Development of analytical and numerical thermal models for cubsats

Giuseppe Landolfo
giuseppe.landolfo@tecnico.ulisboa.pt

Instituto Superior Técnico, Universidade de Lisboa, Portugal

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Abstract

Why do we ever invest more resources to go into space? Technological applications, well-being and mainly the curiosity of what is unknown push human beings to overcome their limits.

Every artificial object we are going to send into the space, it will move from a comfortable environment, Earth's atmosphere, to the hostile environment that is the blackness of space. The body must be able to endure the transfer and to be operating in the new environment.

The student team ECOSAT, that I joined, at the University of Victoria, is participating in the Canadian Satellite Design Challenge, a competition that involves designing and developing a 3U CubeSat. The nanosat *Homathko* will beam photometric light to observe the aerogel.

The aim of the work presented here is to study and ensure safe operation across the whole mission from a thermal point of view. A Finite Element Analysis (FEA) was developed using the Siemens NX 9 software to compute the temperature distribution of the satellite during its orbital mission.
Another software, the ESATAN-TMS by *ITP Engines UK*, was employed to perform the same calculation in order to verify and validate the results, due to the lack of experimental data.

Moreover, an analytical method was developed to estimate, at the conceptual design stage, the temperature distribution that the CubeSat would experience at a system level. Finally, the temperature distribution on the CubeSat is presented and some design recommendations are put forward.

Keywords

Nanosatellite, aerogel, mission, temperature distribution

1. Introduction

Humanity is characterized by the ever present curiosity and desire to learn. The *deep space* is the perfect mix of mystery and unknown for science. This in turn gave birth to the space science and engineering.

Sending an object or even a person into the space is dangerous and expensive. That is why everything intended for space applications must have a high reliability. As it is difficult and expense to test a satellite in space environment conditions, the computational design snd simulations take on an important and essential role.

This thesis focuses on the development of analytical and computational thermal models of a 3U CubeSat developed by ECOSat team at *University of Victoria*. ECOSat is a student-led group that competes in the Canadian Satellite Design Challenge (CSDC), which is a Canadian competition for university teams students to design and built a small CubeSat **?**.
This group at the University of Victoria Centre of Aerospace Research - CfAR, is developing a 3U CubeSat with a defined payload, mission plan and ground station. The *Homathko* mission shall be designed for an operating lifetime of two years.

In this thesis there are two parallel objectives, both about the prediction of temperature distribution on a CubeSat in a LEO orbit. Predicting these temperatures enables the design of thermal control systems [1] and at the end of an iterative process to validate (according to the requirements specified by CSDC, see **section ??**) as safe, reducing the possibility of failure in orbit that would jeopardize the mission.

First, two thermal models and analysis for Homathko are performed using the Nx software and ESATAN-TMS software. The results from both

[1] Every certificated device for space application has operational and survival range of temperatures

the commercial software are compared for validation and evaluation purpose.

Second, an analytical model was designed and implemented for fast and initial estimation of the CubeSat overall temperatures. This model is useful during the conceptual design stage. A detailed overview of the model is presented and the main variables that affects the behaviour are highlighted. this model was validated by comparison with two commercial software tools used in this work.

2. Space environment

Spacecraft thermal control is a process of energy management during its life cycles:

- **ground testing** - conduction, convection and vibrations stress the structure
- **transportation** - bumps and vibrations can damage the body
- **launch** - high velocity and drag would affect the payload (protection systems)
- **orbit transfer** - out of the atmosphere the radiation is the main source of heat
- **operational orbit**

Designing systems and materials that are operative in space is not trivial, the environment outside the Earth's atmosphere is hostile, therefore the need for a "shield", and particularly a thermal protection against it.

2.1. A "Unfriendly Surrounding"

The planet is characterized by a gravitational field and two protective shields: the atmosphere and the magnetosphere.

Out of this limit, an artificial satellite experiences the aggressive radiation from space. It is well known that the main heat exchange phenomena in this environment is the radiative one. Direct Sun radiation, Albedo and Earth IR radiation are not the only heat sources, other environmental effects and sources of heat are:

- **free molecular heating** mostly during launch ascent, individual molecules affect the vehicle
- **interplanetary trajectories** cosmic rays and more intense radiation can hit a satellite (it is not our case)
- **inner heat generation** every device employed for each subsystem, payload or for the TCS itself generate heat by *Joule effect*

All these forms of energy need to be handled.

2.2. Thermal Control System

Each component of the system works properly if is within their required temperature limits (operational and survival).

It is important to divide the thermal managment subsystem into *external thermal control loop* and *internal thermal control loop*: the external one exchanges energy with the outer environment, e.g. it rejects heat towards the deep space; the internal loop collects the heat generated by the equipment and transfers to the external loop.

Let us briefly list how the TCS works from a physical point of view. The heat exchange acts by conduction (**??**), convection (**??**) and radiation (**??**):

$$Q = kA\frac{T_1 - T_2}{\Delta x} \tag{1}$$

Q *energy transfer [W]*, T_1, T_2 *temperature at two points in the solid [K]*, k *thermal conductivity [W/(m K)]*, Δx *length of transfer path [m]*, A *cross section area* $[m^2]$

$$Q = hA(T_s - T_t) \tag{2}$$

Q *energy transfer [W]*, T_s *surface temperature of the solid [K]*, T_f *free stream temperature of the fluid [K]*, k *heat transfer coefficient [W/(*m^2 *K)]*, A *wet surface area* $[m^2]$

Every body with temperature higher than 0 K emits a fraction of radiative energy (according the parameter ϵ) to that emitted by a *black body*:

$$Q = \epsilon\sigma AT^4 \tag{3}$$

Q *radiated energy [W]*, T *absolute temperature [K]*, ϵ *emissivity*, σ *Stefan-Boltzmann constant [W/(*m^2K^4*)]*, A *interested surface area* $[m^2]$

The radiative interaction way prevails in the "empty" space without requiring any means of transport.

