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Phase C

Swisscube
Thermal Analysis

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-

RECORD OF REVISIONS

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1/0	4/04/2008	CREATION	Florian MOLLIET
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INTRODUCTION

A model of thermal analysis has been developed previously to study thermal conditions on the satellite in his space environment. It is critical to design the satellite to make all components stay in their allowable temperature range. The goal of this project is to predict thermal conditions according to orbital parameters for different scenarios and to adapt some parts of the design: fine tuning of the satellite in order to keep the electronics, the battery and the structure in their proper temperature range. The electronics are also sensitive to high heating/cooling rates which occur during sun-eclipse transitions. Temperature evolution on the satellite components is simulated with a finite element model in this work.

Simulations are run into COMSOL; a multi physics solver; the last model used a Matlab code to calculate heat flux boundary conditions. The first goal of this project is to make the previous model interact with a space technology software: STK to get orbital parameters and compare new boundaries conditions given with the one used previously.

The second goal is to simulate the thermal behaviour of the satellite according to new launch information: Vega or Indian. Simulation scenarios are defined using orbital parameters in order to study: structural parts of the satellite, the electronics and the battery. A thermal tuning is proposed in order to stay in the allowable temperature range and minimize thermal heating/cooling rates.

1 ASSUMPTIONS

The list of parameters used for simulation and calculations is given below.

Space parameters:

- Solar constant G_{sun} = 1412 W/m^2

- Earth IR constant G_{earth} = 217 W/m^2

- Albedo ratio = 0.3

- Deep space temperature T_{space} = 2.7 K

- Earth radius = 6378 km

Thermal parameters:

- η_{SC} = solar cells efficiency = 0.2

  (20% of what is received is converted into electricity, 80% contribute to thermal heating)

- Solar cells absorptivity = α_{SC} = 0.9 (visible)

- Frame absorptivity = α_{frame} = 0.24 (visible)

- Solar cells emissivity = ε_{SC} = 0.8 (IR)

- Frame emissivity = ε_{frame} = 0.8 (IR)

  Standard external panels absorptivity

  - case 1 – black paint - α_{bp} = 0.9

  - case 2 – PCB - α_{PCB} = 0.75

- Standard external panels emissivity

  - case 1 – black paint - ε_{bp} = 0.9

  - case 2 – PCB - ε_{PCB} = 0.89

The absorptivity in the IR is assumed equal to the emissivity in the infrared spectrum.

Material parameters have been given by COMSOL data base or are the same than in [R1]

Conductivity for spacers:

K_{Al} = 160 W/(m·K), previous report value according to thermal tests [R1].

K_{POM} = 0.32 W/(m·K)

2 THERMAL MODEL

2.1 COMSOL model components

The model used is geometrically the same than the one developed during the previous work. It is described in the Fig 1 below. It is composed of an aluminum frame, six FR4 boards for panels, and twelve solar cells. Two PCBs used to emulate the temperature evolution in the integrated circuits or the motherboard depending on spacer properties sets, a battery box containing batteries. PCBs are called PCB1: the nearest one from the frame and PCB2 the most inside. All components properties have been dimensioned to be thermally equivalent to the real components in the previous work [R1]. All internal components are separated by small cylinders called spacers. Those are used in this work to study different electronic cards and set in order to insulate satellite components. The current spacer material is POM (polyoxydemethylene) for the battery box and in aluminum everywhere else (sees assumption for conductivity values).
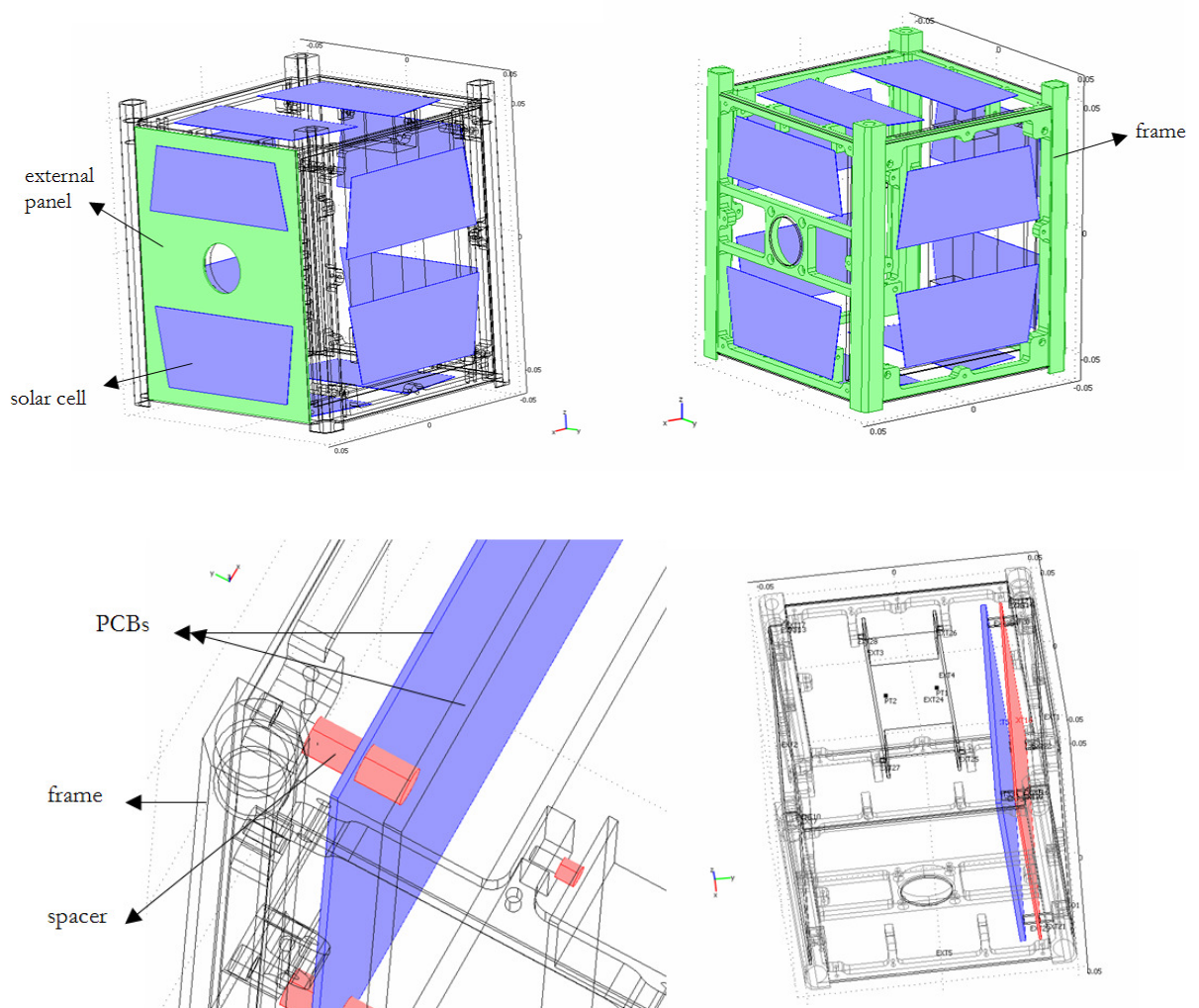

Fig 1: FEM model description [R1]

2.2 STK data: motivations

In order to adapt the previous model to new orbital conditions, it is useful to make it interact with STK: professional space technology software. The previous model was using a Maltab code to give heat flux boundary conditions for the FEM simulations. Some mathematical simplifications have been introduced; therefore it is also interesting to compare previous results and STK professional data.

2.3 Function architecture and data

The first step is to import STK data and transform it in low level data used in further calculations. Orbital and time parameters are set in STK in a scenario, then STK and Matlab are linked to import the punctual data (i.e. sun vector, sunlight times….) and transform it in a intermediary form to minimize further calculation. This data is then saved as a Matlab workspace, for a set of time steps: 10s. More details are given in Appendix I describing those steps and functions.

The second step is to transform this punctual data for FEM boundary condition, for all times. As mentioned previously, the data is given for a set of time values. Linear interpolations are used to get parameters for all required times of the data range (asked during the simulation by COMSOL). These data is transformed to provide COMSOL the received heat flux for all external surfaces: panel, frame and solar cells. These calculations take into account, the sun emission, the earth emission, the albedo (reflection of the sun radiation by the earth). The internal radiation is not taken into account as it wasn't in the previous simulations [R1].

Additional functions characterize heat generation of devices: the battery active heating device, the energy dissipated on the motherboard, the electronic auto heating by joule effect. A scheme of those steps is given in Appendix II and III.

3 VALIDATION OF THE NEW BOUNDARY CONDITIONS: STK DATA

In order to validate the new developed model, a comparison of the sun vector direction in the orbital reference frame has been done. The incoming direct sun emission is computed for both boundary conditions and all faces.

To finally validate the STK model, the total incoming energy is compared.

3.1 Parameters used for the two model boundary comparison

Orbit: Sun synchronous orbit, RAAN =30°, e = 0, h = 1000 km.

Spin of the satellite: $\dot{\alpha} = 0$ [deg/s], $\dot{\beta} = 0$ [deg/s], $\dot{\gamma} = 0$ [deg/s]:

Rotation rates in the orbital reference frame.

Starting date: 01/12/2008 at 13:12:00

Time: Number of seconds starting from 01/12/2008 at 00:00:00

Orbital reference frame: x = earth-to satellite axe, y = velocity vector, z = x cross y.

Spaces constants: see assumptions.

