployed in sun-synchronous orbit (SSO) is investigated, where the Kepler parameters of the evaluated orbit are given in Table 1. The generic view of the satellite under investigation is shown in Figure 1. The satellite deploys two solar panel wings, as shown in this figure. $\pm X$ and $+Y$ panels of the satellite are covered by radiators and MLI blankets, whereas the $\pm Z$ panels are entirely covered by MLI blankets. The satellite is thought to maintain nadir alignment in $+Z$ axis with a velocity constraint in $+X$ axis. $-Y$ face, on which another solar panel is mounted, directly faces the sun. The satellite is assumed to contain a certain number of generic electronic components, in addition to a transponder, a battery, a power conditioning and distribution unit (PCDU), an antenna, and a propulsion unit.

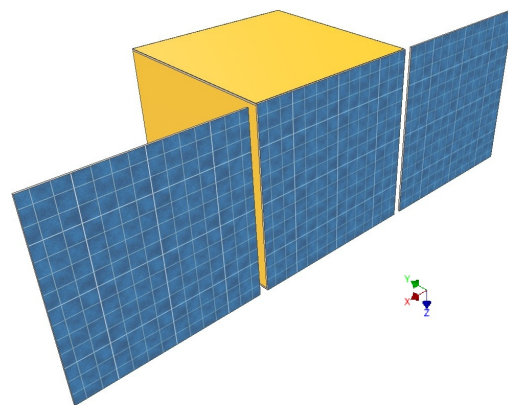


Figure 1. Simplified view of a generic satellite with all solar panels placed on one side.

Table 1. Orbital parameters used in the analysis.

Orbital Parameter	Value
Altitude	500 km
Eccentricity	0
Inclination	97.4°
RAAN	0°
Argument of Periapsis	0°
True Anomaly	0°
Epoch	1 Jan

2.1. Methodology

The methodology of this study, depicted in Figure 2, can be summarized as follows. First, an orbital analysis including the maneuvers and the orientations corresponding to all operational modes is performed using the Systems Tool Kit (STK). From this analysis, the angles between the body axes of the satellite and the solar vector, the satellite–sun distance, fraction of solar intensity reaching the satellite, and the angle between the Earth–satellite vector and the Earth–sun vector are obtained in a transient fashion for a one-year period. This data is used in MATLAB R2021a for the calculation of the solar and the Albedo heat fluxes. The view factors of structural and solar panels to Earth are found using the Monte Carlo Ray Tracing (MCRT) algorithm. After the planetary heat flux is calculated as well, the total absorbed external heat flux is calculated for WHC and WCC separately. Then, the internal heat generation profiles of the electronic components and the PCDU are obtained, which leads to the solution of the coupled equations of lumped temperatures, battery heat dissipation, and solar panel power generation. This analysis is carried out for two separate cases:

- ◇ Throughout a year, assuming EOL properties and maximum heat inputs.
- ◇ Throughout a year, but assuming BOL properties and minimum heat inputs.

Satellite lumped temperatures resulting from these analyses are tracked through time, and the hottest instance in the first analysis is labeled as the WHC, whereas the coldest instance in the second analysis is labeled as the WCC. This is a novel approach which could enable predicting the worst thermal cases that a satellite can be exposed to more accurately than choosing the winter or summer solstices as the extreme cases. Different orientations of the satellite may expose less protected faces to external fluxes; therefore, the instance of the highest solar flux may not be equal to the instance of the highest temperature.

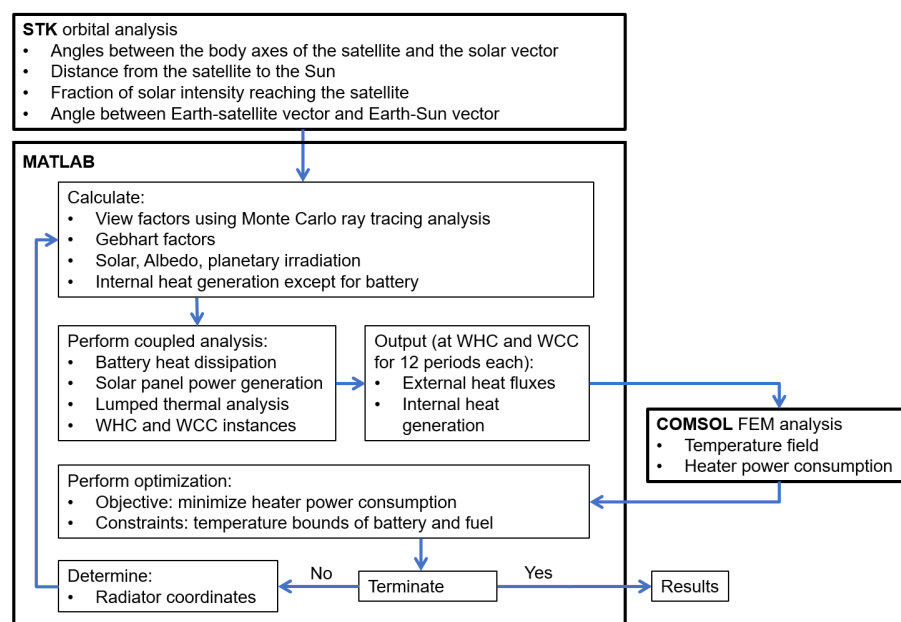


Figure 2. The flowchart of the data flow for the optimization.

After the WHC and WCC instances are detected, a 3D thermal analysis is conducted at both of these instances for 12 revolutions around Earth. This analysis employs a FEA tool, COMSOL 6.3, with WHC and WCC instances occurring midway through the 12-periods analysis durations. In the FEA, the heat fluxes calculated in MATLAB are defined on the external faces, and internally heat generating components are defined accordingly. MLI blankets are modeled using an effective thermal conductivity model [21]. The FEA

model is solved by evaluating the conductive as well as radiative heat transfer between the elements with the proper boundary conditions, and the resulting temperatures of various components are obtained as the output. All these calculations are carried out for a specific set of values defining the dimensions of the radiators that are located on three faces. The optimization algorithm assigns this set of values to minimize the energy consumed by the heaters, while satisfying the upper and the lower temperature limits of certain components. In each iteration, a new radiator geometry is created, which eventually causes a new set of external heat fluxes corresponding to the WHC and WCC instances to be used in the two FEAs.

2.2. Lumped Analysis Formulation

In order to formulate the lumped analysis, relations for the amount of externally absorbed thermal radiation should be identified. When Figure 1 is inspected for this purpose, it is notable that there are two open radiative enclosures on both sides of the satellite. One enclosure consists of the rear surface of the solar panel, the radiator, and the MLI blanket on the +X side, and the other enclosure consists of the same entities on the opposite side. On these surfaces, there are primary and secondary external irradiations. Secondary irradiations are due to multiple reflections within the enclosure. This effect is accounted for in the analysis assuming that all surfaces are diffuse. Therefore, Gebhart factors are used to calculate the secondary irradiations absorbed by each surface. Gebhart factors for the infrared spectrum on the +X side are calculated as follows [22]:

$$\begin{bmatrix} 1 & -F_{12}^+\rho_2 & -F_{13}^+\rho_3 \\ -F_{21}^+\rho_1 & 1 & -F_{23}^+\rho_3 \\ -F_{31}^+\rho_1 & -F_{32}^+\rho_2 & 1 \end{bmatrix} \begin{bmatrix} B_{11}^+ & B_{12}^+ & B_{13}^+ \\ B_{21}^+ & B_{22}^+ & B_{23}^+ \\ B_{31}^+ & B_{32}^+ & B_{33}^+ \end{bmatrix} = \begin{bmatrix} 0 & F_{12}^+\epsilon_2 & F_{13}^+\epsilon_3 \\ F_{21}^+\epsilon_1 & 0 & F_{23}^+\epsilon_3 \\ F_{31}^+\epsilon_1 & F_{32}^+\epsilon_2 & 0 \end{bmatrix} \quad (1)$$

Here, B_{ij} are Gebhart factors which is the fraction of the radiant energy emitted by i that is absorbed by j directly and indirectly. In the notation adopted for the Gebhart factors, 1 denotes the radiator surface, 2 denotes the rear surface of the solar panel, and 3 denotes the MLI surface. The superscripts + and i denote the enclosure on the +X side and the infrared spectrum, respectively. Here, it should be noted that Gebhart factors should be calculated for infrared and solar bands separately, as surface emissivities, ϵ_λ , at these bands are different.