The radiations affect the body in different way, when they hit a surface we observe the absorption, reflection and transmission:

- absorptivity α, percentage of incoming radiation absorbed
- reflectivity ρ, percentage of incoming radiation reflected
- transmissivity τ, percentage of incoming radiation transmitted

Devices and coatings allow to manage all of these parameters are named in **section ??** and **??**.

2.2.1 TCS for Spacecraft

The TCS guarantees both an acceptable global energy balance and local thermal properties.

The TCS can be *passive* or *active*, we consider active a device that uses electric energy or moves fluid between two distinct points instead we consider passive all the other equipments.

The active thermal control is usually complex, expensive and relatively bulky, the main devices are:

- thermostatically controlled resistive electric heaters

- fluid loops
- louvers
- thermoelectric coolers

2.2.2 TCS for CubeSat

A spacecraft in its entire life cycle experiences different environments: ground, launch, orbit transfer and operational orbit. Here, only emphasizes the aspects of nanosatellite TCS, how to protect it from overheating and from too cold.
The passive thermal control is suitable for CubeSat, or small and nano satellite, because of its simplicity and it does not require energy; it includes **?**:

- insulation blanckets
- external surface coatings and finishes
- thermal fillers
- sun shields
- phase change materials
- radiating fins
- heat sinks

For a *large* or *medium* satellite both active and passive TCS equimpents are employed.

3. Homathko

A satellite is comprised a set of subsystems. The *payload* is the main "subsystem" and the whole spacecraft bus[2] is built up only to support and keep it healthy.

We need to know the exact configuration of the analyzed body (since it is still in working progress, assumptions are made on the uncompleted parts, see Figure **??**) to properly perform the simulation and being able to suggest changes if the requirements are not met.

The nanosatellite is equipped with two payloads, so it is able to perform two main functions: the primary mission objective is to provide a photometric source detectable by a ground base. The secondary mission objective is to supply an amateur radio repeater.

3.1. Constrains and requirements

Some specifics about the structural requirements are listed below, (a wider schedule is supplied in a CSDC paper **?**):

- The S/C configuration and dimensions shall be $113 \times 113 \times 340.5$ mm)
- The main S/C structure and corner rails shall be made of aluminum 7075, 6061, 5005, 5052[3]
- On the S/C protuberances are permitted, but they shall not exceed 6.5 mm perpendicularly

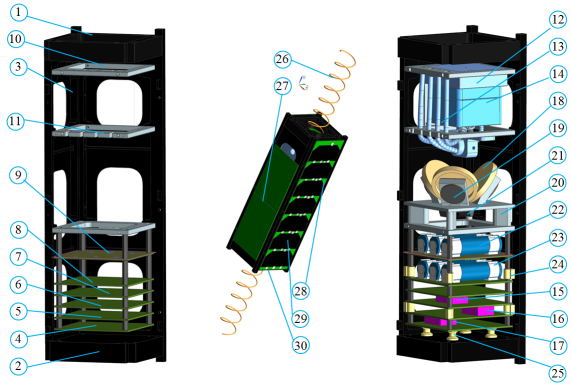

Figure 1: Internal and extarnal view of Homathko. Configuration of subsystems:Top plate (1), Bottom plate (2), panel side (3), AFE's PCB (4), OBC's PCB (5), Voltage regulator (6), PCB (7), Battery package's PCB (8), Battery package's PCB (9), Upper plate payload (10), Lower plate payload (11), Payload case (12), Wire (13), Payload camera (14), Device 1 (15), Device 2 (16), Device 3 (17), Flywheel (18), Motor (19), Support (20), Speed controller (21), Battery (22), Clip (23), Heat transfer (24), Heat transfer bottom (25), Helicoil antenna (26), PCB's antenna (27), Solar board (28), Solar cell (29) and Connector (30)

- The satellite shall have four rails, one per each corner, along the Z axis. The minimum dimensions of section normal to Z axis are 8.5×8.5 mm and the edge of the corner rails shall be rounded to a radius at least 1.0 mm. The material for this component shall be hard-anodised to prevent the cold-welding within the dispenser
- All parts shall remain attached to the S/C during launch, ejection and operation. Pyrotechnic devices shall not be used
- The spacecraft mass shall not exceed 4.0 kg. The center of mass shall be located within not more than 2.0 com from the S/C's geometric center in the X and Y axes and not more than 7.0 cm from the S/C's geometric center in the Z axis.

These requirements are very strict because the CubesSat has to fit the P-POD. The P-POD is an aluminium boxe, inserted inside the launcher, designed to hold tiny satellites (secondary payloads) as CubeSats.

4. Thermal Analytical model

Models are a key to predicting outcomes of physical phenomena.
The present analytical model is the very first step towards the realization of TCS and the feasibility of the mission from a thermal design perspective. Knowning the model of structure, it is demonstrated that a simple solution of the basic equation provides the thermal signature of a satellite.

4.1. Model development

The model is based on the *first law of thermodynamics* that expresses the conservation of energy.

This section explains how the "Basic Differential

[2] We could conceptually split the spacecraft into *payload* and *bus*

[3] Aluminum alloy: 7000 series (Al-Zn) high strength, 6000 series (Al-Mg-Si) easy to machine and weldable, 5000 series (Al-Mg) corrosion resistance

Equation" (hereafter BDE), on which the model is built, was obtained.