The goal is this part of the study is to evaluate the incoming heat flux coming directly from the sun, therefore no material parameters are needed at this point.

3.2 Comparison of the sun vector in the orbital reference frame

3.2.1 Results

The most relevant parameter in the comparison of the incoming heat flux is in our case the direction of the sun in the satellite reference frame (sun vector). The normalized sun vector obtained with STK data (new model) is computed and compared with the previous model in the Fig2 and Fig3 below.
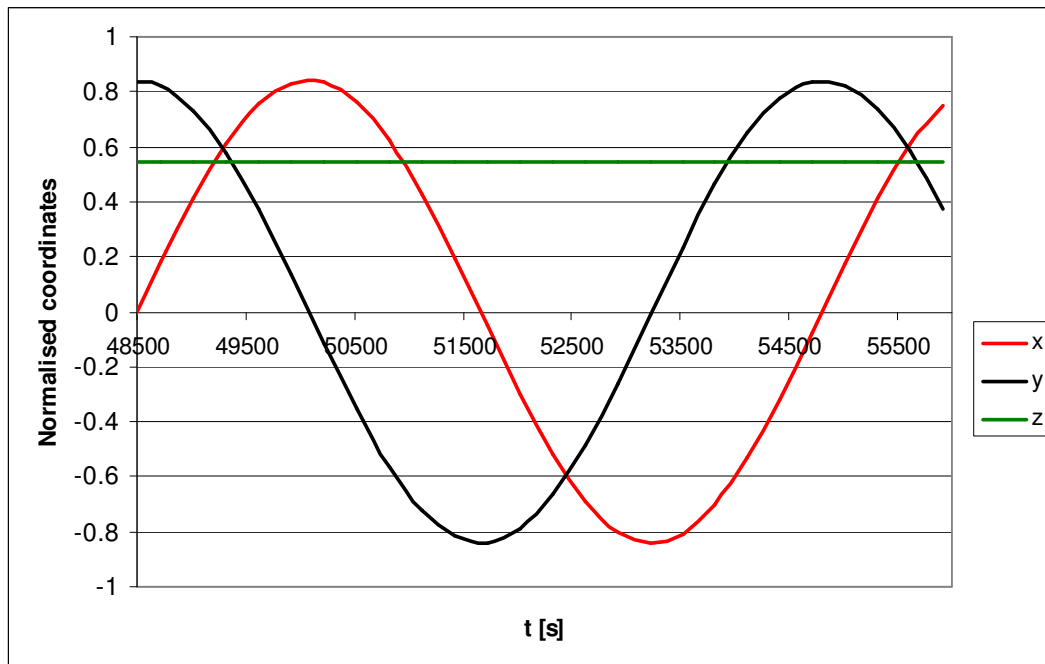
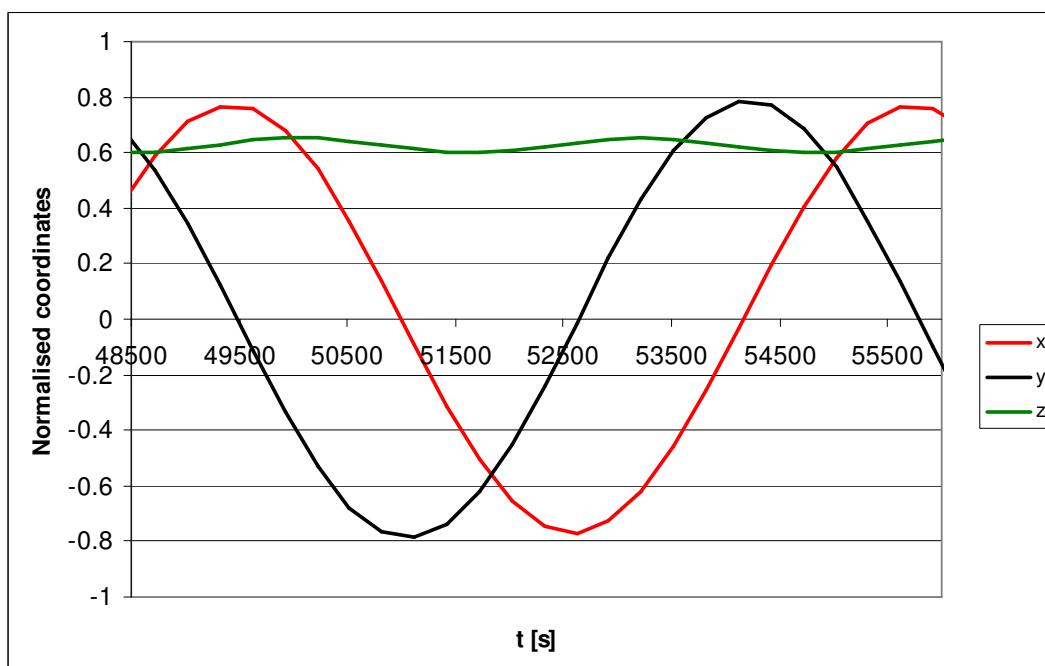*Fig 2: Previous model results: Normalised sun vector components in the orbital reference frame*

Fig 3: New model results (STK linked): Normalised sun vector components in the orbital reference frame

3.2.2   Sun vectors comparison

The period is the same for both conditions (one cycle is about 6600 s), a time shift of about 700 s occurs. The x, y, z amplitude have a similar behavior, values are slightly different but are coherent, z coordinate oscillates in the new model: in Matlab functions the sun orbit is approximate as circular (elliptic in STK), and the velocity on the orbit is considered as constant, STK takes into account perturbations which acts on the motion of the satellite (i.e as moon effects).

Differences can be explained by the higher accuracy of STK compared to what was programmed in the previous model code.

In order to see how those differences affect previous work results [R1], the total incoming solar energy will be compared in the next paragraph.

3.3   Direct incoming sun heat flux comparison

Knowing the sun vector in the satellite reference frame, it is possible to calculate the sun incoming heat power, function of time, for all the satellite faces. It is done using the formula below:

$$P_j(t) = \sum_{j=1}^{j=6} G_{Sun} \cdot S_j \cdot Sun_j(t) \cdot \cos(\vartheta_j) \; \textbf{[W]}$$

j   :   Represents faces of the satellite in the satellite reference frame, j equal (1,2,3,4,5,6) respectively for the (+x, - x , +y, -y, +z, -z).

$Sun_j(t)$:   Value of the rate of illumination from the sun for each faces: full light, penumbra, umbra, and depending on the face exposure [-].

$\cos(\vartheta_j)$:   Angle between the sun vector and faces of the satellite [-].

S_j :   Surface of the face j = 10^{-2} m^2.

G_{Sun} :   Sun radiation constant (see assumptions) W/m^2.

The incoming power for each face of the satellite is given below for previous model calculations and the STK data. Satellite spinning is set to zero (see 3.1). The z coordinate behaviour can be found in those results, eclipses times are cutting the heat power.

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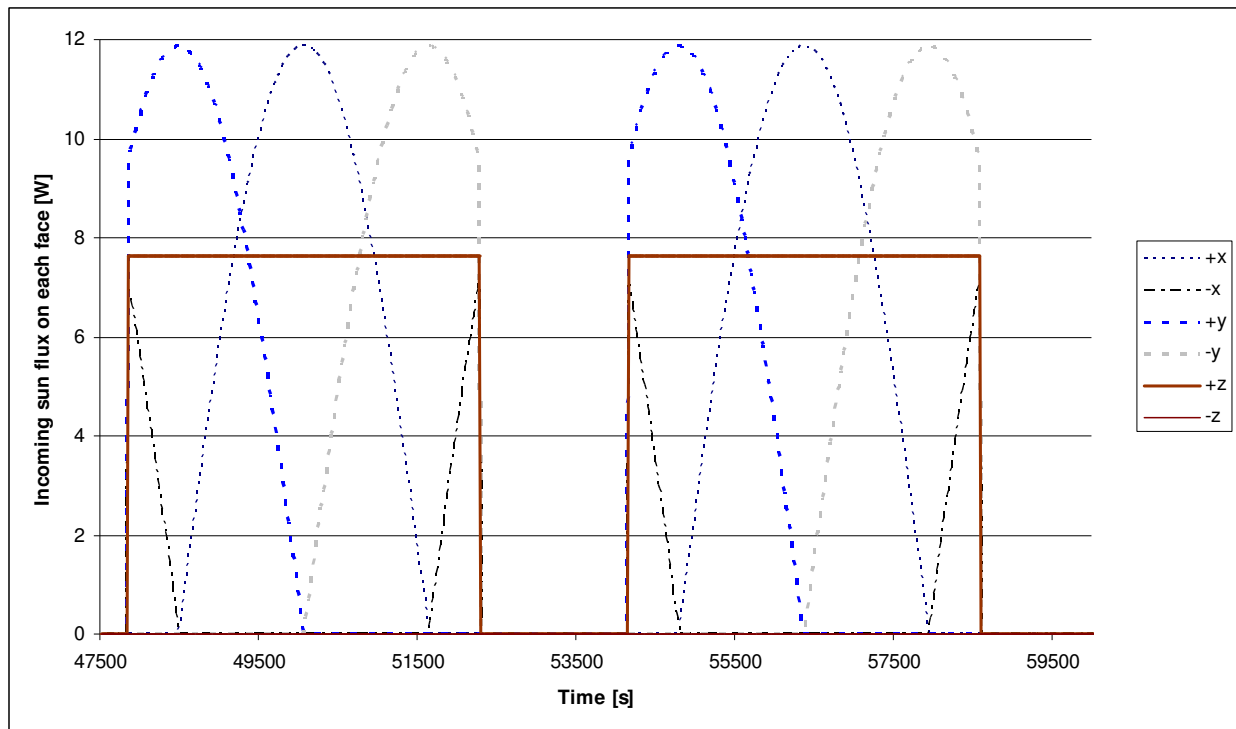
Previous model calculations:

Fig 4: Matlab calculations: incoming sun power for each face-1000km altitude

New STK model data:
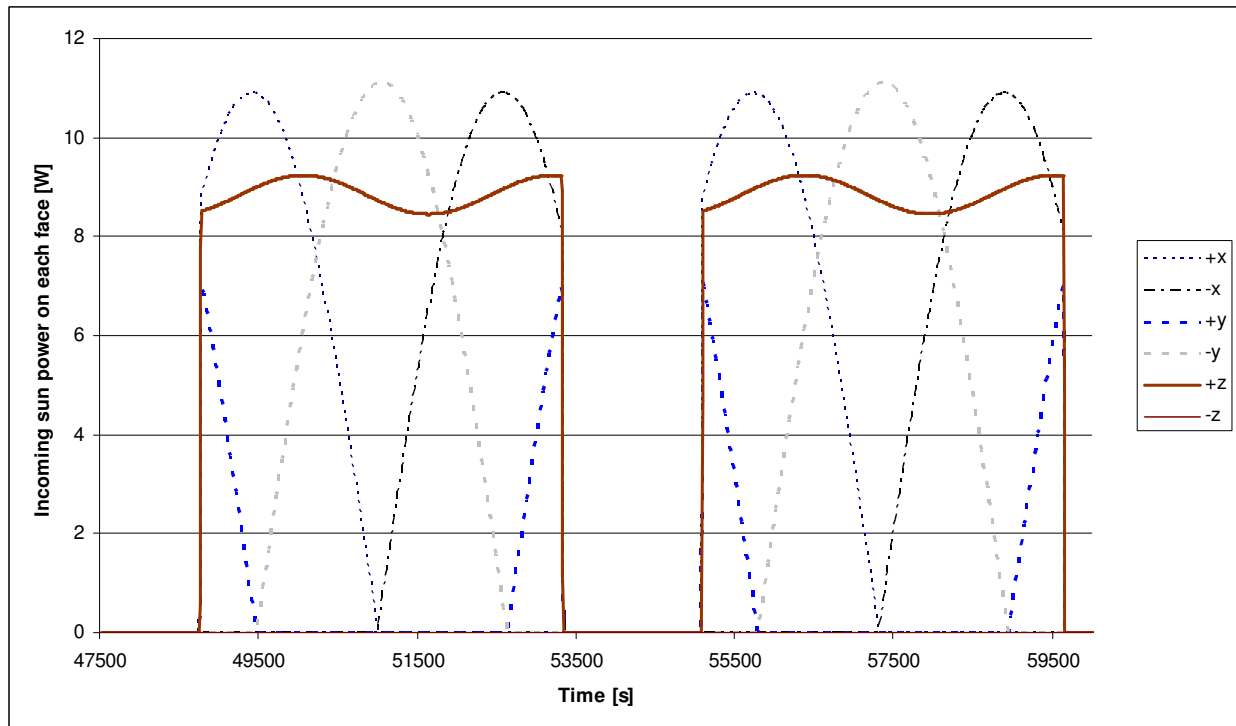
Fig 5: STK data: incoming sun power for each face-1000 km altitude

It is possible to observe the same time shift between STK model and the previous model as in calculations of the sun vector above.

An additional time lag appears in the eclipse time. In the previous model, eclipses time are calculate from the sun vector data in the orbital reference frame, the earth radius and the altitude. Time lag seen previously is transmitted to eclipses times calculations. In addition, the earth is considered as spherical. Those parameters can have an influence on eclipse times and lengths.

The direct sun power coming to each faces can be added in order to compare more easily those results (see Fig 6), the formula below has been used:

$$P_{Tot\_sun}(t) = \sum_{j=1}^{j=6} P_j(t)$$

New STK model and previous Matlab model results:
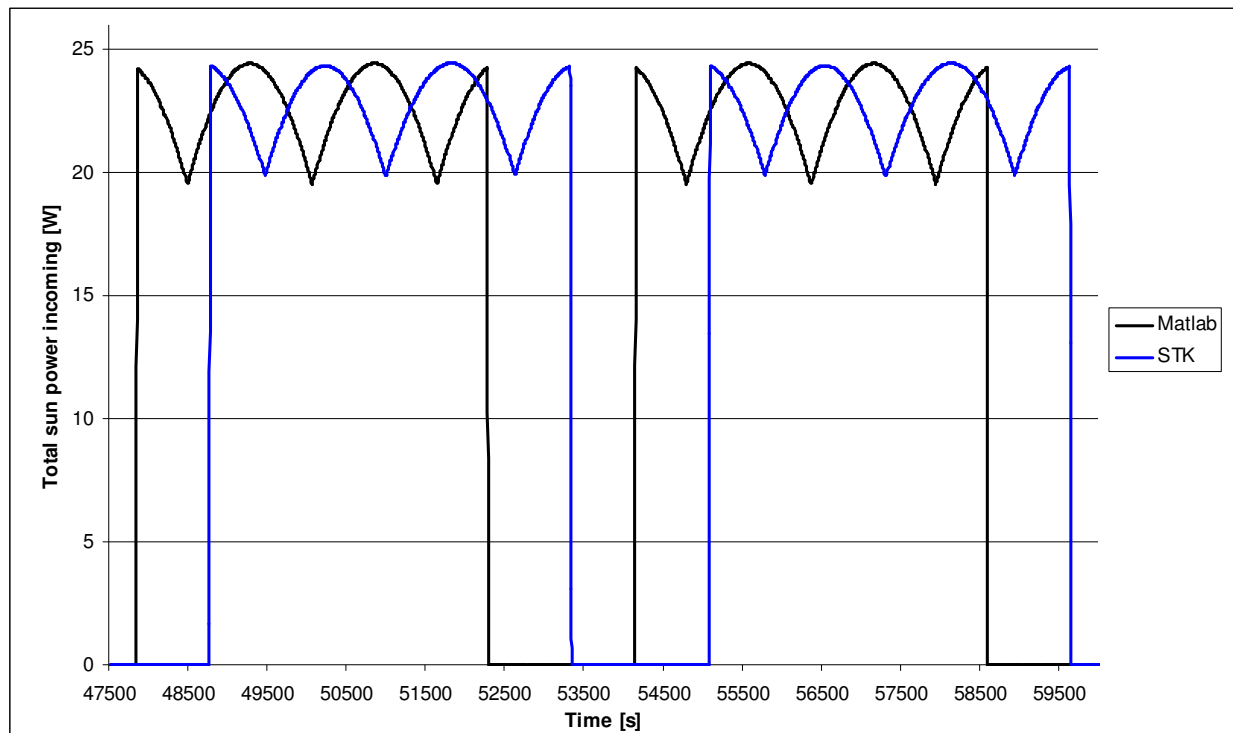
Fig 6: STK and Matlab result of the total sun power coming to the satellite function of the time.

Both curves have similar behavior, the time and period length being slightly different; to compare those results quantitatively, the total energy coming to the satellite from the sun during on cycle is computed:

$$E = \int_{t_0}^{T+t_0} P_{Tot\_Sun}(t) \cdot dt$$

Here t_0 represents the beginning of a sun cycle; T represents the sunlight length.

	STK	Matlab
t start [s]	48770	47860
t end [s]	53340	52290
Sun exposition time [s]	4570	4430
Total energy per cycle [J]	**104003**	**100589**

Table 1: Matlab and STK: sun radiation cycles comparison

Differences between the incoming flux from Matlab and STK (Table 1) are minor: 3.3 %, those can be explained by simplifications introduced in Matlab functions of the previous model which are taken into account in STK. Both models are thermally equivalent; it appears to be better to use STK in order to get more accuracy.

4   STEADY STATE THERMAL STUDY AND ORBITAL PARAMETERS

4.1   Steady state temperatures: cold and hot cases

4.1.1   Introduction

Since information about the launch parameters is not complete, it is necessary to identify worst orbital condition for a thermal point of view considering the altitude of the two possible orbits: Indian and Vega orbits.

Steady state calculation will be computed to evaluate the hottest and coldest temperature equilibrium and compare it with previous simulations results. The relevant parameter for the calculation is the **maximal/minimal** altitude of the orbit given in the table 1 below for Vega and Indian orbits. Theses altitudes are giving respectively the **cold/hot** steady states temperatures.

	Orbit	h_cold [km]	h_hot [km]
Vega launch	1200 x 350	1200	350
Indian launch	650 x 650	650	650

Table 1: Altitude of the satellite for calculations

The hot case would be the equivalent of placing the satellite in the sunlight for ever, the cold case pacing the satellite in the shadow for ever and see the equilibrium temperature.

4.1.2   Method

Heat flux equation is computed to solve the stationary case: the incoming and leaving power are equivalent at the steady state. The satellite external temperature is supposed as uniform.