Another thing to note in Figure 1 is that some parts of the external irradiation towards the surfaces within the radiative enclosures are blocked by other surfaces. Regarding Albedo and the planetary irradiations, this blockage effect is embedded in the calculation of the view factors, where the view factor calculations have been complicated due to this effect. Regarding the solar irradiation, however, a different method should be adopted as it is approximated as rays directed from a fixed point rather than being calculated based on view factors. Thus, the blockage effect for solar irradiation is considered by calculating the shadow area on the +X and −X faces, and assuming that the irradiation is proportional to the illuminated portion. As depicted in Figure 3, two angles are needed to determine the shadow area on $\pm X$ panels. These angles could be determined using the angles between the solar vector and the satellite body axes, which are extracted from STK.

The angles that are shown in Figure 3 are given as follows:

$$\Phi_X = \arccos \left[\frac{\cos \theta_X}{\sqrt{(\cos \theta_X)^2 + (\cos \theta_Y)^2}} \right] \quad (2)$$

$$\Phi_Y = \arccos \left[\frac{\cos \theta_Y}{\sqrt{(\cos \theta_Z)^2 + (\cos \theta_Y)^2}} \right] \quad (3)$$

where θ represents the angle between the solar vector and a particular axis of the satellite. Using Equations (2) and (3), the shadow area on the +X panel could be found as follows:

$$A_{sh,1} = \frac{W_{SP} \tan \Phi_X}{2} (2L_X - W_{SP} \tan \Phi_X \tan \Phi_Z) \quad (4)$$

where it should be noted that, except for the notation of the Gebhart factors, the numbering of surfaces and bodies follows the notation given in Table 2. Then, the solar irradiation incident on the +X panel is found by Equation (5). For other panels, it is much more straightforward, which is why the equations for those panels are not given.

$$q_{s,i,1} = S \cos \theta_X \frac{A_1 - A_{sh,1}}{A_1} \quad (5)$$

where the solar heat flux is given by:

$$S = \frac{C_s f_s}{(R_{sat \rightarrow sun} / AU)^2} \quad (6)$$

where $R_{sat \rightarrow sun}$ is the satellite–sun distance, f_s is the fraction of solar intensity reaching the satellite, AU is the astronomical unit, and C_s is the solar constant ($C_s = 1367 \text{ W/m}^2$). The solar irradiation absorbed by the radiator on the +X panel, taking multiple reflections into account, could be found as follows:

$$q_{s,1,r} = q_{s,i,1} [\alpha_r + B_{11}^{+s} (1 - \alpha_r) + B_{31}^{+s} (1 - \alpha_m)] \quad (7)$$

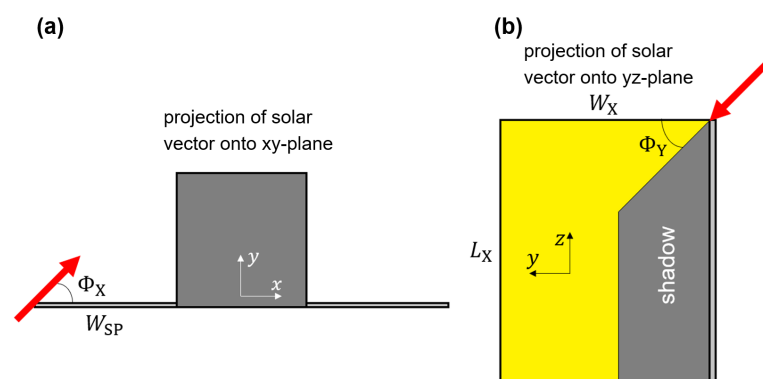


Figure 3. Shading conditions of $\pm X$ panels observed from (a) xy-plane, (b) yz-plane. The red arrow represents the solar vector.

It should also be noted that although the rear faces of the solar panels do not see the sun, due to the multiple reflections within the enclosure, they do absorb some amount of solar irradiation. For the rear face of the solar panel on the +X side, it could be calculated as follows:

$$q_{s,3} = q_{s,i,1} [B_{12}^{+s} (1 - \alpha_r) + B_{32}^{+s} (1 - \alpha_m)] \quad (8)$$

Additionally, Albedo irradiation absorbed by the radiator on the +X panel is given using the Gebhart factors by:

$$q_{a,1,r} = aS \cos \xi [\alpha_r F_{1,e} + B_{11}^{+s} F_{1,e} (1 - \alpha_r) + B_{21}^{+s} F_{2,e} (1 - \alpha_b) + B_{31}^{+s} F_{3,e} (1 - \alpha_m)] \quad (9)$$

where a is the Albedo coefficient. It should be noted that since the Earth–sun distance is orders of magnitude greater than the Earth–satellite distance, solar Zenith angle ζ , the angle between the Earth–satellite vector and the Earth–sun vector, is approximately equal to the angle between the solar vector and the $+Z$ axis of the satellite.

The last external source of irradiation, the Earth heat flux, on the $\pm X$ panels is given by:

$$q_{e,1,r} = I_e \left[\epsilon_r F_{1,e} + B_{11}^{+i} F_{1,e} (1 - \epsilon_r) + B_{21}^{+i} F_{2,e} (1 - \epsilon_b) + B_{31}^{+i} F_{3,e} (1 - \epsilon_m) \right] \quad (10)$$

where I_e is the Earth's radiation intensity. View factors from the $\pm X$ panels and the rear surface of the solar panels to Earth, which are necessary for finding the Albedo and the planetary irradiations, are found using the MCRT method [23]. The algorithm is applied for the geometry represented in Figure 4 to find the view factors to Earth, where the rays hitting the Earth are identified as those which intersect the virtual tangent disk, whose center aligns with the center of the depicted coordinate system. The view factors from other faces to Earth, and those between the solar panels and the structural panels are found using the analytical formulae given in [24].

Table 2. Notation used for the lumped formulation.

Surface	Number	Body	Letter
+X panel	1	Satellite body	A
−X panel	2	+X solar panel	B
+X solar panel rear face	3	−X solar panel	C
−X solar panel rear face	4	BM solar panel	D
+Y panel	5	+X MLI	E
−Y panel	6	−X MLI	F
+Z panel	7	+Y MLI	G
−Z panel	8	+Z MLI	H
		−Z MLI	I

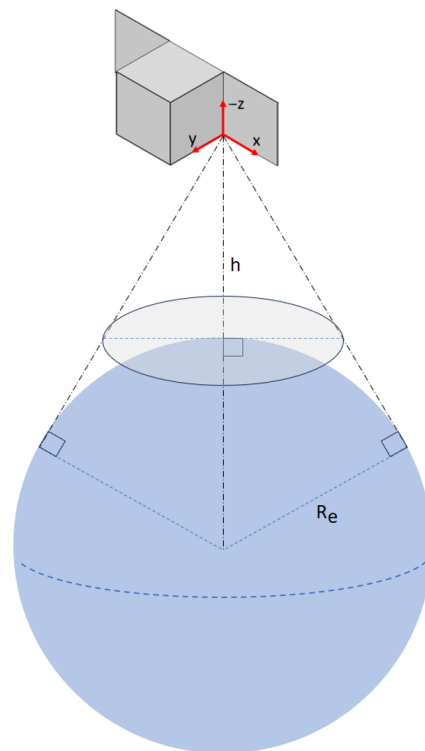


Figure 4. Geometry evaluated in MCRT algorithm.

