Thermal control analysis on a 6U CubeSat equipped with a high-power laser

Bárbara M. Franco, Guilherme L. Dornelles, Luis F. O. Riva, Jhônatas H. Garcia, Djulia L. Guarnieri, et al.

Bárbara M. Franco, Guilherme L. Dornelles, Luis F. O. Riva, Jhônatas H. Garcia, Djulia L. Guarnieri, Kleber V. Paiva, Gary B. Hughes, "Thermal control analysis on a 6U CubeSat equipped with a high-power laser," Proc. SPIE 10769, CubeSats and NanoSats for Remote Sensing II, 107690T (18 September 2018); doi: 10.1117/12.2320750

Event: SPIE Optical Engineering + Applications, 2018, San Diego, California, United States

Thermal control analysis on a 6U CubeSat equipped with a high-power laser

Bárbara M. Franco*[a], Guilherme L. Dornelles[a], Luis F. O. Riva [a], Jhônatas H. Garcia [a], Djulia L. Guarnieri [a], Kleber V. Paiva [a], Gary B. Hughes[b]

[a] Thermal Fluid Flow Group – T2F, Department of Mobility Engineering, Federal University of Santa Catarina, Joinville, SC, Brazil

[b] California Polytechnic State University, Statistics Department, 25-107G, 1 Grand Avenue, San Luis Obispo, CA 93407-0405

ABSTRACT

A satellite in orbit is exposed to external radiation from space, including: solar radiation, reflected light and earth emission. Heat is also generated through its internal components. Thermal design considerations are necessary to prevent any component from being damaged by excessive heating and to guarantee its performance. This paper explores options for thermal control mechanisms onboard a 6U CubeSat that includes a high-power laser. The CubeSat mission represents a space environment demonstration of a NASA Innovative Advanced Concepts (NIAC) project for determining asteroid composition. A Low Earth Orbit (LEO) CubeSat mission is simulated at 650 km altitude, and the sensor system will determine through spectrometric detection the composition of a cold target heated by an on-board laser. An intuitive interface was created using MATLAB's tool GUIDE, where orbit, altitude and satellite position are input parameters that result in the average temperature at each orbit point, presented on a globe. The challenge is to find devices capable of controlling the temperature on the surfaces of the nanosatellite, and to be efficient enough to handle the heat generated by the laser when it is activated, requiring dissipation and expulsion of approximately 200 W. Mechanical designs tested include thermal straps, heat pipes, passive radiative surfaces and active schemes such as thermal louvres. Based on thermal gradients from analytical simulations and state-of-the-art in CubeSat thermal control, some of the tested thermal control schemes maintain the equipment within its working temperature range. The viability of selected thermal control schemes is discussed in this article.

Keywords: nanosatellite, laser, analytical analyses, thermal control

1. INTRODUCTION

An international collaboration between the Federal University of Santa Catarina - Brazil and the Doctor Gary B. Hughes, professor at California Polytechnic State University, United States was established. Students are designing conceptual satellite subsystem for the Nasa Innovative Advanced Concepts Program project 15-NIAC16A-014. The mission in this paper is part of NIAC program and is called molecular composition analysis of distant targets. The goal of the mission is to probe the molecular composition of some solar system targets, like asteroids, planets and even others further away targets. A laser beam will heat up a certain point of the target, melting it and forming a cloud of molecules from which will be made the analysis through spectrometer detection.[5] This concept is to be tested aboard of a 6U Cubesat to reach Technology Readiness Level 5 (TRL-5) with an experiment in space environment in Low Earth Orbit (LEO).

Sustain satellite temperature controlled is essential for electronic components operation and consequently the mission success. Due to the absence of atmosphere, space environment compromises satellites operation for being exposed to extreme conditions. Defining temperature distribution in orbit allows one to choose the correct thermal control devices, avoiding high temperature gradients.[1] Thus, the main goal of the thermal control system is to analyze the satellite temperature of the structure and components, design thermal control mechanisms and testing the proposed solution.

The orbit and satellite position are the main parameters that define the temperature distribution among satellite surfaces. Wherefore variables like inclination, eccentricity and the apogee argument are orbital characteristics that will define the thermal loads applied to the satellite.

CubeSats and NanoSats for Remote Sensing II, edited by Thomas S. Pagano, Charles D. Norton, Proc. of SPIE Vol. 10769, 107690T · © 2018 SPIE
CCC code: 0277-786X/18/$18 · doi: 10.1117/12.2320750

Small satellites were developed to facilitate space experiments in terms of reducing size, cost and development time, increasing the variety of mission applications.[2] However, their reduce size requires an efficient system that satisfies the mission energy requirements.[3] Add to this, energy consumption is compromised when the satellite is in a Low Earth Orbit (LEO), by the fact that it is submitted to a shadow phase in the opposite side of the Earth.[4]

In the present study, a simplified thermal analytical model of a nanosatellite and its components is proposed in order to understand the temperature distribution during active and non-active period of the laser beam. The nanosatellite is designed for a circular and near sun synchronous Low Earth Orbit. In addition, different thermal control mechanisms are listed in a conceptual way as an option for future mission. A mathematical model was developed in MATLAB software for discreet regions, where thermal gradients can be neglected and are represented by nodes, which are characterized by temperature, thermal capacity and by radiation with the surrounding nodes. The objective of this study is to take applicable procedures to ensure all the components will operate in their safe range of temperatures as listed in Table 1.