Cold case:

- Incoming heat flux: earth emission
- Losses: emissivity of the external surfaces

The exposed surfaced is taken as minimum, just one face of the satellite can see the earth (see scheme 1). The real exposed surface to the earth emission is equal to three times this value, see exposition correction bellow (§ **exposition case**).

$$S_{sunexp} = 10^{-2} \, m^2$$

Scheme 1: Cold case : Satellite view from the earth from the satellite/ center of the earth axis

The coldest case is given when the satellite is the further away from the earth to minimize the earth radiation heat flux.
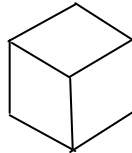
Hot case:

- Incoming heat flux:
  - Sun emission, earth emission, albedo
  - Internal power dissipation equal 3W

  (Power coming from the solar cells that can't be stored).

- Loses: emissivity of all external surfaces

The exposed surfaced to the sun is taken as maximum (Scheme2), three faces of the satellite are seen from the earth and the sun. The earth direction is the (1 1 1) vector in the satellite reference frame.

The equivalent exposed surface is:

S_{sunexp} = 10-2 x 3 x tan(30)= 1.73 10^{-2} m^2

Scheme 2: Hot case: Satellite view from the earth on the satellite/center of the earth axis and the sun

The next paragraph presents formulas used for the steady state calculation.

4.1.3 Formulas and parameters

Calculation of the incoming heat:

- Sun emission: $Q_{Sun} = G_{Sun} \cdot S_{Sun\,exp} \cdot \alpha_{vis}$ W

- Albedo emission: $Q_{Al} = G_{Sun} \cdot \alpha_{vis} \cdot a \cdot \dfrac{R^2}{(R+h)^2} \cdot S_{face} \cdot \sum_i \left((1 + \vec{E}' \cdot \vec{N}_i)/2 \right)$ W

- Earth radiation: $Q_{Earth} = G_{Earth} \cdot \alpha_{IR} \cdot \dfrac{R^2}{(R+h)^2} \cdot S_{face} \cdot \sum_i \left((1 + \vec{E}' \cdot \vec{N}_i)/2 \right)$ W

Calculation of the heat emission:

$$Q_{Leaving} = \sigma \cdot \varepsilon \cdot S_{Tot} \cdot T^4 \qquad\qquad W$$

$S_{sun\,exp}$ is the surface presented in Scheme 2m S_{Tot} is the sum of all satellite external faces, S_{face} is taken as 10^{-2} m^2.
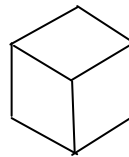
Parameters:

- σ Boltzman constant
- R : earth radius

- h : altitude of the satellite, specified in Table 1 (h cold and h hot).

- Solar constant G_{Sun}

- Earth IR constant G_{Earth}

- Albedo coefficient

- $\vec{E}$: earth unit vector

- $\vec{N}_i$: normal vector of the satellite, face i

Exposition case: Albedo and earth emission exposed surfaces:

Cold case   Hot case

Scheme 3: View of the satellite from the earth and the sun

The term $\sum_i \left((1 + E' \cdot N_i)/2 \right)$ is used as a exposition correction factor for heat flux coming from the earth. This term is equal 3 for the hot and cold case, it means that the satellite receives the equivalent of tree times the area of one side area (0.01 m^2) of albedo and earth radiation.

Sun emission exposed surfaces:

The sun is far from the satellite; the face seen by the sun is equal to the projection all faces surface on a plane perpendicular to the sun/satellite direction equivalent to the geometry in the scheme above (hot exposition case).

Calculation of the coefficients:

Emissivity (IR) = Absorptivity (IR): $\varepsilon = \left[\sum_i S_i \cdot \varepsilon_i \right] \cdot \dfrac{1}{S}$

Absorptivity (visible): $\alpha = \left[\sum_i S_i \cdot \alpha_i \right] \cdot \dfrac{1}{S}$

Here S is the surface of one face: 0.01 m^2, S_i are the external surfaces of panels, solar cells and frame, ε_i, α_i their corresponding emissivity/absorptivity. Result of the mean emissivity and absorptivity are presented in the table 2 below for two types of panel coatings: black paint and PCB.

	Emissivity	Absortivity
Black paint	0.845	0.795
PCB	0.840	0.727

Table 2: Emissivity and absorptivity results

4.1.4 **Results**

Using calculation, hypotheses and results of previous paragraphs, the steady state equations have been solved for Vega and Indian orbit, and two kinds of external panels coating using equivalent emissivity and absorptivity calculations (Table 2). IR emissivity is assumed to be equal to IR absorptivity (grey body approximation).

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		Tframe_cold [°K]	Tframe_hot [°K]
Vega	**Black paint**	191.9	331.8
	PCB	191.9	326.3
Indian	**Black paint**	199.3	329.0
	PCB	199.3	323.5

Table 3: Cold and hot steady states values

It is interesting to compare those results with the panel temperature given by previous simulations of the phase C [R1]. Those results are presented in the figure 7 below.
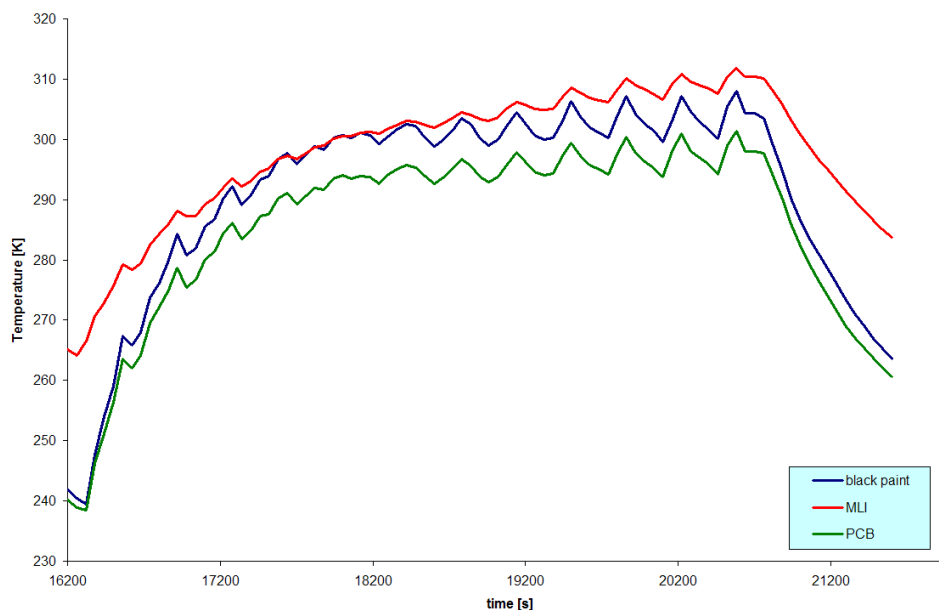
Fig 7: Nominal phase panel evolution temperature [R1]

Cold cases are colder than what was emulate by the model, in the previous orbit, the satellite was not far from staying long enough to reach the steady state. Emulate cases gave about -30°C, the steady state calculations gives: -80/-70 °C. The limit temperature could be -**40/-50°C colder** than in the simulation if the satellite stays in the umbra.

The **hot** temperature was about +30/45°C in simulations and almost reached the steady state. The steady state calculations gives about **15-20 °C higher results**: (Steady state temperature +50/60°C).

4.2 Orbital parameters scenario

4.2.1 Vega: cold and hot orbits

Steady state studies have provided temperature limits; the goal here is to set the unknown parameters of the Vega orbit: RAAN and perigee argument to create a hot and a cold Vega orbital scenario giving the extreme thermal conditions.

Here are the known Vega orbital parameters:

Elliptic orbit: 350 x 1200 Km, inclination = 71°, RAAN and argument of perigee unknown.

Hot case: the satellite has to stay under direct sunlight as long as possible; the satellite is in the sun when it is far from the earth.

Cold case: the satellite stays as far away from the earth as possible (so is moving more slowly) during the eclipse time so it has time to cool down.

The following identified as corresponding to the extreme cases:

- Cold case: Elliptic orbit: 1200 x 350 km, RAAN = 0°, i = 71°, perigee argument = 0°
- Hot case: Elliptic orbit 1200 x 350 km, RAAN = 90°, i = 71°, perigee argument = 0°

Simulations start the 21St March.

Erreur !
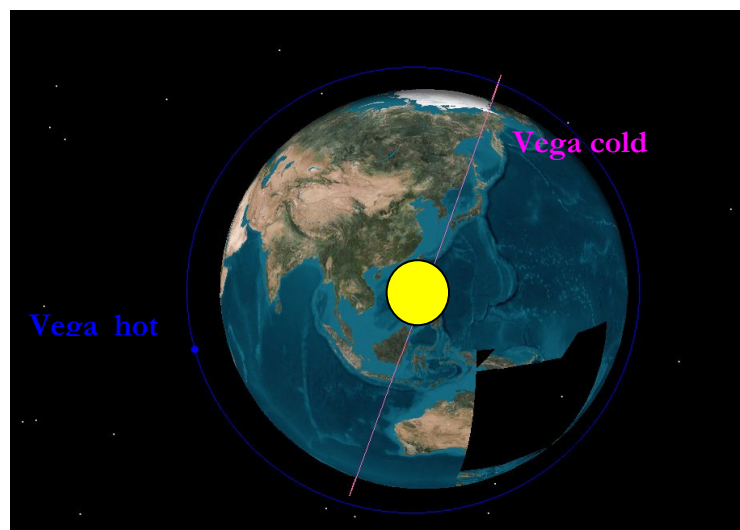
Scheme 4: Orbit position: Vega cold and Vega hot

The Vega hot orbit stays in the sun all the time, the RANN and perigee argument of Vega cold are set to provide the longest time far from the earth in the shadow.