The next step is to identify the total internal heat generation. First, the heat dissipation of all other instruments is specified, using which the heat dissipation of the PCDU can be found, which is given by [25]:

$$Q_{\text{PCDU}} = \sum Q_{\text{comp}} \left(\frac{1}{\eta_{\text{PCDU}}} - 1 \right) \quad (11)$$

where $\sum Q_{\text{comp}}$ expresses the amount of power dissipated by all other equipment, including heaters. Additionally, the power extracted by the PCDU equals $P_{\text{PCDU}} = \sum Q_{\text{comp}} / \eta_{\text{PCDU}}$.

After all external and internal thermal loads except for the dissipation of the battery are identified, coupled analysis begins. In this system, the satellite body, three solar panels, and MLI blankets on five surfaces constitute a thermal network of nine nodes visualized in Figure 5. To begin with, the battery dissipates heat both when charging and when discharging. The heat dissipation of the battery depends on the rate and efficiency of charge and discharge, which is calculated as follows [25]:

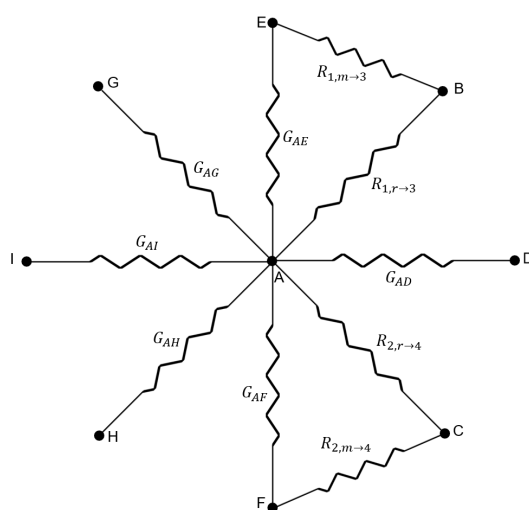


Figure 5. Visualization of the thermal network. The labeling of the nodes matches the labeling of the bodies in Table 2. G denotes conductive coupling and R denotes radiative coupling.

$$Q_{\text{bat}} = \begin{cases} (P_{\text{gen}} - P_{\text{PCDU}})(1 - \eta_{\text{b,c}}), & \text{if } P_{\text{PCDU}} < P_{\text{gen}} \\ (P_{\text{PCDU}} - P_{\text{gen}})(1 - \eta_{\text{b,d}}), & \text{otherwise} \end{cases} \quad (12)$$

where total internal heat generation is obtained by adding those of all components, the PCDU and the battery. The electrical power generated by the solar panels is given by:

$$P_{\text{gen}} = A_{\text{SP}} f_p \eta_c(T) \cos(\pi - \theta_Y) S \quad (13)$$

where f_p is the packing factor, and η_c is the cell efficiency, which depends on the temperature of the solar panels. The temperature of the solar panel wing on the +X side is given by [26]:

$$m_{\text{BCB}} \frac{\partial T_B}{\partial t} = A_3 \left\{ q_3 + q_6 + B_{21}^{+i} \epsilon_b \sigma T_A^4 + B_{23}^{+i} \epsilon_b \sigma T_E^4 - \left[(1 - B_{22}^{+i}) \epsilon_b + \epsilon_f \right] \sigma T_B^4 \right\} \quad (14)$$

and the temperature of the body-mounted solar panel is given by:

$$m_{\text{DCD}} \frac{\partial T_D}{\partial t} = A_{\text{bm},6} \left[q_6 + \frac{k_{\text{bm}}}{t_{\text{bm}}} (T_A - T_D) - \epsilon_f \sigma T_D^4 \right] \quad (15)$$

Additionally, the temperature of the MLI in +X side is given by:

$$m_E c_E \frac{\partial T_E}{\partial t} = A_{m,1} \left[q_1 + B_{31}^{+i} \epsilon_m \sigma T_A^4 + B_{32}^{+i} \epsilon_m \sigma T_E^4 + \frac{k_{m,1}}{t_{m,1}} (T_A - T_E) - (1 - B_{33}^{+i}) \epsilon_m \sigma T_E^4 \right] \quad (16)$$

Finally, the lumped temperature of the spacecraft bus is given by:

$$m_A c_A \frac{\partial T_A}{\partial t} = \sum_{i:r} Q_i + Q_{in} + A_{r,1} \epsilon_r \sigma (B_{11}^{+i} T_A^4 + B_{12}^{+i} T_B^4 + B_{13}^{+i} T_E^4) + A_{r,2} \epsilon_r \sigma (B_{11}^{-i} T_A^4 + B_{12}^{-i} T_C^4 + B_{13}^{-i} T_F^4) + G_{bm} (T_D - T_A) + \sum_{j:m} G_{m,j} (T_j - T_A) - A_r \epsilon_r \sigma T_A^4 \quad (17)$$

In the above equations, all temperatures are lumped temperatures. The system of coupled first-order differential equations is solved using the Euler's method [27].

2.3. FEA Model Details

The satellite FEA model is created in COMSOL based on the computer-aided design (CAD) geometry presented in Figure 6. In this geometry, all electronic components are modeled as cubes, whereas the propulsion unit and the antenna are modeled as a cylinder and a disk, respectively. The mesh, containing 61,617 tetrahedral elements, is generated using COMSOL's physics-controlled mesh tool. This 3-D model is utilized in a transient thermal analysis with conduction and radiation heat transfer modes, where perfect thermal contacts between surfaces are assumed. The relative tolerance of the solver is 0.01 and the time steps are dynamically controlled. Furthermore, view factors are computed using the hemicube method, while setting the ambient temperature at 4 K.

In the model, externally absorbed heat fluxes identified through the lumped analysis are defined as inward heat flux inputs on each external surface, and the radiation boundary condition is specified for all surfaces. The thermo-optical properties and the material properties used in the analysis are presented in Tables 3 and 4, respectively. The heat dissipation values of the electronic components are defined as volumetric heat generation on each part, where the heat generation values of generic electronic components are distributed randomly. The internal heat generation values of the PCPU and the battery are calculated by user-defined functions (UDFs) as explained in Section 2.2. For the WCC analysis, heaters on certain components are also considered where UDFs are utilized to simulate the on-off controlled heaters.

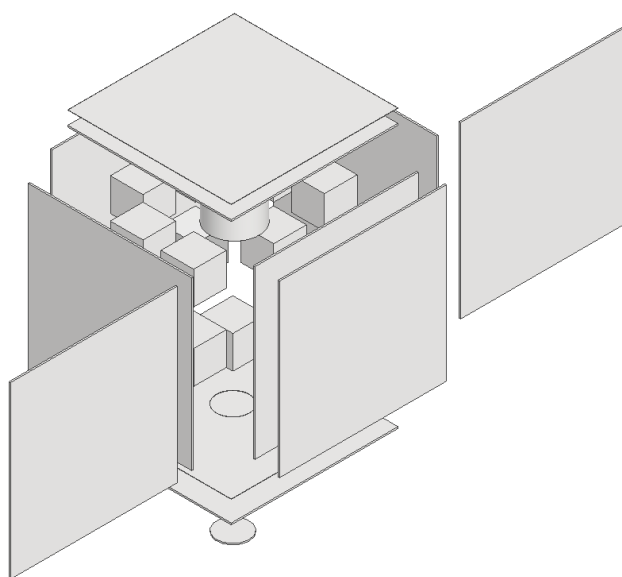


Figure 6. Exploded CAD view of the satellite.

The analysis is controlled from MATLAB with LiveLink. Based on the coordinates assigned by the optimization algorithm, radiators and MLI blankets on the three panels are created as prisms autonomously in the COMSOL CAD tool. Finally, data on time-dependent heater power consumption are extracted from COMSOL and integrated using the trapezoidal rule to obtain the energy.

Table 3. Surface thermo-optical properties.