Table 1. Operating temperature of the nanosatellite's components.

Component	**Operating Temperature range**
ADCS	-10 to 60 °C
Antenna system	-20 to 60 °C
Batteries	-30 to 80 °C -60 to 120 °C
Crystal space P1U "Vasik"	-40 to 85 °C
CubeSence	-10 to 70 °C
Fiber coupled diode laser	20 to 35 °C
Micro propulsion system	-20 to 70 °C
On board computer	-25 to 65 °C
Solar panels	-40 to 125 °C
Spectrometer	-20 to 40 °C
Structure	-40 to 80 °C
Transceiver	-20 to 60 °C

2. CUBESAT CONFIGURATION

2.1 The Cubesat

A Cubesat is a small satellite that consists in an arrangement units (U) of 10 cm by 10 cm. Each unit weighs about 1 to 1.33 kg. The spacecraft studied in this paper is a 6U Cubesat composed by a metallic structure, containing the necessary components to remain in orbit for 3 to 6 months. Solar panels are attached on its external faces, whose emissivity and absorptivity are 0.7. The conceptual design is indicated in Figure 1. Nanosatellites standard specifications have restricted limits in terms of size, mass, mission duration and power consumption. The structure is made of 1 mm thick aluminum plate and its faces are covered with 0.55 mm thick solar panels, with effective area of 90% and efficiency of 20%. The frontal face shown in the Figure 1, is constantly turned to the sun, disregarding the spin. Its external dimensions are 20 x 30 x 10 cm and about 10 kg.

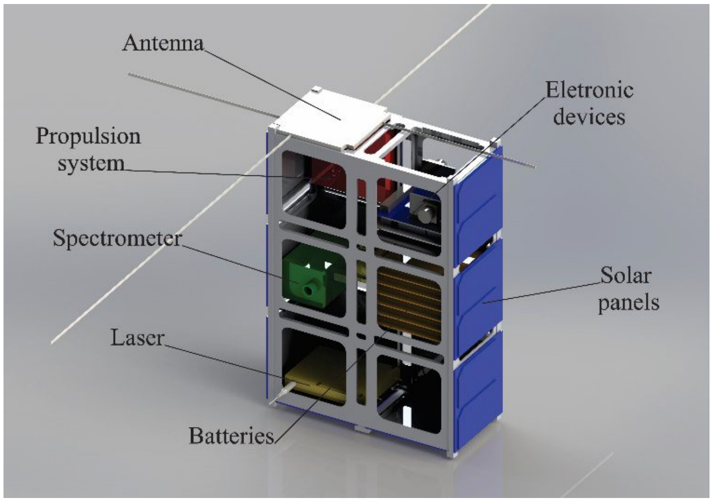

Figure 1. 6U Cubesat configuration.

The laser equipment is the most critical subsystem of the satellite in matters of energy. The Element® e18 Fiber Coupled Diode Laser, Figure 2, was chosen to be used in the CubeSat mission. It offers up to 220W output power from a 200µm fiber[7]. The laser dimensions, as well as the thermal, mechanical, electrical and optical properties were taken from the descriptive of the equipment, provided by the company website[7], as illustrated in Figure 3.

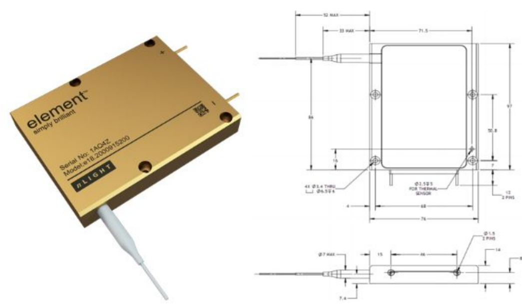

Figure 2. Element® e18 Fiber Coupled Diode Laser

	Units	Lower Spec	Typical	Upper Spec
Optical				
CW Output Power (in fiber)	W		220	
CW Output Power (as measured)	W	191	212	
Wavelength Centroid	nm	910.0	915.0	920.0
Spectral Width (FWHM)	nm		4.9	7.0
Power within 0.18 NA	%		95	
Fiber Core / Clad Diameter	µm		200 / 220	
Fiber NA / Index Type	-		0.22 NA / Step Index	
Electrical				
Electrical-to-Optical Efficiency	%		46	
Threshold Current	A		0.7	
Operating Current	A		15.0	15.0
Operating Voltage	V		31.8	33.4
Mechanical				
Mass	g		510	
Fiber Length	m	1.5	2	
Active Fiber Bend Radius	mm	35		
Fiber Jacketing	-		900 um Hytrel Loose Tube Buffer	
Fiber Termination	-		FPT	
Thermal				
Thermal Resistance	°C / W		0.2	
Waste Heat	W		257	
Operating (Housing) Temperature [2,3]	°C		+30	
Wavelength Temperature Coefficient	nm / °C		0.32	
Wavelength Current Coefficient	nm / A		1.1	

Figure 3. Laser descriptive.

2.2 Orbit model

Table 2 shows the values of the orbit model calculated according to the equations of Martinez[8] and the data provided by the NIAC project.

Table 2. Satellite's orbit data.