4.2.1 Indian orbit:

This orbit is fully defined with the parameters below:

Elliptic sun synchronous orbit: RAAN = 179 °Inclination = 97.95 °, about 650 km altitude, eccentricity = 0.000384, argument of perigee = 5.55 deg.

4.3 Orbital parameters comparison and evaluation

The orbital parameters have been introduced into STK, the eclipse/sunlight length has been computed in the table 4 below. The mean total heat flux power has been computed for the eclipse and sunlight times. The mean heat corresponds to the average of the heat flux power during one period. This calculation has been done for the black paint panels. The previous orbit corresponds to the one used for simulations in [R1].

	Eclipse length [s]	Eclipse heat [W]	Sunlight [s]	Sunlight heat [W]	Mean heat [W]
Vega_cold	2200	4.0	3810	29.4	20.1
Vega_hot	None	None	All the time	28.9	28.9
Indian	2110	4.5	3770	29	20.2
Previous orbit	1730	4.1	4570	28.8	22

Table 4: Eclipse and sunlight incoming heat power

If we compare the orbital scenarios (Table 4) we can see that the mean incoming energy is almost the same during eclipses times and sunlight time. Comparing the mean heat received during the orbit, the hottest scenario is the Vega hot. This result was expected because the satellite is exposed at all times to the direct sunlight. The coldest scenario is the Vega cold orbit which should be a little colder than the previous orbit studied. The Indian orbit is in between those two extreme cases (Vega hot and Vega cold) so if the satellite can handle Vega cold and Vega hot orbit, it could also be launch on the Indian orbit.

Since the steady state wasn't reached on the previous launch parameters during the eclipse time, the coldest temperature of Vega cold should be below the previous studied orbit (longer eclipse time with slightly lower incoming earth radiation).

Vega_hot scenario has a mean incoming heat flux about 30% higher than in the previous orbits scenario, the hot temperature should be above the one of previous simulations.

With results of table 4 it is interesting to calculate what temperature the satellite would reach during the eclipse and sun times for Vega hot and Vega cold scenarios. Those temperatures could be compared with steady states temperature of Table 3 and should not differ much if scenarios have been defined as the most extremes. The formula below is used to calculate the equivalent steady state temperature:

$$Q_{mean} \cdot t = \sigma \cdot \varepsilon \cdot S \cdot T^4 \cdot t => T = \left(\frac{Q_{mean}}{\sigma \cdot \varepsilon \cdot S} \right)^{1/4}$$

Here S = 6 x 0.01 m^2, other parameters are the same than in 4.1.3.

Q_{mean} represents the mean heat received during eclipse or sunlight time respectively for the cold/hot cases. Vega cold is used for the cold temperature calculation, Vega hot for the hot temperature. 3W have been added to the Q_{mean} value of table 4 to take into account the energy dissipation on the motherboard.

	T_cold [°K]	T_hot [°K]
Vega_cold	193.1	
Vega_hot		324.6

Table 5: Steady states temperatures with calculated heat flux of Vega orbits

Vega orbits give almost the hot and cold cases of the steady states calculations. Small differences of the steady state temperature can be explained by the fact that a mean heat flux has been computed. It is not as extreme has the lowest/highest heat flux of the steady state, because the satellite is moving on its orbit.

5 SIMULATION PARAMETERS AND RESULTS

5.1 Simulation inventory

The model used is geometrically the same than the one developed on the previous semester project [R1]. Some spacer properties have been adjusted to emulate the motherboard with the existing model.

The model has twelve solar panels, two PCB cards and a battery box. Two types of simulation have been launched, one to study the motherboard thermal behaviour during the energy dissipation called **first type**. Another one has been used to emulate the behaviour of the electronic called **second type**. Satellite spinning is set to 1° per second on 3 axis rotation. First type and second type simulation are described in paragraphs below.

First type: used output: frame, panel, battery, motherboard

The nearest PCB from the frame is used to emulate the motherboard, the spacer properties have been set to emulate the contact between the frame and the motherboard (contact with a thin copper layer on the PCB) with some geometrical corrections. The energy dissipation is done in this card (3W or 1.4 W depending on the case if the satellite is in the sunlight) see heat scenario in table 6. In this type of simulation the second PCB is not used, his temperature being influenced highly by the motherboard energy dissipation. Spacers around the battery box are in POM.

Second type: used output: PCB1 and PCB2

The goal is to study the thermal range in PCBs (used in first type simulation to emulate the motherboard), different spacer materials have been tested: aluminum, POM, or one of both (the nearest one from the frame acting like a thermal filter). In the second type simulations, the energy dissipation of the motherboard is done in the frame in order to keep the same total internal energy dissipation**.** The motherboard in this model is not emulated anymore. Battery spacers are in POM.

Orbital parameters and internal heat repartition:

Three obits have been tested: Vega cold, Vega hot and the Indian orbit (defined in 4.2.1, 4.2.2).

A cold and a hot scenario have been used with internal heat dissipation and launch start parameters.(see Table 6).

Cold scenario: launch at the beginning of the eclipse time, 1.7 W of energy dissipation in sunlight.

Hot scenario: launch at the end of the eclipse time, 3 W of energy dissipation in sunlight.

Internal heat repartition is specified in the table below:

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	Launch start	Motherboad dissipation [W]	Electronique consumption (1 PCB) [W]	Battery heating system	Battery autoheating [W]
Cold scenario					
Launch phase		0	0.1	ON	
Nominal phase: sunlight	Start eclipse	1.7	0.1	ON	about 0.1W see appendix
Nominal phase: eclipse		0	0.1	ON	
Hot scenario					
Launch phase		3	0.1	ON	
Nominal phase: sunlight	Start begining eclipse	3	0.1	ON	about 0.1W see appendix
Nominal phase: eclipse		0	0.1	ON	

Table 6: *Internal heat generation parameters*

All three obits have been tested using conditions of scenario of table 6. The combination of orbital scenario and internal heat scenario is given in the table 7 below. Each cross corresponds to two simulations, one of the **first type** and two of the **second types (**Spacer scenarios see table 7bis). For the Indian orbit only one second type simulation has been done this orbit giving results similar to Vega cold orbit. All simulation results are shown and explained in the next paragraph.

	Orbit		
Scenario	Vega cold	Vega hot	Indian
Cold scenario	x		x
Hot scenario		x	x

Table 7: Orbit and scenario used for the simulations (both first type and second type)

Spacers	Vega cold	Vega hot	Indian
Frame to PCB1	Alu	Alu	POM
PCB1 to PCB2	Alu	Alu	Alu
Frame to PCB1	POM	POM	
PCB1 to PCB2	POM	POM	

Table 7bis: Spacer scenario used for second type simulations

5.2 Vega orbits results

5.2.1 Vega cold:

First type: output: frame, panel, battery, motherboard

Results for the Vega cold case orbit (first type simulation) show that temperature variations are important: -53°C to +52°C. The frame heats less than the panel, the frame acts as thermal link between internal components, so its heat is transmitted into internal parts. The battery heater keeps the battery above -6°C in the nominal phase, -13°C in the launch phase. The battery box does not heat much during the sun exposure. Its maximal temperature is about 34°C. The heating rate of the battery box is slowed compared to the frame and motherboard heating rate. That is due to the POM spacers placed around the battery box. The motherboard is following the panel temperature and is heating up until the eclipse start. Frame, panel, battery and motherboard temperature evolution can be seen on Fig 8, the maximal and minimal temperatures have been computed in table 8.
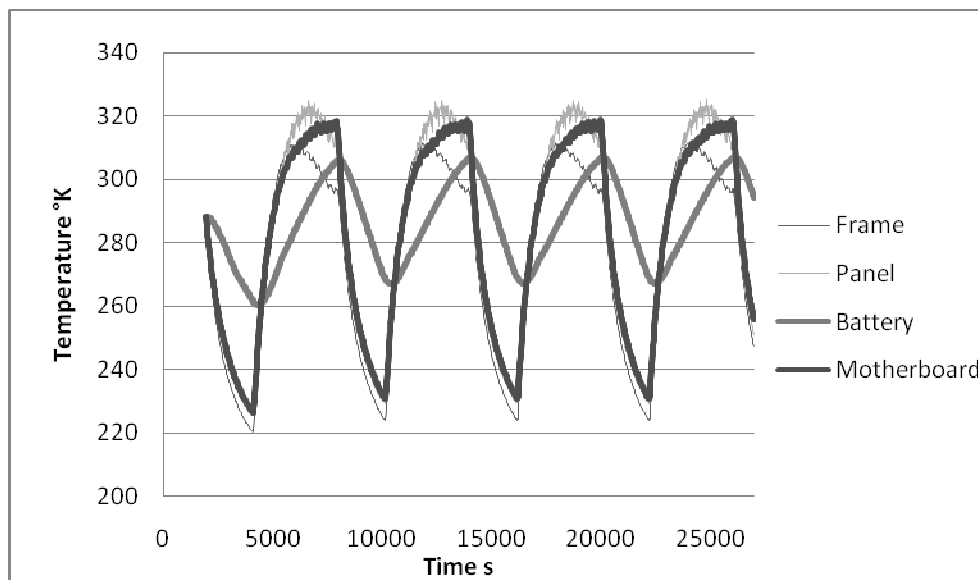
*Fig 8 : Vega cold orbit, cold scenario, four periods, **first type***

	Frame	*Panel*	*Battery*	*Motherboard*
Min °C	-53.0	-47.6	-13.0	-46.8
Min Nom °C	-49.3	-43.9	-6.0	-42.4
Max °C	38.5	52.3	33.8	45.9

Table 8 : Vega cold orbit, cold scenario, min, min nominal, max temperature

The battery box stays in its allowable temperature range (-20/+60°C), so does the motherboard (-40/+85°C).