Surface Property	WHC	WCC
$\alpha_{\text{MLI blanket}}$	0.3	0.4
$\epsilon_{\text{MLI blanket}}$		0.6
α_{radiator}		0.15
$\epsilon_{\text{radiator}}$		0.89
$\epsilon_{\text{structural panel interior}}$		0.03
$\epsilon_{\text{electronic components}}$		0.4
$\alpha_{\text{solar panel front side}}$		0.64
$\epsilon_{\text{solar panel front side}}$		0.82
$\alpha_{\text{solar panel rear side}}$		0.2
$\epsilon_{\text{solar panel rear side}}$		0.2

Table 4. Material properties used in the FEM model.

Material	ρ (kg/m ³)	c_p (J/kgK)	k (W/mK)
Structural panels	1600	900	130
Electronic components	1500	900	130
MLI blankets	600	600	$10^{-5} (k_{\perp}), 5 (k_{\parallel})$
Propulsion unit	2640	1200	5
Deployable solar panels	1600	900	130
Body-mounted solar panel	1600	900	$2 \times 10^{-8} (k_{\perp}), 130 (k_{\parallel})$
Radiator and antenna	2700	900	130

3. Results and Discussion

3.1. Verification

Prior to the optimization study, the external irradiations on six different panels of the satellite at the summer solstice were calculated based on the methodology explained in Section 2.2. In order to verify the accuracy of the calculations, irradiation results are compared with those obtained using COMSOL. As presented in Figure 7, there is no significant discrepancy between the two results. On the +Y panel, there is a maximum of 11 W/m² difference between the two results. This originates from the fact that our method assumes that the reflection of solar heat flux from the Earth surface is entirely diffuse, whereas, in a more realistic view, it is between diffuse and specular. However, considering that only a part of the irradiation is absorbed and that the entire panel is only a quarter square meter in area, the externally absorbed power predicted by the two methods will not differ significantly.

A further verification study is conducted using [28] as a reference, where the lumped temperature of a CubeSat is studied. In that study, a polar circular orbit with zero Beta angle was assumed. To match this, eccentricity, inclination, RAAN and epoch date are set to 0, 90°, 0°, and 22 September, respectively, in STK. The time-dependent temperature profile is obtained from the article using a plot digitizer tool. As presented in Figure 8, results of the reference study are reproduced successfully using the current methodology. Now that the methods have been verified, external heat flux values calculated based on

STK and MATLAB can be used in the optimization study, and the time-dependent lumped temperatures can be used to detect WHC and WCC.

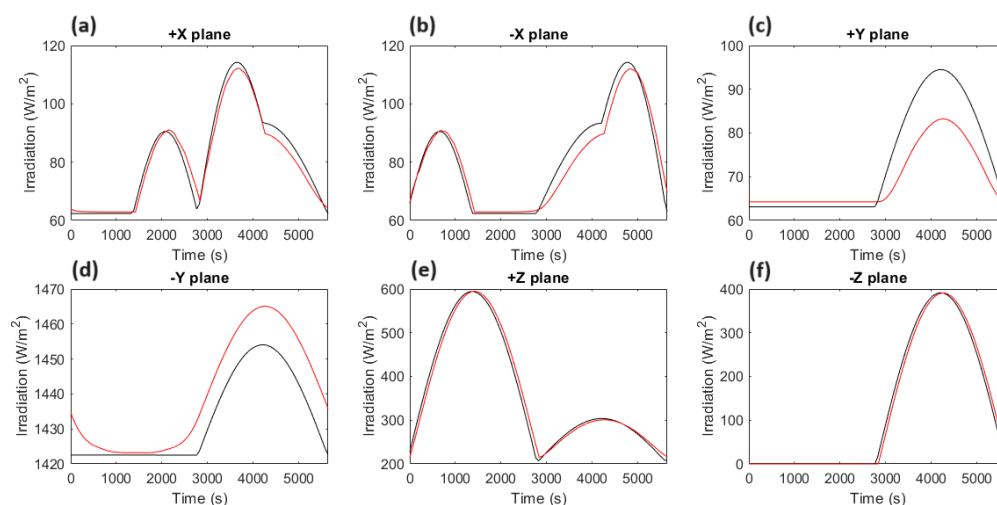


Figure 7. Comparison of the surface irradiances as calculated based on the current methodology (black curves) and as calculated by COMSOL (red curves) due to external sources on (a) +X, (b) −X, (c) +Y, (d) −Y, (e) +Z, and (f) −Z panels of the satellite.

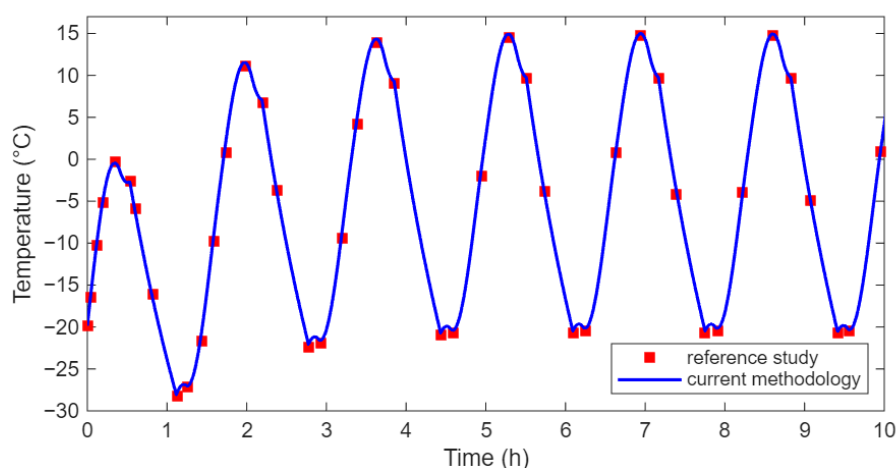


Figure 8. Comparison of the time-dependent temperature profile obtained from the current methodology in comparison to [28].

3.2. Optimization

The main objective of this study is to minimize the heater energy consumption of the satellite in the WCC, under the constraints imposed by the temperature intervals of the battery and the propulsion tank. The objective function is given as follows:

$$F(\chi) = \int_0^{t_{\text{end}}} Q_{\text{heater}}(t) dt \quad (18)$$

where Q_{heater} is the heater power consumption during the WCC period, t_{end} denotes the 12-periods duration, and $\chi = [X_{\text{left}}, X_{\text{right}}, \dots, Y_{\text{down}}, Y_{\text{up}}]^T$ is the vector of design variables, which are the coordinates of the corners of the three radiators. On the other hand, the constraints can be formulated as follows:

$$\begin{bmatrix} T_{\text{battery, WCC}} \\ T_{\text{propulsion tank, WCC}} \end{bmatrix} > \begin{bmatrix} 0^\circ\text{C} \\ 5^\circ\text{C} \end{bmatrix} \quad (19)$$

$$\begin{bmatrix} T_{\text{battery, WHC}} \\ T_{\text{propulsion tank, WHC}} \end{bmatrix} < \begin{bmatrix} 40^\circ\text{C} \\ 45^\circ\text{C} \end{bmatrix} \quad (20)$$

Bayesian optimization algorithm in MATLAB is utilized, which is trained with 20 randomly sampled initial data points. Expected improvement plus acquisition function is preferred in this study. The acquisition function determines the next point in the input space to evaluate the objective function based on the exploration-exploitation trade-off, where the exploration ratio is 0.5. In this, the acquisition function takes into account the mean and deviation of the surrogate model, which is a Gaussian process with an automatic relevance determination (ARD) Matérn 5/2 kernel, representing the objective function. The coupled constraint is also modeled by a separate surrogate model using the same kernel.

Bayesian optimization is shown to be suitable for minimizing computationally expensive functions where the number of evaluations is limited to a few hundreds [29]. In this optimization study, 120 function evaluations are carried out, where each evaluation takes around 2 h. The variation in the minimum objective function as a function of the number of evaluations performed is presented in Figure 9. The figure illustrates the minimum value of the objective function. The first design point provides a lower objective value, yet it does not meet the constraints; thus, it does not continue as the best point. After that, the surrogate model predicts new design points that provide a lower objective value and also meet the constraints, where the stopping criteria is to reach 120 iterations.