Starting with the first law of thermodynamics which for an *open system* takes the form of (**??**) and expresses the conservation energy principle, an open system is a control volume $\mathcal{V}$ permeable:

$$\frac{\partial E_{tot}}{\partial t} = \dot{\mathcal{E}} + \dot{\mathcal{Q}} + \dot{\mathcal{L}} \tag{4}$$

The total energy (**??**), sum of internal kinetic and potential energy, is useful to get the system temperature.

$$E_{tot} = \int_{\mathcal{V}} \rho e_{tot}\, d\mathcal{V} = \int_{\mathcal{V}} \rho(e + \frac{U^2}{2} + gz)\, d\mathcal{V} \tag{5}$$

The time variation of total energy of a system is due to:

- the flow energy $\dot{\mathcal{E}}$ of the same total energy that goes in or out from volume $\mathcal{V}$ per time unit

$$\dot{\mathcal{E}} = -\int_{\mathcal{S}} \rho e_{tot} \vec{U} \cdot \vec{n}\, d\mathcal{S} \tag{6}$$

- the absorbed heat per time unit $\dot{\mathcal{Q}}$

$$\dot{\mathcal{Q}} = \dot{\mathcal{Q}}_{outer} + \dot{\mathcal{Q}}_{inner} \tag{7}$$

- the absorbed work per time unit $\dot{\mathcal{L}}$

$$\dot{\mathcal{L}} = \dot{\mathcal{L}}_p + \dot{\mathcal{L}}_t + \dot{\mathcal{L}}_m \tag{8}$$

To obtain the BDE, it is necessary replacing each term in the first law and doing some assumptions:

$$\frac{\partial}{\partial t}(\int_{\mathcal{V}} \rho e_{tot}\, d\mathcal{V}) = \underbrace{-\int_{\mathcal{S}} \rho e_{tot} \vec{U} \cdot \vec{n}\, d\mathcal{S}}_{=0} + \dot{\mathcal{Q}}_{outer} + \\ + \dot{\mathcal{Q}}_{inner} + \underbrace{\dot{\mathcal{L}}_p + \dot{\mathcal{L}}_t}_{\simeq 0} + \underbrace{\dot{\mathcal{L}}_m}_{=0} \tag{9}$$

The volume control is the CubeSat (from 1U to 16U), so all physical quantities like density ρ, volume $\mathcal{V}$, surface $\mathcal{S}$ are related to it. The first right hand term is null because it is not supposed mass exchange for the nanosatellite. The two penultimate terms can be negligible, they would take into account the reversible power generated by pressure normal stress ($\dot{\mathcal{L}}_p$) and irreversible power coming from pressure shear stress ($\mathcal{L}_t$) respectively. The pressure stresses are negligible because the control volume take place in near-vacuum. The last term indicates the power from inner mechanics devices, so also this term is null since there is no moving body that develops work.
Deleting those terms:

$$\frac{\partial}{\partial t}(\int_{\mathcal{V}} \rho(e + \frac{U^2}{2} + gz)\, d\mathcal{V}) = \dot{\mathcal{Q}}_{outer} + \dot{\mathcal{Q}}_{inner} \tag{10}$$

The motion of a body i into the gravitational potential of a body j is characterized by the constant sum of own potential and kinetic energy. The kinetics and potential fractions of (**??**) can be delete for an orbiting object since they have to be derived from time.

$$\frac{\partial}{\partial t}(\int_{\mathcal{V}} \rho e\, d\mathcal{V}) = \dot{q}_{sun} + \dot{q}_{albedo} + \dot{q}_{IR} \\ - \dot{q}_{emitt} + \dot{\mathcal{Q}}_{inner} \tag{11}$$

The left-hand side of the (**??**) dependences only on temperature at fixed structure.

Then to get the shape of (**??**) the heat exchanged with the outside $\dot{\mathcal{Q}}_{outer}$ is splitted into several contributions:

- $\dot{q}_{sun}$ absorbed heat from direct solar radiation
- $\dot{q}_{albedo}$ absorbed heat from solar radiation reflected by Earth
- $\dot{q}_{IR}$ absorbed heat from Earth infrared
- $\dot{q}_{emitt}$ emitted heat from structure self

Now we see how calculate the right-hand side of (**??**):

$$\dot{q}_{sun} = A_{\perp} \cdot \alpha \cdot I_{sun} \tag{12a}$$

$$\dot{q}_{albedo} = A_n \cdot a \cdot \alpha \cdot I_{sun} \cdot K \cdot F_{e \to s} \tag{12b}$$

$$\dot{q}_{IR} = A_n \cdot \epsilon \cdot S_{IR} \cdot F_{IR} \tag{12c}$$

$$\dot{q}_{emitt} = \epsilon \cdot \sigma \cdot A_{tot} \cdot T^4 \tag{12d}$$

Only one term depends directly on the temperature, the emitted heat from the structure and we can rewrite it as $\dot{q}_{emitt} = DT^4$ and collect all the other terms in a single value q.