Altitude (H)	500 km
Relative altitude (H/R_E)	0,078
Inclination (i)	52º
Larger semi axis (a)	6871 km
Betha angle (β)	67º
Velocity (v)	7,6 km/s
Period (T_0)	5668 s
Relative period of eclipse (T_e/T_0)	0,31
Period of eclipse (T_e)	1757 s
Angle of entrance – eclipse (ϕ_{es})	2.16 rad
Angle of exit – eclipse (ϕ_{ee})	4.12 rad

3. CUBESAT THERMAL CONTROL MECHANISMS REVIEW

Thermal control mechanism are mechanical components whose function is directly involved in temperature control. As spacecraft are exposed to external radiation and internal heat from its components, the thermal design must guarantee a restrict temperature range. It can be divided in two groups, the passive and the active thermal control systems.

3.1 Passive thermal methods

Passive thermal control systems are the most recommended for use in CubeSats, because they are designed to control the energy flow with no input of power and they usually present low cost, low volume and weight, essential aspects for components in a nanosatellite.[9] Some of the commonly used devices are the following.

3.1.1 Thermal insulation

Thermal insulation and coating work as a barrier for radiation, blocking heat exchanges in the spacecraft. They are usually used in electronic devices and batteries in order to maintain their temperatures controlled. Multi-layer insulation (MLI) blankets are thermal insulators that consists in many layers of lightweight and reflective films. These layers are usually made of polyamide or polyester that are vapor deposited with aluminum. The MLI does not have the same efficiency in a nanosatellite as it has in a larger spacecraft. The optical characteristics of the surface material of the CubeSat, as the emittance and solar absorptance, can be altered with matte paint. The paint is a TRL 9 technology for CubeSats and it's the most common method. Black paint absorbs all the heat and the white paint limits the heat absorptance.

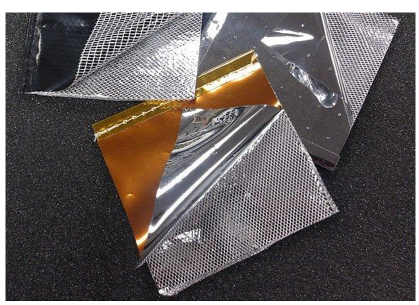

Figure 4. Multilayer insulation.

3.1.3 Thermal straps

Thermal straps are very convenient once they are passive system. The heat transfer mechanism is conduction and their length can be adjusted according to the design. The most common type of thermal straps is made of copper and aluminum. Other technologies employing different materials as graphite fiber have been studied. These thermal straps are known for their light weight, high thermal conductivity and efficiency. This technology hasn't been tested in small spacecraft platforms, but can still be applicable to small spacecraft. Hybrid thermal strap technologies are under development, e.g, graphite encapsulated in another material (usually aluminum) making the thermal straps ligther.[10]

Figure 5. Copper thermal strap.

3.1.4 Thermal Louvers

Thermal louvers are an old technology used to prevent components of larger spacecrafts from getting too hot or too cold. Despite the challenge of integrating this technology in small spacecrafts, Goddard Space Flight Center successfully created a miniaturized passive thermal louver that will be tested on a 6U CubeSat called Dellingr. This louver design consists of front and back plates, flaps and bimetallic springs. As the temperature in the CubeSat rises, bimetallic springs expands opening the flaps altering the thermal radiation of the exterior surface. When the springs cools down the flaps close. According to their results, this technology could dissipate 14.4W.[11]

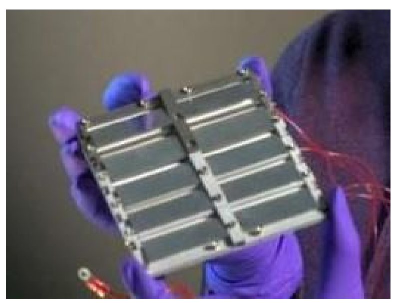

Figure 6. Cubesat thermal louver.

3.1.5 Heat pipes

Heat Pipes are components largely used in aerospace applications. Its operation consists in passively transport excess of heat from the hot source to the cold source by means of a two-phase system. The components of a Heat Pipe (HP) are: a capillary evaporator, condenser and an adiabatic section.[12] The liquid is transported through capillary structure, usually cylindrical, to the hot source of the system called evaporator. The fluid evaporates releasing the latent heat, changing its phase to vapor. Then, the vapor is directed to the condenser where it will condense and the cycle restarts (see Figure 7).

Figure 7. Heat pipe adapted for Cubesat. [13]

3.2 Active thermal methods

In an active thermal control method, power input is necessary. They are more accurate and more durable than passive systems. Their application in CubeSats are very limited due to the size and limited power.

3.2.1 Active thermal straps

An active thermal strap is a thermally conductive path that provides the mitigation of hot spots while they reduce the integration overload. Utilizing solid-state active thermoelectric devices (TEDs), they improve the performance of the thermal straps.[14]

3.2.2 Heaters

Heaters can be used to satisfy the minimal conditions of temperature during the cold cycles of the orbit. It consists in heating by Joule effect through an electrical resistance, distributing the heat to the components that are sensitive to cold. The temperature control is provided by a thermostat or temperature sensor.

3.2.3 Storage units

Although the technology is under development for Cubesat, the storage units can remove heat from a component in a phase change material (PCM) and then dissipate that heat over longer periods. By doing that, this mechanism is capable of reducing temperature peaks, protecting certain sensible components, maintaining a desired temperature or storing energy for future use. Due to the diversity of PCMs, it is possible to find one that meets the requirements of the project.