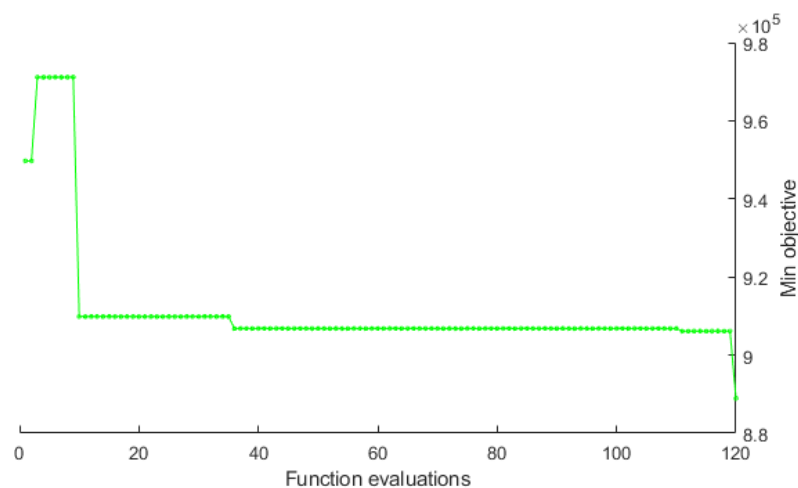


Figure 9. Tracking of Bayesian optimization.

On the other hand, Table 5 presents the design bounds and the set of optimal values for the coordinates of the corners of the radiators. It should be noted that each face of the satellite is a 50 cm by 50 cm square. In the table, the capital letter represents the panel on which the radiator is mounted. Additionally, subscripts “left” and “right” denote the distances of the radiator’s vertical edges from the left edge of the panel, whereas subscripts “down” and “up” denote the distances of the radiator’s horizontal edges from the lower edge of the panel. This notation is visualized in Figure 10.

The relationship between the total radiator area and the total heater energy consumption across different optimization iterations is shown in Figure 11 where the green and red colors, respectively, denote whether or not the constraints are satisfied. Contrary to expectations, there is no visible correlation between these two parameters. There are several iterations which yield roughly the same energy consumption employing varying radiator areas. Conversely, some iterations employ roughly the same radiator area, but the resulting heater energy consumptions are different. These observations mean that not only the

radiator area but also the placement of the radiators influence energy efficiency, considering the distribution of the heat-generating components on the panels.

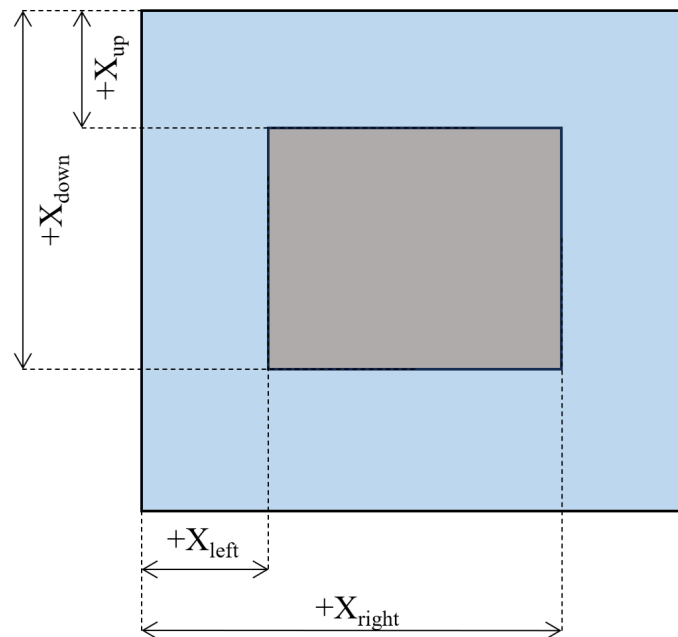


Figure 10. Demonstration of corner coordinates for the radiator mounted on the +X panel.

Table 5. Optimized coordinates of the corners of the radiators.

Variable	Lower Bound (mm)	Upper Bound (mm)	Optimum Value (mm)
+X _{left}	10	110	14.6
+X _{right}	390	490	470.6
+X _{down}	10	110	38.4
+X _{up}	390	490	443.6
−X _{left}	10	110	87.9
−X _{right}	390	490	466.8
−X _{down}	10	110	85.0
−X _{up}	390	490	464.4
+Y _{left}	10	110	70.6
+Y _{right}	390	490	480.3
+Y _{down}	10	110	76.1
+Y _{up}	390	490	436.6

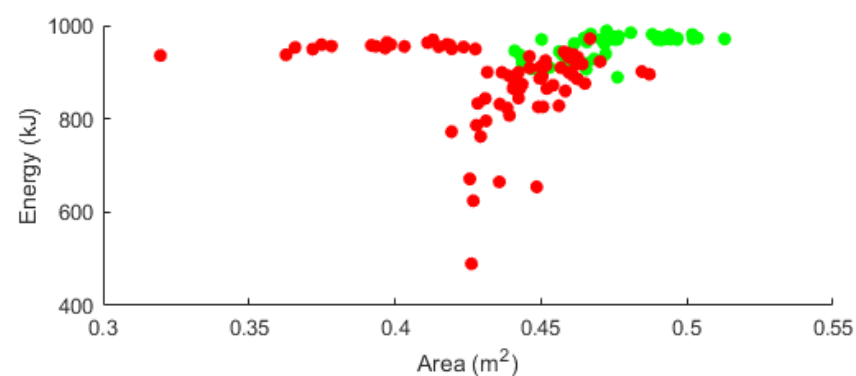


Figure 11. Relationship between heater energy consumption and radiator area. Red points: constraints are not met; Green points: constraints are met.

3.3. Lumped Analysis

As explained in Section 2.1, two lumped analyses are performed to only detect WHC and WCC, providing the external heat flux inputs corresponding to these instances for the FEA. As mentioned before, although the method and formulation could be extended to any small satellite, a satellite in SSO is considered in this study. The orbital analysis in STK utilized the Kepler parameters presented in Table 1. In this analysis, satellite ephemeris is generated using the High-Precision Orbit Propagator (HPOP), along with the EGM96 model to calculate gravitational perturbations. Furthermore, eclipses are calculated using the conical shadow model to account for penumbra and umbra regions. The result of this one-year analysis is utilized in both scenarios. It should also be noted that two altitude hold maneuvers are considered for 1 May and 29 August, providing ΔV of 6 m/s and 5 m/s, respectively.

In the lumped analysis, a total mass of 60 kg and an effective specific heat capacity of 940 J/kgK representing the satellite bus are utilized. Suitable parameters are also applied for the three solar panels and six MLI blankets, separately. Additionally, it is assumed that the Albedo coefficient, a , changes from 0.35 in WHC to 0.25 in WCC, whereas the planetary radiation intensity, I_e , changes from 236 W/m² in WHC to 211 W/m² in WCC.

In Figure 12, the number of epoch days to reach WHC and WCC across different iterations of the optimization is presented. Apparently, WHC remains almost constant throughout different iterations, whereas WCC changes appreciably. Another noteworthy point is that both extreme cases occur at rather arbitrary dates, not at the winter or summer solstices, which are April 22 and June 30 for WHC and WCC, respectively. This proves that lumped analyses are indeed necessary to find the actual extreme cases.