Given the structure, the volume $\mathcal{V}$ and density ρ are constant over time, after integrating they can exit the derivative. The internal energy per unit of mass for a solid assume the following form $e = cT$ in which c is the *specific heat* that we assume constant. Then we keep out of the derivative the specific heat and multiplying with the mass m ($c \cdot \rho \cdot \mathcal{V} = c \cdot m = C$) we show up the thermal capacity C taken in the following equation. The temperature $T(t)$ is the only variable over time, moreover it is exclusively dependents on the time, so the partial derivative can be replaced by total derivative.

$$C\frac{dT}{dt} = q - DT^4 \tag{13}$$

These results instantly detect the volume average temperature of the entire body along its orbit around the Earth.
In the table the factors that make the model sensitive to geometry, orbit and time are listed: Known the equations and the factors that determine them we are able to solve the BDE and finally get the temperatures.
I implemented this final step in two ways:

	Factor	Variability range	Unit
Ennvironment	Solar constant	1323÷1414	[Wm^{-2}]
	Albedo	23÷57	[%]
	Planerary IR	227÷241 ?	[Wm^{-2}]
	Eclipse time	-	[s]
	Altitude	160÷2000	[km]
Geometry	Number of face	1÷3	-
	Total surface	6÷50.1	[dm^2]
Structure	Absorptivity	0.1÷0.9	[-]
	Emissivity	0.2÷0.9	[-]
	Mass	-	[kg]
	Specific heat	-	[$Jkg^{-1}K^{-1}$]

Table 1: List of the parameters that affect the temperature

- **Matlab** I wrote an interactive code that ask you all factors regarding environment, geometry that you can set and after a few seconds a display would appear with plotted the average temperatures.
- **Graphs** I drew some graphs that make you able to approximate the same temperatures come out from the Matlab code, but without using a calculator, just your "hands".

4.1.0.1 Matlab code The *script* consists of just over hundred lines and is made up by some nested *cycles for*. It was used the solver *ode45* because it is the standard function to solve differential equations and performs well with most ODE problems that are non-stiff[4]. This solver also show a medium accuracy in the results it is enough for our studies. This function implements a *Runge-Kutta method* with a variable time step for efficient computation, the its inputs are basically the equation, the integration interval and the initial condition.

The code asks to the user one by one the inputs and for each number is suggested a physical doable range (e.g. see **table ??**).
The required data are general, no structural or material details. To be more clear let's explain quickly some inputs: the *"Orbit period"* times *"Number of orbit"* is set as integration interval; the *"Starting point"* make you able to start the simulation from sunlight or umbra phase.

To solve this problem it only takes a few seconds and at the end a temperature profile appears with different color during sunlight (red line) and eclipse time (blue line), as showed in Fig. (**??**) and the temporal gradient are computed and placed in the table **??**:

4.1.0.2 Graphic method The graphic method is based on the same equations, since the analytical model is unique. It involves the use of some "extra equations" and of course some graphs.
Remember that the purpose of the thermal analysis

[4]A stiff equation is numerically unstable, it means the equation has some terms that can lead to rapid variation in the solution if the method and time step is not adequate

Table 2: Thermal temporal gradient for each orbit during sunlight phase and umbra

Sunlight					
	1st orbit	2nd orbit	3rd orbit	4th orbit	5th orbit
Grad [$°C\,s^{-1}$]	0.0025	3.872e-04	4.779e-04	8.763e-04	0.0011
Umbra					
	1st orbit	2nd orbit	3rd orbit	4th orbit	5th orbit
Grad [$°C\,s^{-1}$]	0.0073	0.0060	0.0055	0.0052	0.0051

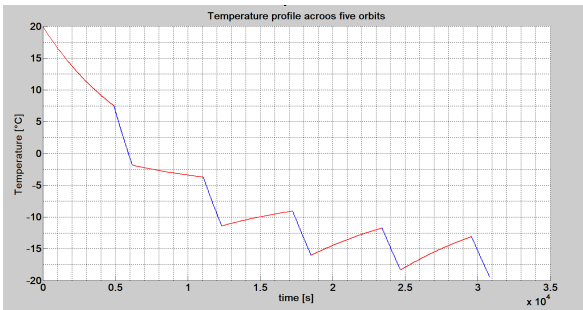

Figure 2: Ex: temperature profile of a simulation from Matlab code

performed by this method is only to design a rough TCS, so we have to focus on *hot* and *cold* case.

The data required are always equal as those mentioned above (environmental, geometric, orbital and so on) and through the equations (**??**), the BDE (**??**) is known. So far nothing is new, but now we have to perform by hand these calculations.
We need to rearrange the equation (**??**) to make it resolvable by our charts, dividing the BDE by the thermal capacity C:

$$\frac{dT}{dt} = q^\star - D^\star T^4 \tag{14}$$

where $q^\star = q/C$ and $D^\star = D/C$. These two variables with the initial temperature (hereafter T_0) and the eclipse time are the "macro variables" we need.

Finally straight to the graphs, four graphs have been drawn:

- The first chart give us the opportunity to compute the temperature at the end of a sunlight period, from now on we will call this temperature T_{up};
- The other three graphs have to be combined together, they provide the known term and two coefficients of an extra equation. This quadratic equation pulls out T_{down}, the temperature after eclipse time.

At the end we only got a few points that identify the extreme temperatures, T_{up} and T_{down}, to which the body would be subjected. The final step might be to place these points on a temperature-time graph and join them by a line. The result should look like *fig.* (**??**) obtained from Matlab.

The first graph (*fig.* **??**) plots T_{up} versus $q_s^\star$ (the subscript "s" indicates that $q^\star$ is computed for the

sunlight period, the variable $q^\star$ and $D^\star$ have to be adapted to each section).
For clarify, only a few curves are plotted, but the real graph (*see figs.* is more detailed.