Second type: output: PCB1 and PCB2

Results of the second simulation type for Vega cold have been carried out to study the electronic behavior for two different scenarios. POM or aluminum spacers linking the two PCBs have been tried. PCB1 refers to the PCB of the model which is near the frame, PCB2: the second one. Temperature variations are a lot smaller in PCBs using POM spacers. Temperature variations are smaller on PCB2. The first spacer acts as a thermal filter. The nominal temperature range is about 0°C to 50°C for PCB1 and 40°C to 53°C for PCB2 in the nominal range which is low compared to the -50 +52°C of the panel. POM spacers seem appropriate in this case but PCB2 heats already quite a lot compared to the frame temperature evolution.
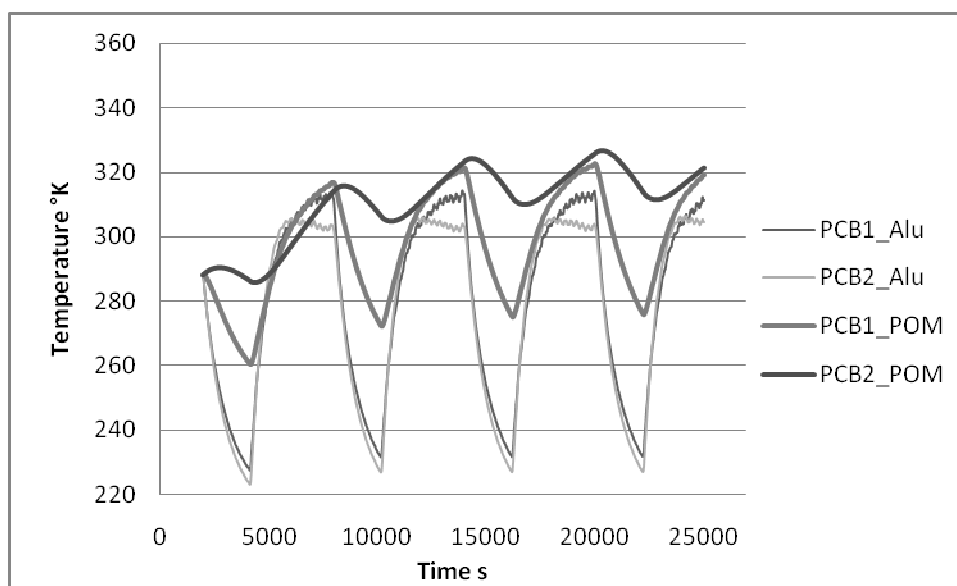
*Fig 9 : Vega cold orbit, cold scenario, four periods, **second type***

	PCB1_Alu	PCB2_Alu	PCB1_POM	PCB2_POM
Min °C	-45.8	-50.1	-12.8	12.7
Max °C	41.0	33.4	49.8	53.4

Table 8 : Vega cold orbit, cold scenario, min, min nominal, max temperature

5.2.2 Vega hot

First type: output: frame, panel, battery, motherboard

Vega hot results are presented in a similar way than Vega cold results and can be seen in fig 10 and table 10 below. Those conditions: orbit and scenario are the most extreme of the simulations planed. The satellite is always in the sun light, variations on the curves of the figure 10 are due to the orbital period. Since the orbit is elliptic, there are some altitudes variations. It affects the value of the total incoming heat flux acting on the albedo and earth radiation. The frame and panel temperature range here is about 13°C to 48°C, variations are a lot smaller than in the Vega cold case above. The power dissipation in the motherboard for this hot scenario is 3W, combined with the permanent sun light, it leads to heat up the motherboard to 62°C which is still in its allowable temperature range.
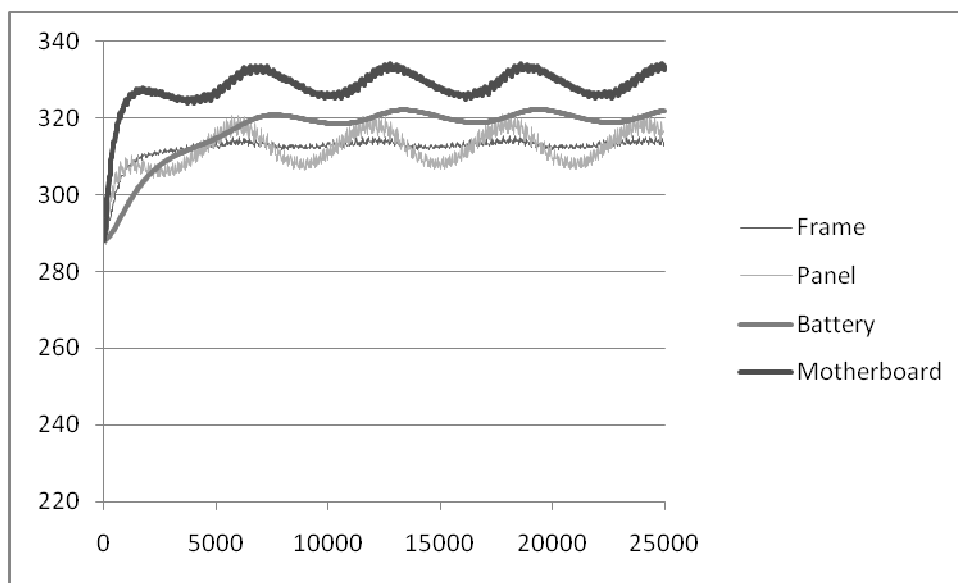

*Fig 10 : Vega hot orbit, hot scenario, four periods, **first type***

	Frame	Panel	Battery	Motherboad
Min °C	14.8	13.9	15.0	15.0
Min Nom °C	38.5	33.4	45.8	52.0
Max °C	42.2	48.2	49.2	61.3

Table 10 : Vega hot orbit, hot scenario, min, min nominal, max temperature

Second type: output: PCB1 and PCB2

POM and aluminium spacers have been tried to link the two PCBs of the model and are presented in the fig 11 and table 11 below. Temperature variations are small as in the first type simulations. POM spacer are insulating to much, PCB2 is heating up to nearly 85°C which is the limit of the alowable temperature range. In this case aluminium spacers seems more appropriate, maintaining the temperature of the PCB between 45°C and 55°C.
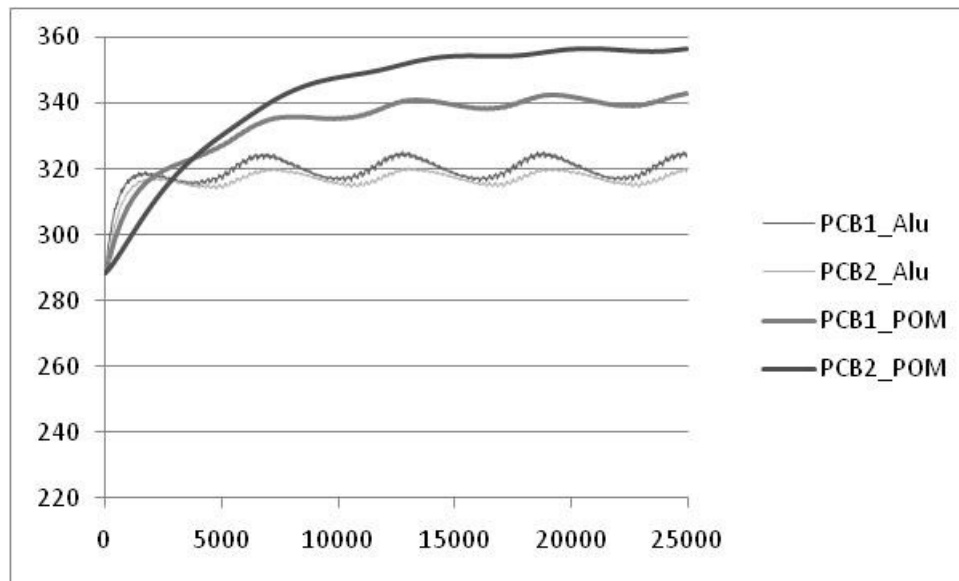
*Fig 11 : Vega hot orbit, hot scenario, four periods, **second type***

	PCB1_Alu	PCB2_Alu	PCB1_POM	PCB2_POM
Min °C	15.0	15.0	15.0	15.0
Max °C	52.1	47.5	69.9	83.3

Table 11 : Vega hot orbit, hot scenario, min, max temperature

5.3 Indian orbits results

Results for the Indian orbit (Table 12/13, Fig 12 /13)are expected to be similar as those of the Vega cold orbit in temperature amplitude, their eclipse and sunlight lengths being similar (see Table 4 and paragraph 4.3). The temperature range of the panels, frame is about -50°C/+50°C for the hot and cold cases. The variation range is similar with Vega cold (see results in table 8 and figure 8). In the Indian orbit case, only the motherboard energy dissipation and the launch start vary between the cold to the hot scenarios. The temperature range of Indian hot case is 10°C higher than Indian cold case. In the hot Indian case, frame and panel reaches similar maximum temperature compared with the Vega hot case.