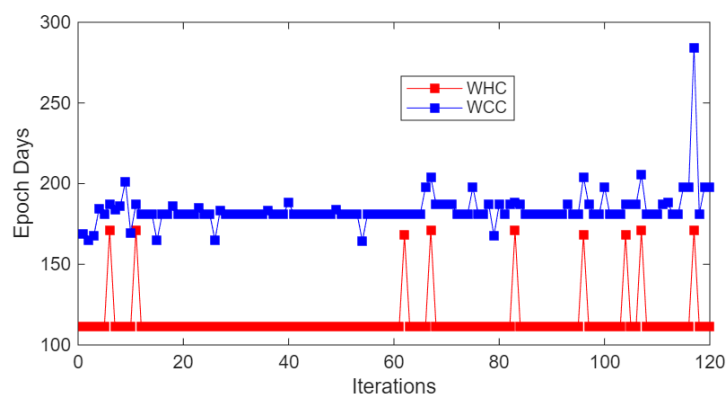


Figure 12. The change of epoch days to reach WHC and WCC based on the results of the lumped analysis from iteration to iteration.

The total external thermal loads and the resulting satellite lumped temperature profiles in the optimized case for both WHC and WCC are presented in Figure 13 for a duration of an entire year. It is evident that the lumped temperature of the satellite changes only a few degrees of Celsius within a particular scenario, whereas it changes significantly from one scenario to another. In WHC, the transponder's duty cycle causes larger temperature oscillations, while in WCC, with the transponder off, temperature oscillations result only from orbital variations in external heat fluxes.

Similarly, the lumped temperature of the body-mounted solar panel is given in Figure 14 along with the total generated power. Obviously, the temperatures are similar in both cases, independent of the lumped satellite temperature, which is due to the conductive isolation. Another notable point is the occurrence of short eclipses during a specific period of the year, caused by the changing Beta angle. During this period, the solar panel temperatures oscillate rapidly due to their small thermal mass. Although the solar panel temperatures in both scenarios are similar, the electrical power generated by the

solar panels differs significantly. The power depends on the cell efficiency as presented in Equation (13), which is expressed separately in WHC (EOL) and WCC (BOL) as follows:

$$\eta_{c,WHC} = 0.44 - 4.53 \times 10^{-4}T \quad (21)$$

$$\eta_{c,WCC} = 0.465 - 4.5 \times 10^{-4}T \quad (22)$$

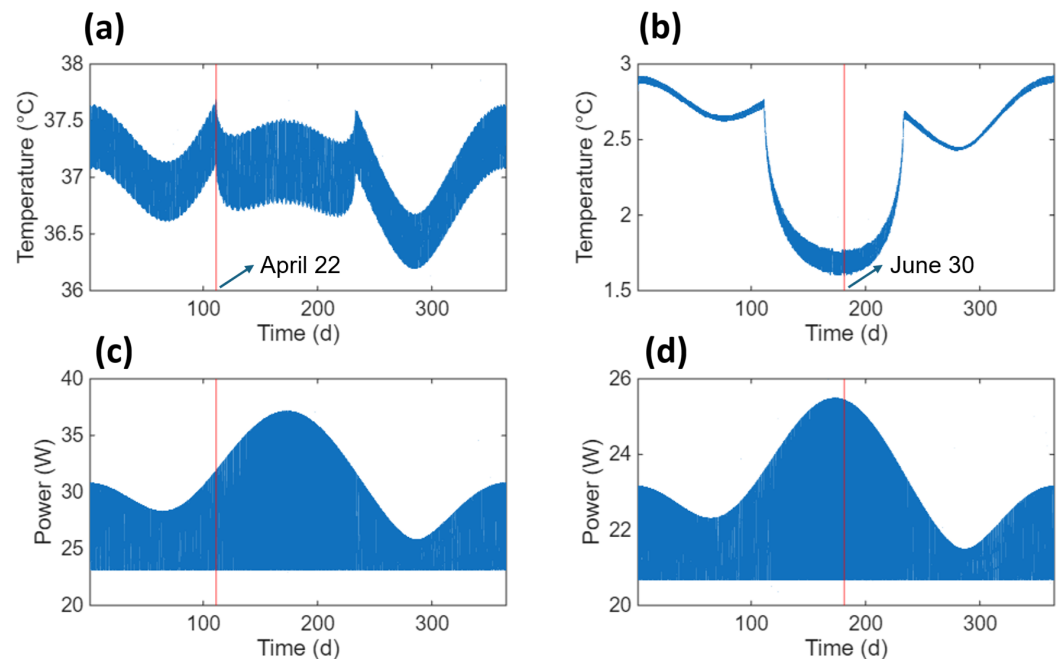


Figure 13. Lumped analysis results for satellite temperature in (a) WHC and (b) WCC, and total external thermal load in (c) WHC and (d) WCC.

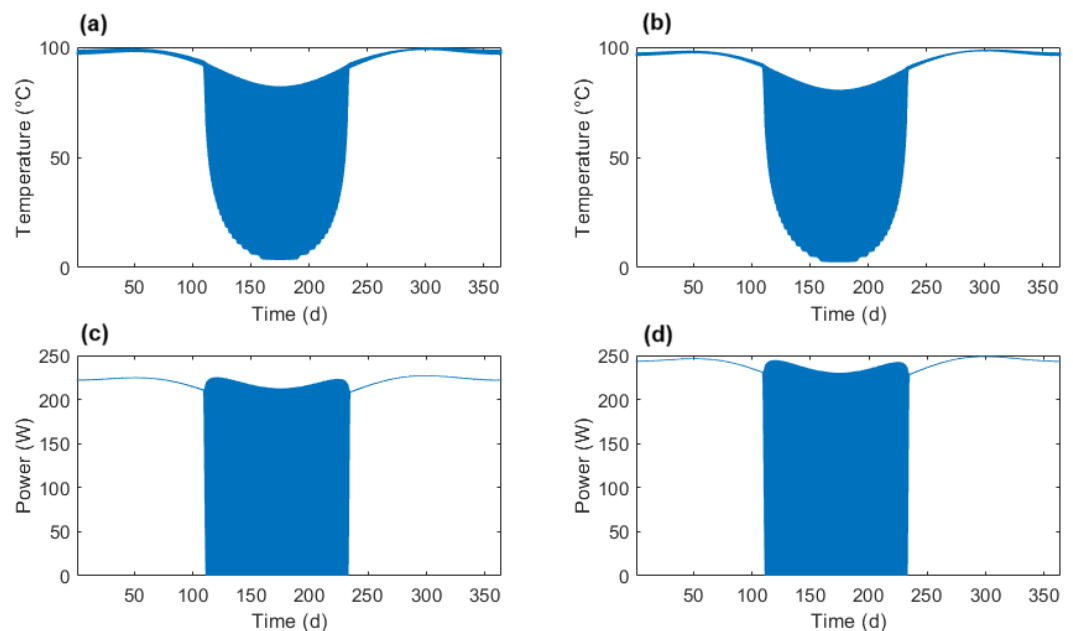


Figure 14. Lumped analysis results for body-mounted solar panel temperature in (a) WHC and (b) WCC, and total generated power in (c) WHC and (d) WCC.

3.4. Finite Element Analysis

External heat flux values identified through the lumped analysis are given as inputs into the FEA. On the other hand, battery heat dissipation, electrical power generation,

and heater power consumption values are recalculated within the FEA by UDFs. Time-dependent values of battery heat dissipation and total internal heat generation in the optimized case for both scenarios identified through the FEA are illustrated in Figure 15. As evident from the figure, the total internal heat generation profile in the WHC follows a pulsed trend due to the transponder's duty cycle. It is assumed that the transponder is active five times per revolution, with the on-state corresponding to around 26% of its duty cycle. Nevertheless, in WCC, the transponder is off and the total internal heat generation is much smaller. Furthermore, it is assumed that η_{PCDU} , $\eta_{b,c}$, and $\eta_{b,d}$ are equal to 0.9, 0.91, and 0.95, respectively.