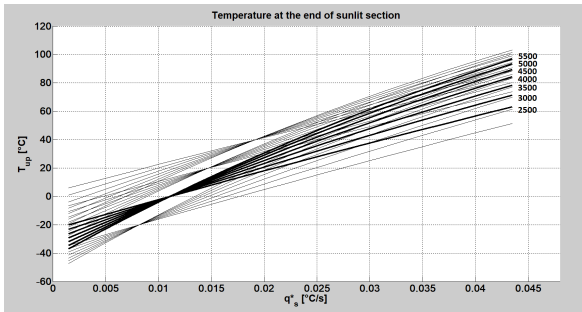

Figure 3: Trend of the temperatures at the end of sunlight section

As already hinted, $q_s^\star$ is equal to the sum of solar flux, earth irradiation, albedo and the heat loss coming from powered devices during sunlight divided by the average thermal capacity of the whole satellite. Known this values, expressed in degree Celsius over seconds, we select it on the abscissa axis of this first graph (*fig.* **??**), individuate one of the oblique line, regarding the interested case, and then we can read the corresponding temperature, (T_{up}) in degree Celsius, on the ordinates axis.
To find the right line that works for our case, we must note the presence of different bundles of straight lines (four bundles in *fig.* **??**) converging in a single point and seven lines for each bundles: the bundles identify the "initial condition", the intersection point ordinate suggests the value of the temperature at the beginning of the simulation; the seven lines express the duration of sunlight period in seconds. For example, if the our boundary condition at the beginning is 0°C for a sunlight period of 4000 s, we will go to select the central bundle (intersection point on 0°C) and choose the seventh line from the bottom.
Obviously the number of bundles has to be increased in order to make the method as general as possible, and also the number of line per bundles could be increased.

From the following three graphs we can draw the coefficients that make us able to compute T_{down}.

Those extracted coefficients don't have a physical meaning, but they represent the coefficients of the following quadratic equation:

$$T_{down1} = a_2 \cdot q_u^{\star 2} + a_1 \cdot q_u^\star + k \tag{15}$$

Handling this equation we compute the T_{down} temperature in °C, it stands for the temperature at the end of eclipse time.

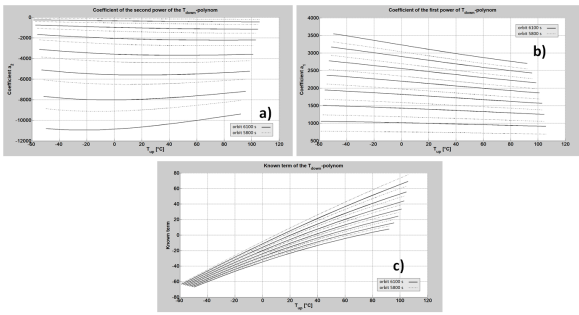

Figure 4: **a)** Coefficient trend in (s^2 °C^{-1}) of the second power term of the equation **??**, **b)** Coefficient trend in (s) of the first power term of the equation **??**, **c)** Known term trend in (°C) of the equation **??** (it appears as "k")

5. Thermal Analysis - FEA

Obviously a software analysis is just a hypothesis, that can be more or less close to the reality according to the assumptions and it does not have the same reliability as an experimental test.

5.1. Finite element analysis performed by NX 9

To perform the simulation two software are exploited, this section section is dedicated to the use of NX 9 Siemens.
Let's start from our model[5] already finished. It is necessary to carry out a previous analysis of how we should set the parameters in order to cover most cases possible the satellite can meet in its operative life.
Two main simulations have been run to estimate the maximum and minimum temperatures the satellite could reach. The parameters for the so-called "Hot and Cold" case are choosen, they are summed in table **??** and are relative to the orbit, inner devices and space environment. A quick study was done to chose the orbits running several simulations with a simple body (to be fast) and changing the orbital elements could affect the temperatures: firstly the altitude that works on the albedo value and IR Earth flux, the inclination (we need a sunsynchronos orbit) and the RAAN (right ascension of the ascending node) that affected the sunlight and umbra time, consequently the temperatures. During this several simulations also the solar flux it has been changed even if it is an environment element.

While the choices of the conditions on the inner devices came from project decisions, e.g. they are related to whether or not a device is powered on or off during an orbit or even about a possible its failure.

[5]Homathko CAD model was partially realised by a team member and I added subsystem, devices and details

Table 3: Parameters set for hot and cold case

	Parameter	Hot case	Cold case
Orbit	kind	synchronous	synchronous
	Altitude	300 km	800 km
	RAAN	105°	0°
Environment	Position	perihelion	aphelion
Inner devices	OBC	normal mode	low mode
	Payload	on	off
	Downlink	on	off
	Uplink	processing	waiting

5.1.1 Starting base analysis - FEA

The CAD model was updated with the following subsystems: the antenna group, the ADCS, the EPS (batteries and solar panels).
The first simulation was run with the "primitive" structure, it means without any improvements, but with every subsystems added. After these first results came out several simulations have been done and some improvements are suggested and analysed until the temperatures are within the safety limits.

Siemens NX 9 was used with the solver NX SPACE SYSTEMS THERMAL. The mesh is created, the orbit, thermal loads and contacts between components are defined. It's necessary to reduce the holes, blend edges, faces in order to get less nodes and a faster simulation. Before you idealize the structure you need to be sure that the changes made to the original body do not affect the results.

5.1.1.1 Mesh optimization To save time an appropriate mesh should be used, selecting the type and the size.
A study on a *panel side* of the aluminium structure is performed in order to idealise the body as much as possible and get the biggest mesh size acceptable. The following analysis show how and why those parameters are setted. Let's consider the panel side of our structure, it has been subjected to a similar condition of the whole satellite. Firstly a simulation with tight mesh (6 mm element size) and real structure was done:

- the results sampling was setted at the minimum, 10 samples per orbit
- the integration time is variable according to the convergence criterion
- less than 0.1 °C is the acceptable different between two successive orbits

The last point marks the end of transition phase and then defines automatically the integration time, this means that the temperatures will be, theoretically, forever the same.
In the following analysis only the mesh size was changed; the other parameters are satisfactory, their changes would lead to a useless increase in the solution time. The panel side was idealized (chamfer and small holes deleted) and another simulation has been run increasing the mesh size to 12 mm then a third one has been done with the element size equal to 18 mm.