4. TEMPERATURE ANALYTICAL METHOD REVIEW

Thermal analysis is usually conducted by commercial software that links the computational fluid dynamics analyses (CFD) and the finite element method (FEM). Even with several settings and tools that provide detailed results, there are limitations in CFD when it is applied for space environment due to the rarefied atmosphere and the heat transfer phenomenon controlled by conduction and radiation. Furthermore, commercial software employed to satellite thermal control are usually high cost.

Analytical model can solve energy equation through nodal analyses.[15] Typical procedure uses the structure as a heat storage, improving the heat transfer among components through radiation and conduction and reducing the temperature range inside the satellite, where the most sensitive components are mounted.

The author Martinez[8], as an example, developed a thermal model of a nanosatellite in a circular orbit (300 km) around the moon. Analytically, the CubeSat is divided in 11 nodes. Each one of these nodes is characterized by an equation that considers heat sources acting in the structure and power radiated to the space or to the other faces of the satellite. The result showed that temperature behaves as a periodic function given by the initial condition of 300 K. Finally, the author concluded that the face that is subjected to the highest amount of heat must be covered with a low emissivity material, in order to minimize the load.

Bulut and Sozbir[2] present an analytical thermal analysis of a CubeSat in 98° of inclination. The analysis is made for different altitudes, varying from 500 to 2000 km and considering both extreme cases: hot case - satellite exposed to the maximum solar radiation and the cold case, where the cubesat is in eclipse in a large part of the orbit. The authors obtained the temperature distribution in the solar panels which varied from – 35 to 40 ° C. This temperature range can guarantee the proper function of the electronic devices, except for the batterie in the cold case. In this way, the authors recommended a more detailed analysis, increasing the number of nodes by face besides of coupling of a small heater to control the low temperatures on the batteries.

Tsai[16] used a general mathematical model to determine the thermal gradient of a satellite. He also affirmed that a simple analytical method does not guarantee a detailed analysis, which shows the importance of combining analytical and numerical methods when it comes to a detailed design.

5. METHODOLOGY

In this work a simplified analytical model is employed to calculate the temperature distribution of a 6U CubeSat. The analysis employed heat transfer equations[6] considering internal and external heat sources. Based in the method adopted by Martinez[8], the mathematical model is composed of a number of discrete regions where temperature gradients can be neglected and are represented by nodes. Each node is characterized by its temperature, thermal capacity, thermal dissipation, radiation interfaces and conduction with the surrounding nodes, resulting in one equation for each node. As the nodal structure is complex, a few simplifications in order to guarantee convergence of the results was implemented, e.g., only analyzing the most critical nodes.

The thermal model will be evaluated in two extreme cases, considering the following hypotheses: heat transfer among the main components inside the satellite is computed. The laser beam was the only internal equipment modeled with internal generation due to its high heat transfer rate and orbit inclination is $\beta = 90°$.

- Case 1: without eclipse
- Case 2: with eclipse.

Satellite behavior in second case is shown in Figure 8, where it can be seen that node 1 is always pointing towards the Sun.

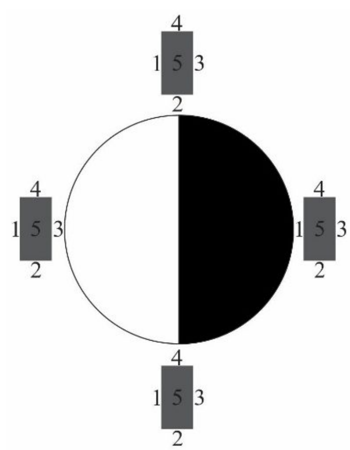

Figure 8. Orbit in the second case.

5.1 Heat transfer in the satellite

In a satellite, the energy balance related to a closed system (Equation 1) varies due to the temperature gradient, equation 3. [8] Energy conservation law stablishes that the amount of energy in an isolated system remains constant. This analysis is conducted in transient regime since the temperature distribution varies over a period of time.

$$\frac{dE}{dt} = \dot{Q} - \dot{W} \tag{1}$$

$$\frac{dE}{dt} = mc_p\frac{dT}{dt} \tag{2}$$

According to Lavine, Incropera and Bergman[6], heat transfer rate, $\dot{Q}$, refers to the liquid amount of energy transferred into the system by means of conduction and/or radiation. The work rate, $\dot{W}$, represents the net power energy transferred out of the system, in form of electrical energy, through the heaters and solar panels, electromagnetic energy, through the laser and antenna, or mechanical as friction.