Temperatures of battery, propulsion unit and solar cells as well as the average temperature of the satellite body in the optimized case, are depicted in Figure 16. Starting from a 20 °C initial condition, periodic steady state has been reached within the 12 period duration. To test the mesh independence, the number of elements is increased from 61,617 to 142,576, where the results for WHC are presented in Figure 17. It is noteworthy that the average temperature of the front surface of the solar panels is regarded as the solar cell temperature. That temperature is utilized in the calculation of the electrical power generation and thus the battery heat dissipation through Equations (12), (13), (21) and (22). As also predicted in the lumped analysis, solar cell temperatures are subject to large swings due to their small thermal mass. Figure 16 shows that the temperature profiles stay within the safe limits. The battery temperature is successfully maintained over 0 °C by on-off heaters in WCC. Table 6 present the temperature intervals of several parts of the satellite. As a final remark, while lumped analysis is an effective method for identifying worst-case scenarios, FEA results show that it inevitably overestimates the temperature in WHC and underestimates it in WCC.

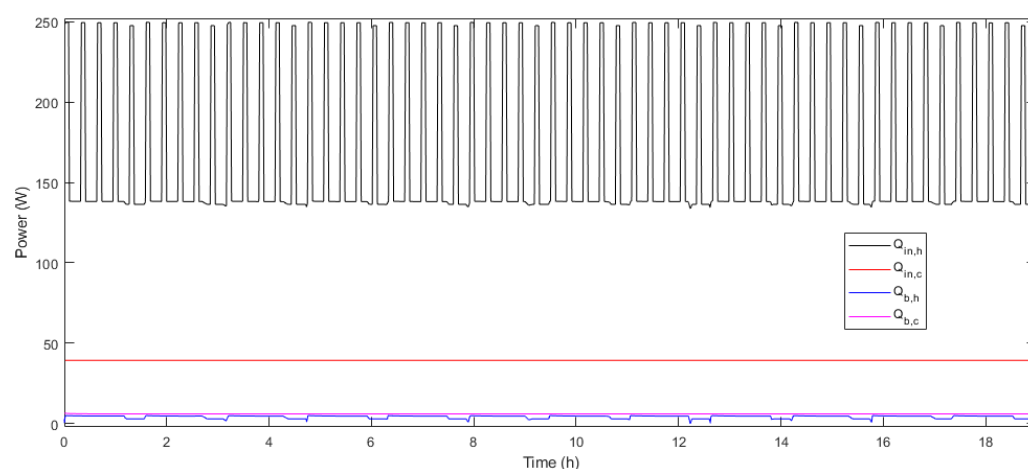


Figure 15. Internal heat generation values of battery and the entire satellite in both extreme cases.

Table 6. Minimum and maximum temperatures of several satellite parts, where “op” denotes operating, and “est” denotes estimated.

Satellite Part	$T_{\min, \text{op}}$ (°C)	$T_{\max, \text{op}}$ (°C)	$T_{\min, \text{est}}$ (°C)	$T_{\max, \text{est}}$ (°C)
Battery	0	40	3.68	34.6
Propulsion unit	5	45	16.5	23.5
Electronic components	−40	85	4.00	36.7
Solar cells	−40	100	−10.5	62.6
Structural panels	−80	150	3.96	38.6

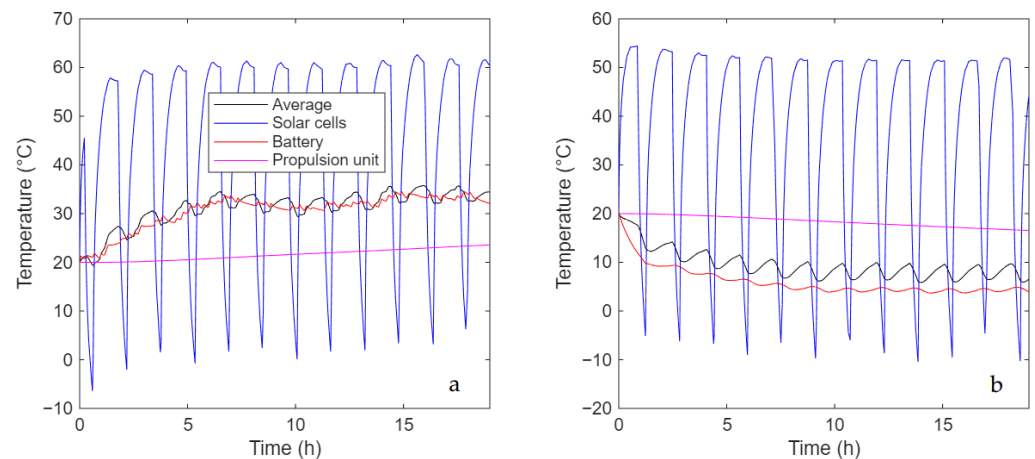


Figure 16. Temperature profiles of various entities modeled by FEA over time in (a) WHC and (b) WCC.

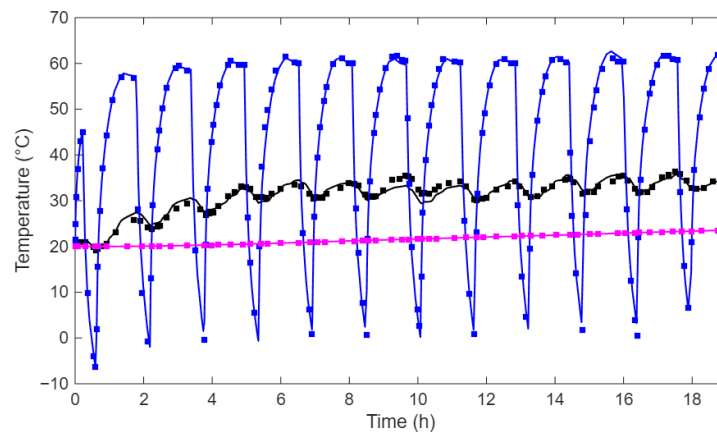


Figure 17. Testing of mesh independence in WHC, where the solid lines denote the original mesh and the squares denote the refined mesh. Blue indicates the solar cells, black indicates the average, and magenta indicates the propulsion unit.

4. Conclusions

Satellites should be maintained as close to room temperature as possible since most components of the satellite operate safely at these temperatures, where some components require more stringent thermal management compared to others. This is impossible without a thermal control subsystem, as the satellite is exposed to significantly varying external and internal thermal loads. In satellite thermal design, two of the possibly worst thermal cases are considered. In the design, a passive thermal control strategy, such as MLI blankets and radiators, is desirable; however, heaters are mostly utilized as well. It is desirable to minimize the heater power consumption and the radiator area considering power and mass budgets.

This study aimed to minimize the heater energy consumption under the constraints imposed by the operating temperature intervals of specific equipment. Bayesian optimization is utilized for the first time in a satellite thermal design study. The optimization algorithm determines the corner coordinates of rectangular radiators. On the modeling side, several analysis steps were assembled from a systems engineering perspective. First, an orbital analysis is performed to provide inputs for lumped analyses. Two lumped analyses are conducted, representing thermally extreme cases. These analyses provided the external heat flux inputs for finite element analyses. Additionally, in contrast to previous studies, the dates corresponding to the worst hot and cold cases are determined by these

two lumped analyses. Finally, two finite element analyses are performed, according to which the temperature profiles of several components as well as the necessary heater power are determined.

The results show that there is no considerable correlation between the radiator area and the heater energy consumption. That being said, not only the radiator area but also its location influences the energy efficiency. Additionally, the dates corresponding to the extreme thermal cases are affected by the radiator selection, and are not solstices. Consequently, it could be safely argued that Bayesian optimization method enabled solving a computationally expensive optimization problem and thus achieving reduced heater energy consumption while satisfying temperature constraints. The approach in this study could bring an integral and realistic perspective to satellite thermal control. When integrated into satellite research and development, this methodology could provide a time-efficient thermal design process, as well as energy and mass savings in satellite operation.

Author Contributions: Conceptualization, Y.M.K.; methodology, Y.M.K. and M.C.; software, Y.M.K.; validation, Y.M.K.; formal analysis, Y.M.K.; investigation, Y.M.K.; resources, M.C.; data curation, Y.M.K.; writing—original draft preparation, Y.M.K.; writing—review and editing, M.C.; visualization, Y.M.K.; supervision, M.C.; project administration, M.C.; funding acquisition, M.C. All authors have read and agreed to the published version of the manuscript.