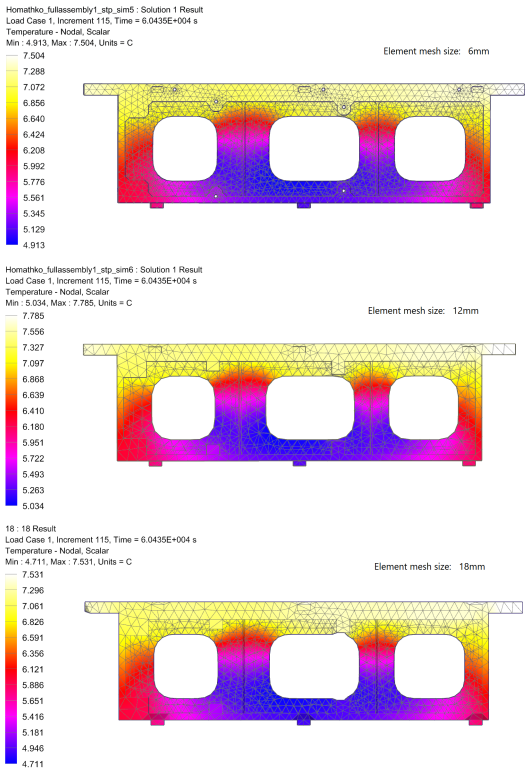

Figure 5: Analysis of a alluminium panel side with idealizations and different meshes

Table 4: Computational details of three simulations

Mesh size	Thermal nodes	Elements	Surf. elements	Sol. time
6 mm	3597	10395	1225	24h 35min 56sec
12 mm	1219	3183	519	2h 52min 43sec
18 mm	1525	3945	872	1h 17min 46sec

The temperatures between them are very close, the differences are less than 0.4 °C. The 18 mm mesh seems better to approximate the results of "real case" (6 mm mesh). Paradoxically to fit the panel side with 18 mm size it is necessary more elements than 12 mm size, that's why a smaller error is gotten.

5.1.2 Structural idealization and FEM model

The following description is about the final model with all changes on and not about the primitive structural.

The final model is very complex and full of small details. Every chamfer, blend edges and hole (under 2 mm diameter) are deleted.
According to the *fig.* **??** the idealized components are: $\langle 1 \rangle, \langle 10 \rangle, \langle 3 \rangle, \langle 11 \rangle, \langle 2 \rangle, \langle 12 \rangle, \langle 14 \rangle, \langle 18 \rangle$ and $\langle 19 \rangle$; almost everything about the aluminium structure.

Now the structure is ready to be meshed. This is the most crucial part about the reliability of the results. It depends on some parameters:

- how the software is built and which solver is used
- how the model reflects the reality with all of idealizations
- simulation settings

Depending on the aim, type of simulation, level of reliability required and available time you should define the above points.
The main constrain was the time: a simulation of complete model takes from 3 to 5 days because of default solver settings, the employed calculator and the complexity of the model. It is necessary to run a lot of simulations to analysed the structure under different cases and verify the improvements, then basic simulation settings were chosen.

The solver SST uses an implicit integration method to be be a bit more accurate but computationally acceptable. The time step is constant during one simulation, 100 time steps per orbit. If an orbit takes 5400 s (as a synchronous orbit at 300 km) the time step will be 54 s. Two simulations were run with a simple body once with a constant time step and then with a variable one, the results were almost the same, but the second case takes a bit longer.

The CAD model made at the beginning reflects the reality, every detail of the structure and the inner subsystem is sketched. The satellite was idealized and from an analysis performed on the panel side came out that it doesn't affect too much the results.

After this preliminary analysis the solver, analysis type and solution control are setted, we proceed to mesh every component. Some of them are arranged in the following tables.

The 2D elements (triangular element with 3 nodes) are employed only for two components the clips $\langle 23 \rangle$ and the solar cells $\langle 29 \rangle$, the mesh parameters are summed in the **table ??**.

Table 5: Features of 2D elements

Component	Element size	Thickness	Material
Solar cell $\langle 29 \rangle$ $\times$**21**	20 mm	0.38 mm	Germanium
Clip $\langle 23 \rangle$ $\times$**12**	10 mm	0.20 mm	Steel

The rest of the structure is meshed with 3D elements (tetragonal element with 4 nodes) with different size and the appropriate material.

5.1.3 Boundary conditions

The solver SST makes us able to simulate every heat load a body could meet into the "black space". Through some commands it allows us to set the orbital elements, environment and the inner heat losses to simulate the cases we need.

5.1.3.1 Orbital Heating The command *Orbital Heating Simulation Object* sets the orbit and then it computes automatically the environment heat load such as the solar flux, the albedo, the IR Earth flux and so on.
Two orbits were defined at 300 km and 800 km of altitude because when the simulation was designed the exactly orbit was unknown. The solar flux was over defined to simulated the perihelion and aphelion extreme cases (respectively 1414 Wm^{-2} and 1323 Wm^{-2}). From a previous analysis also the RAANs were chosen at $105\circ$ and $0\circ$ respectively for hot and cold case.

5.1.3.2 Radiation All outer surfaces of Homathko model have to be sensitive to the radiation in order to simulate exposure to the sun. This command (*Radiaton Simulation Object*) has to be coupled with the setting of thermal optical properties of each material, in the table **??** these properties are defined.