Indian cold case:
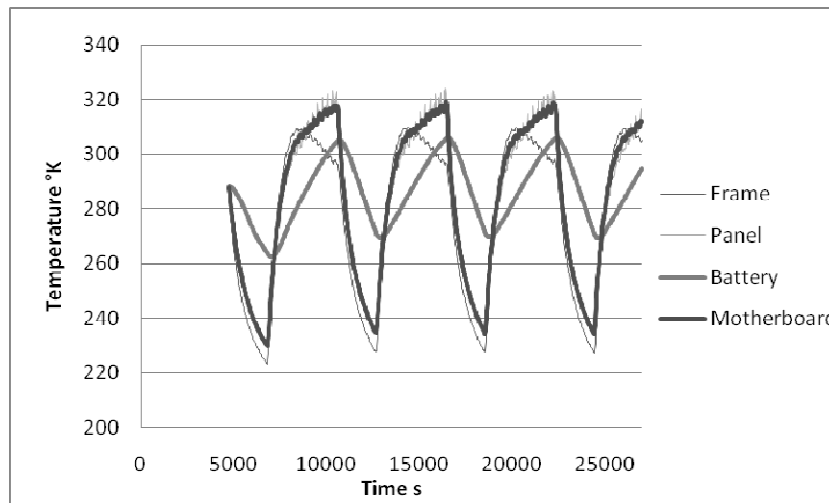
Fig 12 : Indian orbit, cold scenario, four periods, **first type**

	Frame	*Panel*	*Battery*	*Motherboard*
Min °C	-49.9	-43.4	-10.8	-43.3
Min Nom °C	-45.9	-39.3	-3.8	-38.9
Max °C	37.0	50.8	33.0	46.5

Table 12 : Indian orbit, cold scenario, min, min nominal, max temperature

Indian hot case:
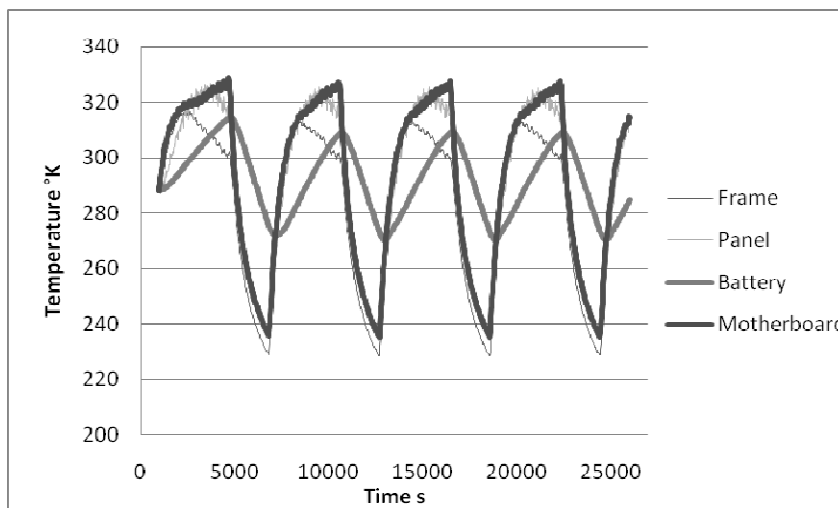
Fig 13 : Indian orbit, hot scenario, four periods, **first type**

	Frame	*Panel*	*Battery*	*Motherboard*
Min °C	-45.0	-39.0	-3.0	-37.9
Max °C	43.7	53.7	41.3	55.7

Table 13 : Indian orbit, hot scenario, min, max temperature

Indian cold case: second type simulation

In result of the temperature range of the PCB from Vega second type simulations 5.21 and 5.22. A different simulation has been launched for the Indian orbit with the first spacer in POM (connecting PCB1 to the frame) and the second spacer in aluminum (in between the two PCBs). The goal of the first spacer is to act as a filter in order to get lower temperature variations. The second spacer is set in aluminum, to keep a good conductivity in between cards and avoid the most internal card to heat up, each card dissipating 0.1W by joule effect in the simulations. Results are presented in the figure and table 14 bellow. The second PCB is following the temperature of the first one. Heating/cooling rand are reduced with is important for the electronic. The temperature range is 17°C to 46°C in the nominal range.
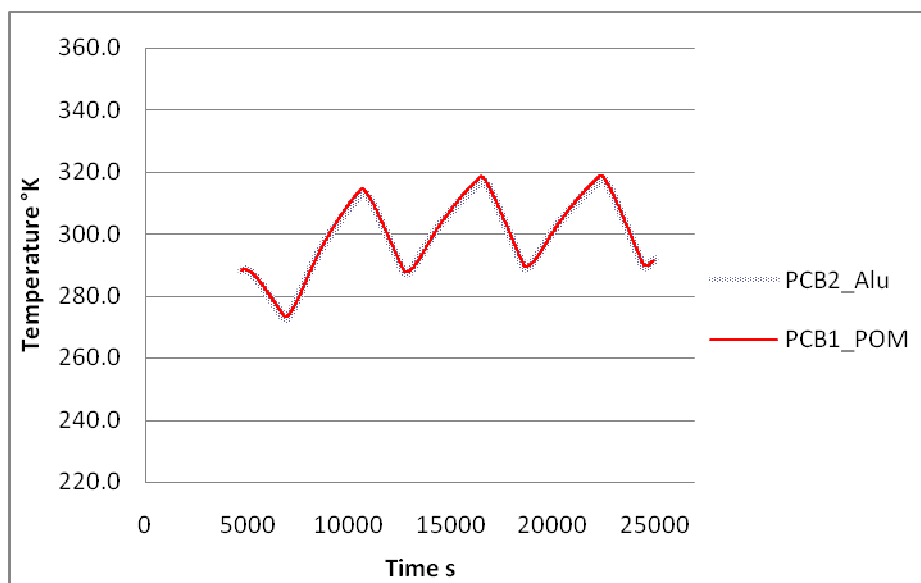

*Fig 14 : Indian orbit, cold scenario, four periods, **second type***

	PCB1_POM	PCB2_Alu
Min °C	0.1	-0.5
Max °C	45.8	44.5

Table 14 : Indian orbit, cold scenario, min, max temperature

5.4 Thermal design and comments

General comments: for all orbits, the battery heating system is sufficient to keep the battery above -13°C (Vega cold), the logic activates the heating system below -3°C, see logic in appendix IV. The maximal battery temperature is 49°C for Vega hot. The battery should then stay in between its temperature range: -20/+60°C at all times.

Temperature variation ranges are very different from Vega hot and Vega cold and are similar for Vega cold and the Indian (cold case) orbit as expected before. The maximum temperature range for the frame is -53/+42°C. Heat is taken from the frame to heat up the electronic and battery box, which acts on the frame temperature. The temperature of the graphs has been taken from points of the structure; small differences can be expected especially on the frame (top curve shape) which acts as thermal bridges between the various parts of the structure. The panel could reach 54°C.

The electronics have to stay in a range from -40 to 85°C, different scenarios using various spacer materials have been tried. POM spacer linking PCB cards to the frame and PCBs together gives good results on Vega cold decreasing temperature variation and heating/cooling rate that affect the electronic. Used on Vega hot orbit, the PCB cards heat too much, the internal cards would reach temperatures such as 83°C. A solution is to use POM spacers as link between the first PCB and the frame, other spacer in between PCB cards should be set in aluminum. This solution has been tried on the Indian scenario orbit and gives satisfying results; the first POM spacer acting has a thermal filter decreasing thermal variation and heating/cooling rates. The second PCB seems to follow the first one, the connection between those two being about 500 times better than the one linking the electronics to the frame.

The motherboard temperature range during heat dissipation is -46 to 61°C, 61°C being reached with 3W energy dissipation which is an extreme case. An additional simulation has been run placing 1mm thick POM spacers ensuring the thermal contact between the motherboard and the frame. It did not improve the situation, the motherboard reaching 90°C. The best solution would be to place some additional thermal drains linking the energy dissipaters to cooler parts inside or to the frame to avoid hot points. Ideally, the thermal dissipation should be done by those, then to insulate the motherboard with POM spacer to decrease heat cooling and heating rates (if energy dissipaters are not thermally linked to the frame). An additional simulation has been carried out with 1mm thick POM spacers linking the motherboard to the frame, this insulation heated up the motherboard way to much.

The temperature evolution in Vega hot is due to the albedo and earth radiation variation, this orbit being slightly elliptic.

6 CONCLUSION

The previous model has been upgraded to be linked to STK and has become more flexible, the accuracy of the previous boundary condition has been checked. After a steady state study, orbital parameters have been set in order to get the most extreme Vega orbit according to the provided orbital information which was not complete. Scenario have been launched, results are very different between the Vega cold orbit and Vega hot orbit. Vega cold orbit and the Indian orbit give similar results for the cold case.

According to simulation results, POM spacer should be used around the battery box and as thermal filter to link the first PCBs to the frame. Aluminum should be used for other spacer linking the other PCB cards. Energy dissipaters on the motherboard should have a thermal drain to avoid the motherboard to heat up too much. If those drains are sufficient to drain enough heat, it could be a good idea to put thin POM spacers between the motherboard and the frame. The battery heating device seems sufficient to keep the battery above -6°C.

Those results are given by FEM calculation, the internal radiation is not taken into account in this model. Structural modifications should follow validations steps by proper thermal testing in order to be validated. Internal radiation could give important heat flux contribution depending on the internal design.

7 ACKNOWLEDGEMENTS

8 REFERENCES

[R1]   Yassir Madhour, *Phase C Thermal Tests report*

[R2]   Gilmore, D.G., *Satellite Thermal Control Handbook*, The Aerospace Corporation., California 1994.