Funding: This research received no external funding.

Institutional Review Board Statement: Not applicable.

Data Availability Statement: Data will be made available upon request.

Conflicts of Interest: The authors declare no conflicts of interest.

References

1. Gilmore, D.G. *Spacecraft Thermal Control Handbook*; The Aerospace Press: El Segundo, CA, USA, 2002; Volume 2.
2. Foster, I. *Small Satellite Thermal Modeling Guide*; Air Force Research Laboratory: Albuquerque, NM, USA, 2022.
3. Selvadurai, S.; Chandran, A.; Valentini, D.; Lamprecht, B. Passive Thermal Control Design Methods, Analysis, Comparison, and Evaluation for Micro and Nanosatellites Carrying Infrared Imager. *Appl. Sci.* **2022**, *12*, 2858. [\[CrossRef\]](#)
4. Shen, M.; Zhang, L.; Sun, Z.; Kong, L.; Liu, Y.; Xue, Z. Models, simulations, and applications of small satellite thermal analysis. *Adv. Space Res.* **2024**, *74*, 836–860. [\[CrossRef\]](#)
5. Yang, K.; Peabody, H.; Rivera, R. *Rapid Thermal Design, Modeling, and Analysis of Spaceflight Instruments*; NASA Goddard Space Flight Center: Greenbelt, MD, USA, 2022.
6. Boushon, K.E. Thermal Analysis and Control of Small Satellites in Low Earth Orbit. Master's Thesis, Missouri University of Science and Technology, Rolla, MO, USA, 2018.
7. Thurman, J.L. Optimization of Steady-State Thermal Design of Space Radiators. *J. Spacecr. Rocket.* **1969**, *6*, 1114–1119. [\[CrossRef\]](#)
8. Muraoka, I.; Galski, R.L.; De Sousa, F.L.; Ramos, F.M. Stochastic Spacecraft Thermal Design Optimization with Low Computational Cost. *J. Spacecr. Rocket.* **2006**, *43*, 1248–1257. [\[CrossRef\]](#)
9. Galski, R.L.; De Sousa, F.L.; Ramos, F.M.; Muraoka, I. Spacecraft Thermal Design with the Generalized Extremal Optimization Algorithm. *Inverse Probl. Sci. Eng.* **2007**, *15*, 61–75. [\[CrossRef\]](#)
10. Chari, N. Spacecraft Thermal Design Optimization. Ph.D. Thesis, University of Toronto, Toronto, ON, Canada, 2009.
11. Stout, K.D. Design Optimization of Thermal Paths in Spacecraft Systems. Master's Thesis, Massachusetts Institute of Technology, Cambridge, MA, USA, 2013.
12. Escobar, E.; Diaz, M.; Zagal, J.C. Evolutionary Design of a Satellite Thermal Control System: Real Experiments for a CubeSat Mission. *Appl. Therm. Eng.* **2016**, *105*, 490–500. [\[CrossRef\]](#)
13. Silva, D.F.d.; Muraoka, I.; Sousa, F.L.d.; Garcia, E.C. Multiobjective and Multicase Optimization of a Spacecraft Radiator. *J. Aerosp. Technol. Manag.* **2019**, *11*, e0518. [\[CrossRef\]](#)
14. Xiong, Y.; Guo, L.; Yang, Y.; Wang, H. Intelligent Sensitivity Analysis Framework Based on Machine Learning for Spacecraft Thermal Design. *Aerosp. Sci. Technol.* **2021**, *118*, 106927. [\[CrossRef\]](#)
15. Wang, T.; Zhang, L.; Zhang, F.; Lu, Y. Optimization strategy of active thermal control based on Kriging metamodel and many-objective evolutionary algorithm for spaceborne optical remote sensors. *Appl. Therm. Eng.* **2024**, *242*, 122494. [\[CrossRef\]](#)

16. Bai, T.; Zhang, Y.; Kong, L.; Ao, H.; Yu, J.; Zhang, L. Thermal Optimization Design for a Small Flat-Panel Synthetic Aperture Radar Satellite. *Aerospace* **2024**, *11*, 982. [\[CrossRef\]](#)
17. Cannon, J.R.; Havey, K.N.; Housley, N.S.; Mulford, R.B.; Iverson, B.D. Internally stowed, radially deployed radiator panels for passive CubeSat thermal control. *Appl. Therm. Eng.* **2025**, *270*, 126281. [\[CrossRef\]](#)
18. Chen, Y.; Zhao, T.; Geng, C.; Chang, Y.; Lu, J.; Zhao, Q.; Chen, Y.; Ouyang, H.; Dou, S.; Gu, J.; et al. Adaptive thermal radiation design for spacecraft heat dissipation. *Results Eng.* **2025**, *27*, 106262. [\[CrossRef\]](#)
19. Tribak, F.; Bouzem, A.; Bendaou, O. Optimal thermal management and reliability analysis of a 1U CubeSat in low earth orbit: A comprehensive study on beta angle influence and multi-objective optimization. *J. Mech. Sci. Technol.* **2026**, *40*, 1427–1444. [\[CrossRef\]](#)
20. Shahriari, B.; Swersky, K.; Wang, Z.; Adams, R.P.; De Freitas, N. Taking the Human Out of the Loop: A Review of Bayesian Optimization. *Proc. IEEE* **2015**, *104*, 148–175. [\[CrossRef\]](#)
21. Sutheesh, P.; Chollackal, A. Thermal Performance of Multilayer Insulation: A Review. In *Proceedings of the IOP Conference Series: Materials Science and Engineering*; IOP Publishing: Bristol, UK, 2018; Volume 396, p. 012061.
22. Gebhart, B. Surface Temperature Calculations in Radiant Surroundings of Arbitrary Complexity—For Gray, Diffuse Radiation. *Int. J. Heat Mass Transf.* **1961**, *3*, 341–346. [\[CrossRef\]](#)
23. Ertürk, H.; Howell, J. Monte Carlo Methods for Radiative Transfer. In *Handbook of Thermal Science and Engineering*; Springer International Publishing: Cham, Switzerland, 2017; pp. 1–43.
24. Martinez, I. *Radiative View Factors*; Universidad Politécnica de Madrid: Madrid, Spain, 2015.
25. Halvorson, M.; Cho, J.; Farkas, S.; Boyd, L.; Cartier, N.; Cole, K.; DeAngelo, N.; Edwards, T.; Middleton, T.; Moquin, C.; et al. High-Fidelity Spacecraft Thermal Modeling: Synthesis of STK, SPENVIS, MATLAB, Simulink, and Thermal Desktop using Model-Based Systems Engineering. In *Proceedings of the 36th Annual Satellite Conference*, Logan, Utah, 6–11 August 2022.
26. Kim, H.K.; Han, C.Y. Analytical and Numerical Approaches of a Solar Array Thermal Analysis in a Low-Earth Orbit Satellite. *Adv. Space Res.* **2010**, *46*, 1427–1439. [\[CrossRef\]](#)
27. Mathews, J.H.; Fink, K.D. *Numerical Methods Using MATLAB*; Pearson Prentice Hall: Upper Saddle River, NJ, USA, 2004; Volume 4.
28. Garzón, A.; Villanueva, Y.A. Thermal Analysis of Satellite Libertad 2: A Guide to Cubesat Temperature Prediction. *J. Aerosp. Technol. Manag.* **2018**, *10*, e4918. [\[CrossRef\]](#)
29. Frazier, P.I. A Tutorial on Bayesian Optimization. *arXiv* **2018**, arXiv:1807.02811.

Disclaimer/Publisher’s Note: The statements, opinions and data contained in all publications are solely those of the individual author(s) and contributor(s) and not of MDPI and/or the editor(s). MDPI and/or the editor(s) disclaim responsibility for any injury to people or property resulting from any ideas, methods, instructions or products referred to in the content.