Table 6: Thermal optical properties of outer materials

Ex. element	Material	Emissivity (ϵ)	Absorptivity (α)
Exterior structure	Aluminium	0.84	0.14
Pcb	Fr-4	0.94	0.123
Solar cells	GaAs	0.89	091
Helicoil antennas	Steel	0.92	0.15

5.1.4 Suggested changes and results

The first simulation pointed out several spread problems: the helical antennas reached temperatures over thousand degree Celsius, the inner devices went beyond 130 °C and also the aluminium suffered the overheating.
We itemize the modifications applied to the model:

- the antennas were coated with white paint to reduce the ratio α/ϵ
- the whole exterior structure of aluminium was black anodized
- heat transfers were added to extract the heat from the inside
- the supports of plate wheel were changed to improve the heat exchange

The temperatures of the main components and their operating temperature range are presented in the following table (**table ??**).

The dark cells point out some over-cooling problems, that should be solved by new arrangements of some devices and modifying a little bit the thermo-oprtical properties.

Table 7: Resluts of FEM analysis (NX 9). The dark cells point out the state out of the safety temperature range

	Operating range		Cold case		Hot case	
Component	T_{min} °C	T_{max} °C	T_{min} °C	T_{max} °C	T_{min} °C	T_{max} °C
Payload box	-100	100	-39.1	-27.42	11.49	14.60
Payload camera	-20	60	-38.89	-27.4	11.56	14.66
Flywheel	-30	85	-37.94	-28.64	12.09	14.15
Motor	-25	80	-39.6	-26.53	11.73	14.79
Speed control	-25	60	-40.22	-27.35	11.63	14.72
AFE	-40	85	-39.32	-26.4	12.77	15.83
OBC	-40	85	-40.25	-27.4	11.57	14.67
Volt regulator	-40	85	-39.83	-26.95	12.01	15.1
PCB 1 (AFE)	-40	110	-39.34	-27.8	11.7	14.75
PCB 2 (OBC)	-40	110	-40.15	-27.22	11.57	14.67
PCB 3 (Volt reg.)	-40	110	-40.14	27.20	11.83	14.92
PCB 4 (Charge contr.)	-40	110	-34.36	-30.97	13.45	15.35
PCB 5 (batt. pack 1)	-40	110	-17.81	-15.93	29.87	30.28
PCB 6 (batt. pack 2)	-40	110	-25.57	-19.92	25.1	26.17
Plate wheel	-100	100	-40.19	-27.36	11.620	14.71
Battery	-20/0	45/60	-15.68	-13.73	32.2	32.78
Antenna board	-40	110	-39.48	-27.10	11.47	14.6
Antenna helicoil	-40	70	-39.95	-30.86	4.43	7.02
Solar panel X+	-40	85	-39.32	-27.46	11.54	14.65
Solar panel X-	-40	85	-39.35	-27.5	11.44	14.54
Solar panel Y+	-40	85	-39.29	-27.44	11.49	14.6
Structure	-100	100	-15.68	-13.73	32.2	32.75

The payload is the main subsystem and its overheating or excessive cooling could compromise the entire mission, an inner view of payload box and camera with their distributed temperatures are showed (*fig.***??**):

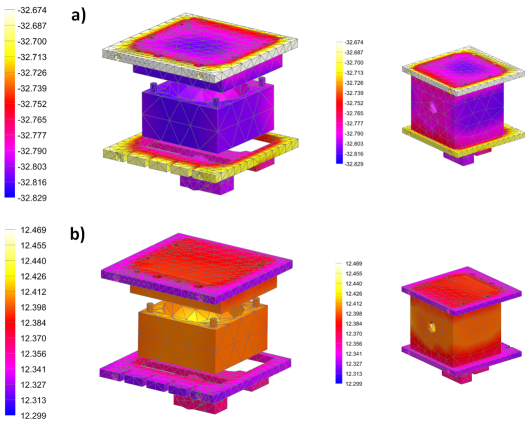

Figure 6: Temperature distribution of payload

The thermal gradient are not critical thanks to the high thermal conductivity of aluminium structure and the small size. The crucial component are the antenna and the panel sides: the thermal spatial gradient of helical antenna is 0.64 $°\mathrm{C\,cm^{-1}}$ instead the thermal temporal gradient is 0.0015 $°\mathrm{C\,min^{-1}}$; the thermal temporal gradient of a panel side is 0.087 $°\mathrm{C\,min^{-1}}$.

5.2. Supplementary software analysis - ESATAN

The results of ESATAN-TMS are summed in the **table ??** and are comparable with NX's ones. The *lumped method* was used, it means that each modelled component of the satellite was considered as a point.

5.3. Thermal analysis by analytical model

This section is a bit out of the question, because it is still a thermal study, but not performed by FEM analysis. It was used the analytical model previous named. Only the results and some additional explanation will be presented, the entire method to get the final temperatures by analytical model is already showed in **section ??**.

In the **table ??** the values to set the Matlab model are estimated, bot Hot and Cold cases are shown. The results (see *fig.* **??**) got the convergence after four orbits. The temperatures shows a trend close to the results came out from NX analysis, but comparing them in details, calculating the relative errors between them, it doesn't make a lot of sense because they have different meaning: the temperatures of analytical model identify the behaviour of a center of mass of Homathko during four orbits and for three different configurations with N from 1 to 3.