As mentioned above, satellite temperature depends of the influence of the following heat sources: solar ($\dot{Q}_{\text{solar}}$), albedo ($\dot{Q}_{\text{albedo}}$), infrared ($\dot{Q}_{\text{IR}}$), radiation ($\dot{Q}_{\text{rad}}$) and components generation ($\dot{Q}_{\text{gen}}$). The term $\dot{Q}_{\text{out}}$ sums the dissipated energy from the satellite to the space environment. In this way, the energy balance is presented as:

$$mc_p\frac{dT}{dt} = \dot{Q}_{solar} + \dot{Q}_{albedo} + \dot{Q}_{IR} + \dot{Q}_{gen} - \dot{Q}_{out} \tag{3}$$

The solar heat flux is assumed constant, $\dot{Q}_s = 1367$ W/m², and depended of the projected area A_p. Part of the incident solar energy is reflected and the other absorbed by the satellite surface. Even though a solar parcel is transferred to the satellite, part of it generates electricity in the solar panels, $\eta F_p \dot{Q}_s A_{frontal}$. During eclipse period, $\dot{Q}$s = 0 and $\beta \leq \pi/2$, the view factor, Fe, is 0 and when is exposed to the sun is 1.

$$\dot{Q}_{solar} = \alpha A_p \dot{Q}_s F_e cos\beta \tag{4}$$

The albedo rate absorbed by the satellite is modeled as:

$$\dot{Q}_{albedo} = \alpha A_b F_{b,p} \alpha_{albedo} \dot{Q}_s F_a \tag{5}$$

where A_b is the area of the satellite base, $F_{b,p}$ the view factor and α_{albedo} the albedo coefficient regarding the solar radiation reflected by the Earth, value that varies according to the characteristics of Earth surface, e.g., vegetation, the presence of clouds, etc. Part of the albedo energy contributes to generate electricity in the solar panels: $\eta F_p \dot{Q}_s \alpha_{albedo} F_{b,p} A_{frontal}$. In this case, the coefficient used in this work was 0.26 for albedo and 257 W/m² for InfraRed[19]. For orbits over 320 km altitude, the position of the satellite, ϕ, impacts albedo coefficient, whose relation is given by the entrance angle, ϕ_{es}, view factor of eclipse, F_e, and the beta angle, β:

$$F_a = \left(\frac{1+cos\phi}{2}\right)^2\left[1-\left(\frac{\phi}{\phi_{es}}\right)^2\right]cos\beta F_e \tag{6}$$

Earth heat emission influences the temperature of the satellite as well, $\dot{Q}_p$. The satellite has itself a temperature, in this way it emits radiation to the space, $\dot{Q}_{out}$. As the space average temperature is small (T_∞= 2.7 K) it can be neglected.

$$\dot{Q}_p = \alpha A_b F_{b,p} \varepsilon_p \sigma T_p^4 \tag{7}$$

$$\dot{Q}_{out} = A_b \varepsilon_b \sigma T_b^4 \tag{8}$$

where b and p indicate the surface of the satellite and the Earth.

The view factor related to albedo and irradiation rate, $F_{b,p}$, is the parcel of radiation that leaves one surface and intercepts the other. Earth is a emitter source and can be approximate as a sphere, larger than the satellite. Therefore, according to Martinez [8], the calculation of the view factor can be expressed as:

$$F_{b,p} = \frac{1-\sqrt{1-\frac{1}{h^2}}}{2} \tag{9}$$

$$h = \frac{H+R_E}{R_E} \tag{10}$$

where R_E is the earth radius, 6371 km, and H the altitude of the mission.

In this paper, heat generation is computed due to the laser beam, $\dot{Q}_{gen}$.

Therefore, expanding the energy equation by means of thermodynamic and geometric parameters and neglecting the mechanical work,

$$mc_p \frac{dT}{dt} = \dot{Q}_{solar} F_e(\phi) + \dot{Q}_{albedo} F_a(\phi) + \dot{Q}_p + \dot{Q}_{gen} - \dot{Q}_{out} \tag{11}$$

$$mc_p \frac{dT}{dt} = \alpha A_p \dot{Q}_s cos\beta F_e(\phi) + \alpha A_b F_{b,p} \alpha_{albedo} \dot{Q}_s F_a(\phi) + \alpha A_b F_{b,p} \varepsilon_p \sigma T_p^4 - A_b \varepsilon_b \sigma T_b^4 + \dot{Q}_{gen} \tag{12}$$

From Equation 12, temperature distribution over the satellite along the orbit can be solved. An initial condition solution, 300 K, and a numerical method, Runge-Kutta, is employed. The transient decays until a periodic solution stablishes. This procedure is performed in Matlab software.

5.2 Multi nodal model

This method allows to consider the heat exchange also among parts of the satellite, which results in a more detailed analysis of the project. Nodes selection are conducted, mainly, regarding the contact with electronic devices, such as batteries. These nodes are the ones that must be analyzed more specifically because of their influence in the overall thermal system. The thermal energy balance of a generic node 'i' in a 'N' nodes discretization is described as:

$$C_i \frac{dT_i}{dt} = \sum \dot{Q}_{i,j,input} \tag{13}$$

where only heat sources are considered. Therefore, the most detailed energy balance in relation to the terms of heat and the discretization of time have the following form:

$$C_i \frac{T_i^+ - T_i}{\Delta t} = \sum_{j=0}^{N} \dot{Q}_{ij} = \dot{Q}_{diss,i} + \dot{Q}_{s,i} + \dot{Q}_{a,i} + \dot{Q}_{p,i} - \dot{Q}_{\infty,i} + \dot{Q}_{gen} + \sum_{j=1}^{N} \dot{Q}_{cond,ij} + \sum_{j=1}^{N} \dot{Q}_{rad,ij} \tag{14}$$

where C_i is the total heat capacity of the node 'i', T_i^+ and T_i are the temperatures of this node before and after advancing a Δt in time, respectively. The heat term due to the electric dissipation, $\dot{Q}_{diss,i}$, is input data; the heat generation term, $\dot{Q}_{gen}$, is obtained from the laser datasheet; and the term due to electromagnetic dissipation, $\dot{Q}_{s,i}$, is calculated in the form:

$$\dot{Q}_{s,i} = (\alpha_s - \eta F_p) E_s A_{frontal}(t) F_e(t) \tag{15}$$

For albedo, the following equation is used:

$$\dot{Q}_{a,i} = (\alpha_s - \eta F_p) \rho_p E_s A_i F_{i,p}(t) F_a(t) \tag{16}$$

where ρ_p is the albedo factor of the planet, $A_i F_{i,p}$ is the part of energy of the planet reflected in the area of the node 'i' and F_a is the albedo factor observed from the satellite, being equal to 1 when directly exposed to the sun light and 0 when it is in eclipse. The heat emitted by the near planet depends of the emissivity of the node 'i', ε_i, the emissivity of the planet, ε_p, the Stefan-Boltzmann constant σ and the temperature of the node 'i', T_i:

$$\dot{Q}_{p,i} = \varepsilon_i A_i F_{i,p}(t) \varepsilon_p \, \sigma T_i^4 \tag{17}$$

The heat transfer rate emitted by the satellite to the space is calculated disregarding the influence of the temperature of the space environment, T_∞. Therefore, we have:

$$\dot{Q}_{\infty,i} = \varepsilon_i A_i F_{i,\infty}(t) \, \sigma T_i^4 \tag{18}$$

Since the Sun and the Earth are not considered as objects that energy can be transferred to and only acts as heat sources, the view factor of the nodes 'i' related to the space are equal to 1, that is, $F_{i,\infty} = 1$.

The next step is to analyze the heat transfer among nodes of the satellite. The internal radiation summarized all internal heat sources that happens among the components of the satellite. In this case, the components act as heat sink from the external heat sources and as emitters and reflectors of heat to the surrounding components according to the radiation equations. Besides, all the eletronic components will have an electric inefficiency in a way that a fraction of the operating power is lost in the form of heat. This heat lost is constantly absorbed and emitted by the component. Assuming that all the surface heat dissipation is absorbed or reflected by the other surfaces of the satellite, the internal radiation can be modeled as a nodal network. The heat exchange by conduction among nodes can be described as:

$$\dot{Q}_{cond,i} = \sum_{j=1}^{N} \dot{Q}_{cond,ij} = \sum_{j=1}^{N} C_{ij}(T_j - T_i) = \sum_{j=1}^{N} \frac{k_{ij,eff} A_{ij,eff}}{L_{ij,eff}} (T_j - T_i) \tag{19}$$

where C_{ij} is the conductance between nodes that represents the thermal resistance to conduction; $k_{ij,eff}$ is the effective conductance of the material; $A_{ij,eff}$ is the effective area of the heat transference and $L_{ij,eff}$ is the effective distance between the analyzed nodes. The conductance parameter must be made manually, aside from the most commercial packages of thermal analyses, what means an additional computer resource. The heat exchange between nodes due to the radiation is computed by the thermal radiation resistance, R_{ij}, considering that all surfaces nodes are black bodies:

$$\dot{Q}_{rad,i} = \sum_{j=1}^{N} \dot{Q}_{rad,ij} = \sum_{j=1}^{N} \sigma R_{ij} \left(T_j^4 - T_i^4\right) = \sum_{j=1}^{N} \sigma A_i F_{ij} \left(T_j^4 - T_i^4\right) \tag{20}$$

Thus, the equation that represents the heat transfer in each node is given by:

$$C_i \frac{dT_i}{dt} = \dot{Q}_{s,i} + \dot{Q}_{a,i} + \dot{Q}_{p,i} + \dot{Q}_{gen} + C_{ji}(T_j - T_i) + \sigma A_i F_{i,j}(T_j^4 - T_i^4) - \varepsilon_i A_i \sigma T_i^4 \tag{21}$$

From an initial value of each node and a given time period, in order to find the temperature distribution of the satellite, equations above are solved.

6. RESULTS

In this section, the analytical results of the temperature on the external surfaces of the satellite are presented. In Figure 9, the CubeSat is divided in 6 nodes. By assuming infinite conductivity, it is possible only analyzed the interaction of the satellite with the space environment. In the multi nodal case, each face has a resulting equation to model the heat transfer. The Temperature verification of the surfaces in orbit will be made assuming that the laser operates for 10 seconds at each 10 minutes.

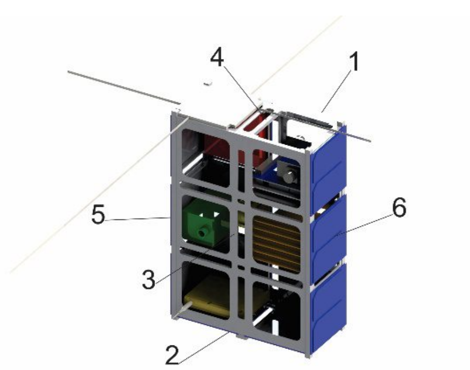

Figure 9. Configuration of nodes.

In Table 4, thermal and optical properties of the satellite are presented.