Table 8: Results of analysis performed by ESATAN

	Average temperature °C	
Component	**Cold case**	**Hot case**
Up plate payload	11.1	17.45
Wheel plate	11.3	17.7
PCB 1 (AFE)	16.9	21.5
PCB 2 (OBC)	24.55	25.8
PBC 3 (Volt reg.)	23.85	25.05
PBC 4	14.6	18.85
PBC 5 (pack 1)	10.45	18.2
PBC 6 (pack 2)	21	21.25
Battery	21.6	21.45
Intermediate plate	10.8	16.95
Antenna board	5.4	20.25
solar panel (Z-)	11.45	17.15
solar panel (Y-)	7.8	24
solar cell (Y+)	14.95	13.9
Structure	9.05	15.15

Table 9: Input for thermal analysis by Matlab code

Input	Hot case	Cold case
Solar constant	1414 [$\mathrm{Wm^{-2}}$]	1323 [$\mathrm{Wm^{-2}}$]
Altitude	300 [km]	800 [km]
Planerary IR	237 [$\mathrm{Wm^{-2}}$]	237 [$\mathrm{Wm^{-2}}$]
Albedo	0.306	0.306
Number of face	1÷3	1÷3
Total surface	14 [$\mathrm{dm^2}$]	14 [$\mathrm{dm^2}$]
Volume	1.46 [$\mathrm{dm^3}$]	1.46 [$\mathrm{dm^3}$]
Initial cond. (temp.)	293 [K]	293 [K]
Average [6] specific heat	876.2 [$°\mathrm{C^{-1}\,s^{-1}}$]	876.2 [$°\mathrm{C^{-1}\,s^{-1}}$]
Average conductivity	143.7 [$\mathrm{W\,m^{-1}\,K^{-1}}$]	143.7 [$\mathrm{W\,m^{-1}\,K^{-1}}$]
Average density	3331 [$\mathrm{kg/m^3}$]	7829 [$\mathrm{kg/m^3}$]
Emissivity	0.88	0.74
Absorptivity	0.39	0.34
Heat loss from devices	7.8 [W]	3.6 [W]
Number of orbits	4	4
Starting point	sunlight	sunlight
Orbit period	5423 [s]	6043 [s]
Duration of umbra	0 [s]	2103 [s]

6. Conclusion

The distributed temperatures that came out from this work assert the operation within safety temperature limits. Of course, the reliability of these

[6] If the object has to be simulated consists of more than one material

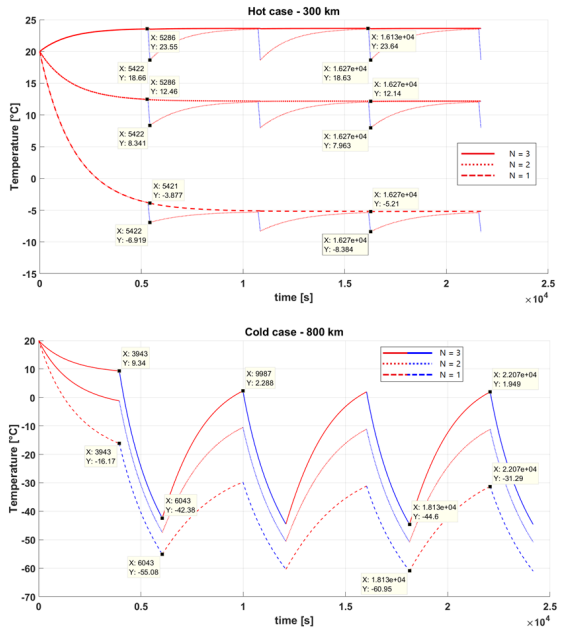

Figure 7: Global temperature of Homathko for "Hot" and "Cold" case

results comes from a calculation, as we have been already said it is impossible for us to compare with experimental data.
Nothing is able to replace a practical test, but efforts can be made to make acceptable the calculations as doing the same surveys by different software and methods.

6.1. Outcome

The conditions, under which it was subjected, cover the entire working life of the satellite, let's see them arranged in the Table **??**:

Table 10: Analyzed operational conditions

Condition	Comment
Sunsynchronous orbit	The kind orbit that will be used during the whole mission
BOL	The surface and devices conditions are at Begin Of Life, without any damage
Altitude 300-800 km	The nanoSat will not be out of this band around the earth, it is the main streak for a CubeSat
Operating equipment	Maximum and minimum powered devices according different mode of working. Hypothesized breakdown or total payload shutdown.
Orbital situation	Boundary conditions of aphelion and perihelion; maximum and minimum eclipse time

One of the crucial points might be the estimation of the temperatures at the end of the mission. During the mission, for example a device could break and we took in to account this (Operating equipment, in the Table **??**), but the surface finishing and them thermal optical properties alter. The temperatures tend to rise (the ratio α/ϵ increases), observing the values of the external and internal elements, we can see a low trend of temperatures, none goes over 30°C.

So, there should be no overheating problems untill the end of the mission.

Consideration about methods of analysis

Siemens NX 9 even allows the creation of complicated sketches like every CAD software, it's useful for small subsystem and elements.

Using ESATAN-TMS to draw an object is quite hard, it's not intuitive like NX 9 to relise the sketch, because every point of the model has passed from command window. It is therefore extremely complex make the satellite with small details, it is really fast and helpful for large satellite with size over one meter.

The analytical model does not include the use of a model, we cut off the time to realise it and the simulation time. The only necessary time is a couple of hours to solve some equations as in a common exercise.

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References

2017 - The Canadian Satellite Design Challenge - Website by Pamela Rubio
url = http://www.csdcms.ca

Spacecraft thermal control - from Wikipedia, the free encyclopedia url = https://en.wikipedia.org/wiki/Spacecraftthermalcontrol

title: The Canadian Satellite Design Challenge - General Rules Requirements Issue 3a - October 2014

title: The Canadian Satellite Design Challenge - Design, Interface, and Environmental Testing Requirements Issue 3a - October 2014