Table 4. Properties of the principal materials.[17]

	Aluminum	Solar Panels
Density [kg/m³]	2700	2300
Specific heat [J/kgK]	900	700
Emissivity	0,4	0,7
Thermal conductivity [W/mK]	130	1,03
Thermal capacitance [J/K]	24,3	8,05

In Figure 10, results for each node for the first case are illustrated. It can be observed that the maximum temperature for the satellite is 315.96 K (42.81 °C) on face 1 due to the its direction towards the Sun . One can observer that the influence of the laser on temperature distribution is significant. The laser activity increase the thermal field every 10 minutes when it is operating (10 s only), creating peaks along of the temperature distribution. According to Table 1, that presents the operation temperatures of the components manufacture, laser and spectrometer will exceed the maximum operating temperature at 7.8 °C and 2.81 ° C, respectively. Further analysis and solution for thermal control will be required.

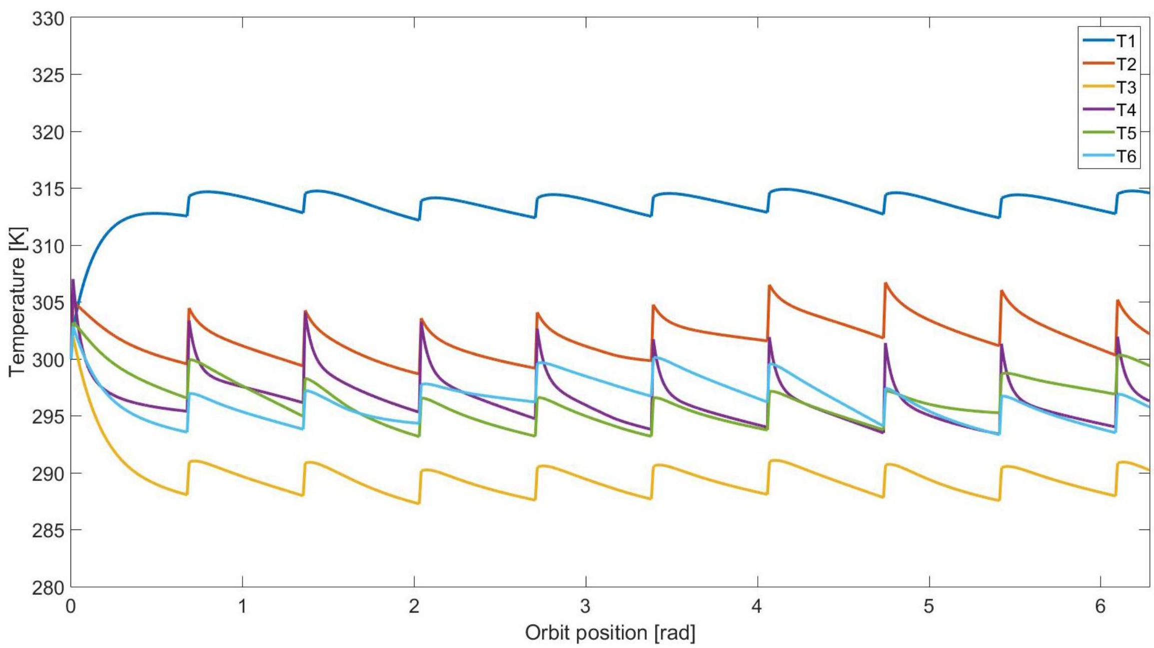

Figure 10. Nodal temperatures in the first case – without eclipse.

The temperature distribution of the second case is shown in Figure 10. It can be noted that the maximum temperature of the satellite is around 315 K for face 1 as well. Due to the eclipse period, temperature distribution has changed significantly. The lack of solar radiation contributes to the temperatures drop of the entire satellite. The minimal temperature is 274.75 K on face 3. Thus, analyzing the components operating range just the laser is below the minimal condition requiring a thermal control system.

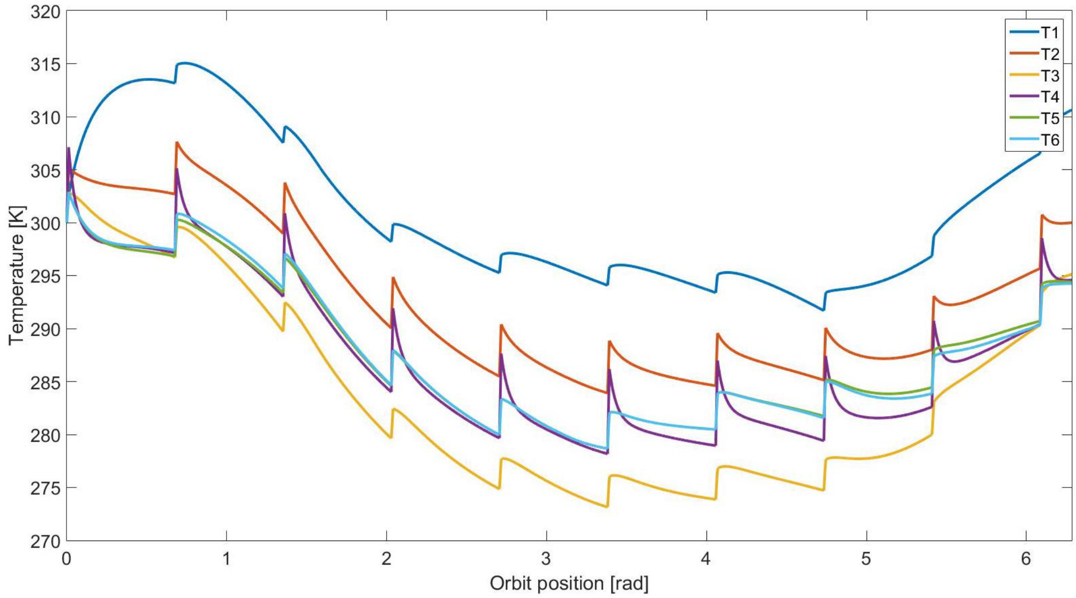

Figure 11. Nodal temperatures in the second case – with eclipse.

In both scenarios, with and without eclipse, a thermal control system will be required. Due to Cubesat dimension limitation, these solutions must be implemented without compromising the components of the satellite. The mechanism with lower impact is modifying the optical properties of the external surfaces not occupied by the solar panels, increasing the emissivity and keeping the solar absorptivity relatively low. In order to achieve this goal, the solutions identified are: surface treatments, such as oxidation, and surface paints. The first, however, is not feasible because it is complex and expensive. The second is the most common, using white painting, for instance, it can decrease the temperature by up to 30 °C.

Even modifying the external properties of the satellite, temperatures still have to be controlled. One solution could be heat pipes, although the use of these mechanisms for the satellite under study must be verified due to geometric restriction, their performance are quite efficient and has been employed on space mission since 1960.

For the cold case scenario, it is suggest using skin heaters on the critical equipment. By Joule effect it is possible to heat up the component temperature, avoiding its collapse.

7. CONCLUSION

This work analyzed the temperature distribution in a nanosatellite under NIAC project. In order to calculate the average temperature of the nanosatellite in orbit hypotheses were made. The results indicated that, despite the simplifications adopted, temperatures gradient are relevant and should be analyzed during the conceptual design phase of the satellite.

The results are preliminary, since some parameters are still unknown at the beginning of a satellite project. One of the objectives accomplished was to provide an initial tool for nanosatellites temperature determination.

Due to the high power demand of the laser system, it was concluded that it is necessary a thermal control mechanism mainly when the laser is active with or without eclipse. The authors suggest starting modifying the optical property of the satellite surface where there is no solar panel. Heat pipe and skin heater can be employed as well, since dimensions are not a restricted parameter. However, for results that are more accurate, the thermal design must be complemented and updated with real conditions. Although a specific study considering the numerical solution should be done, an analytical methodology can guarantee a preliminary temperature distribution in short period of time.

REFERENCES

[1] ABOUEL-FOTOUH, M. A. et al, "The effect of satellite orientation on satellite surface temperature distributions," Journal of Applied Sciences Research. Paper 2(12),1286-1292 (2006).

[2] BULUT, M., SOZBIR, N., "Analytical investigation of a nanosatellite panel surface temperatures for different altitudes and panel combinations," Journal El Selvier, Paper 75, 1076-1083 (2014).

[3] MAZARIAS, A. L., CLARK, C. S., "Power system challenges for small satellite missions," 4S Symposium Small Satellites Systems and Services, ESA SP-625, (2006).

[4] MOCANU, B., et al, "Determining optimal orbital path of a nanosatellite for efficient exploitation of the solar energy captured," First International Conference on Advances in Satellite and Space Communications, Bucharest, Romania (2009).

[5] HUGHES, G. B., LUBIN, P., "Molecular composition analysis of distant targets," NIAC Phase I, Final Report (2017).

[6] LAVINE, A., INCROPERA, F. D. D., BERGMAN, T., [Fundamentals of Heat and Mass Transfer], John Wiley & Sons, New York (2006).

[7] nLIGHT, "element® e18 Fiber Coupled Diode Laser," <http://www.nlight.net/products/diode-lasers/120-element-e18-Fiber-Coupled-Diode-Laser > (25 June 2018).

[8] MARTÍNEZ, I., "Spacecraft Thermal Modelling; Radiation view factors; Heat transfer and Thermal Radiation Modelling; STC systems, missions and needs," < http://webserver.dmt.upm.es/~isidoro/ > (10 June 2018).

[9] BATURKIN, V., "Micro-satellites thermal control – concepts and components," Acta Astronautica, Paper 55, 161-170 (2005).

[10] Aavid Thermacore, "Thermal Straps: Efficient Cooling without Structural Loading," < https://www.thermacore.com/products/thermal-straps.aspx > (15 July 2018).

[11] NASA, "State of the Art of Small Spacecraft Technology: thermal control," < https://sst-soa.arc.nasa.gov/07-thermal > (02 May 2018)

[12] RIEHL, Roger R., DUTRA, Thiago. "Development of an experimental loop heat pipe for application in future space missions," Applied Thermal Engineering, 25(1), 101-112 (2005).

[13] BROUWER, H., DE GROOT, Z., VAN GERNER, V. J., GUO, J., "Solving the Thermal Challenge in Power-Dense CubeSats with Water Heat Pipes," Space Science Conference, 2(06) (2017).

[14] Aerospace Structures Load Path, "Flexible and Enhanced Active Thermal Straps (FEATS)," (2015).

[15] TSUYOSHI, T. et al. Journal of Thermophysics and Heat Transfer. 28(3), 524-533 (2014).

[16] TSAI, J. Journal of Spacecraft and Rockets, 41(1), (2014).

[17] Gilmore, D. G., [Spacecraft Thermal Control Handbook: Fundamental technologies], 2nd ed, The Aerospace Corporation, California, (2002).