[R3]   Cédric Blondel, *Phase C Thermal Tests report*

[R4]   Gregoire Bourban *Thermal calculations on Labview*

[R5]   Oscar de la Torre, *Phase B Thermal Management report*

[R6]   Stefano Nebuloni, and Ricardo J. da Silva Lima, *Thermal Management phase A report*

9 APPENDIX

9.1 Appendix *I: STK data collection / COMSOL to Matlab connection schemes.*
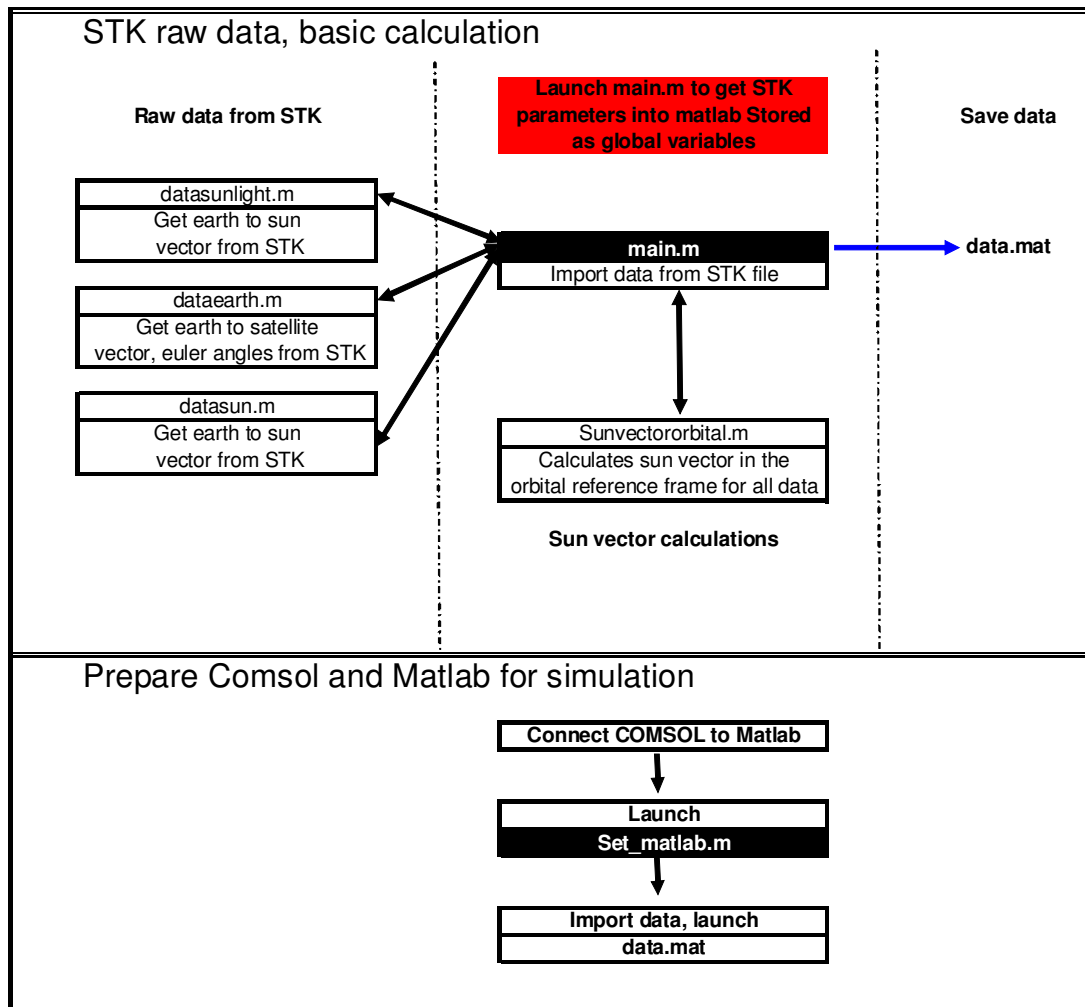
9.2 Appendix *II Boundary function description*

Open a COMSOL model, link COMSOL to Matlab in the File menu. It will open Matlab. Then run Set_matlab.m to declare and set some global variables used after in calculation, open the right Workspace Matlab data file saved previously in step one (*.mat). Then everything is ready to run a FEM simulation.

9.3 **Appendix** *III Boundary function description*

Data linear interpolation and data

datasunlightlinear.m
Linear interpolation of the earth to sun vector data

datalinearearth.m
Linear interpolation earth to satellite vector data

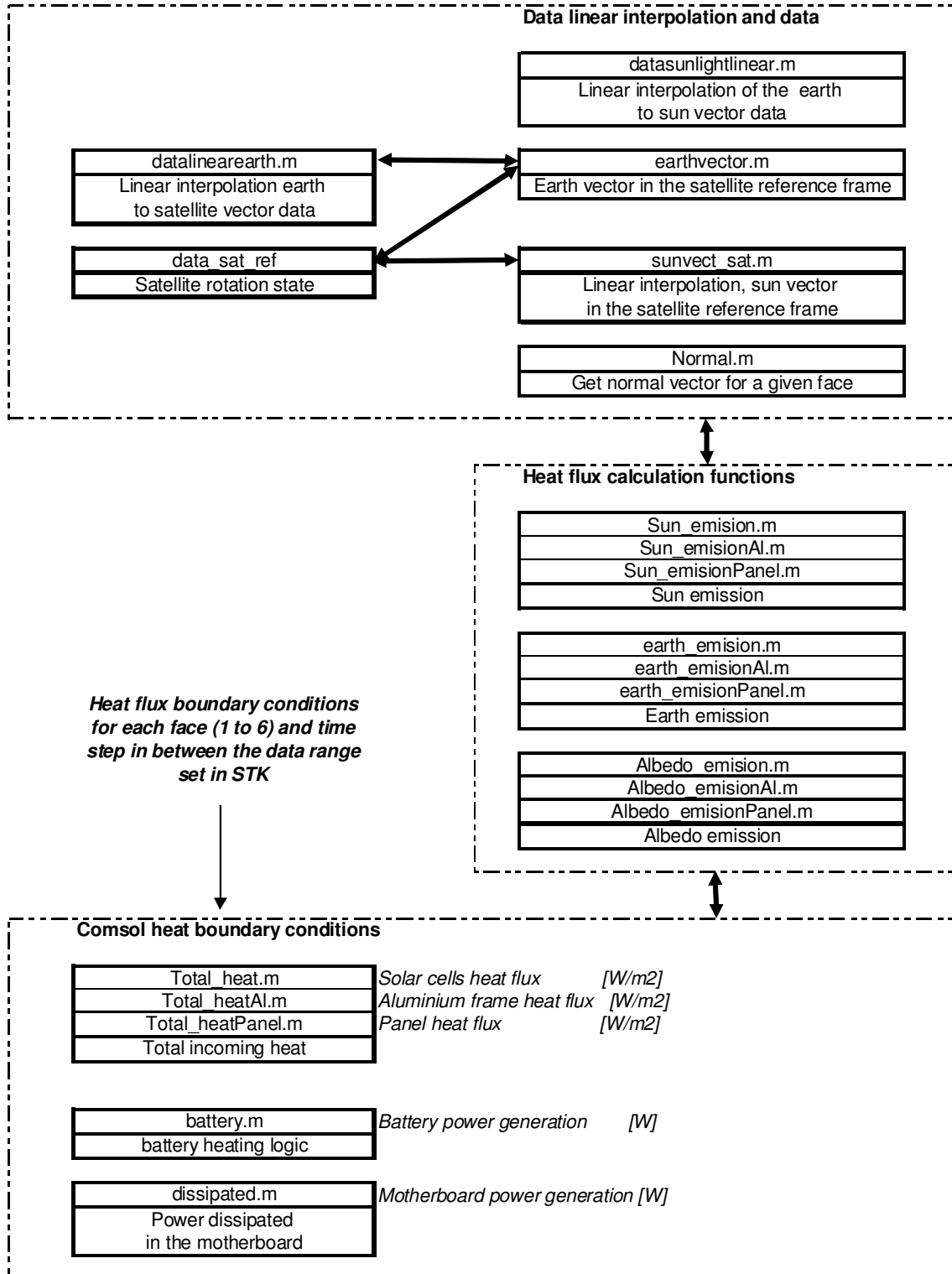| earthvector.m |
|---|
| Earth vector in the satellite reference frame |

data_sat_ref
Satellite rotation state

sunvect_sat.m
Linear interpolation, sun vector in the satellite reference frame

Normal.m
Get normal vector for a given face

Heat flux calculation functions

Sun_emision.m
Sun_emisionAl.m
Sun_emisionPanel.m
Sun emission

earth_emision.m
earth_emisionAl.m
earth_emisionPanel.m
Earth emission

Albedo_emision.m
Albedo_emisionAl.m
Albedo_emisionPanel.m
Albedo emission

Heat flux boundary conditions for each face (1 to 6) and time step in between the data range set in STK

Comsol heat boundary conditions

Total_heat.m	*Solar cells heat flux [W/m2]*
Total_heatAl.m	*Aluminium frame heat flux [W/m2]*
Total_heatPanel.m	*Panel heat flux [W/m2]*
Total incoming heat	

battery.m	*Battery power generation [W]*
battery heating logic	

dissipated.m	*Motherboard power generation [W]*
Power dissipated in the motherboard	

COMSOL launch steps:

Ref.: **S3 Phase C reportV4 - Thermal Simulations.doc**

9.4 Appendix IV: Battery heater and heating logic

A resistor is soldered on the copper plate on one side of the battery box. On the opposite side of the box there are two thermal sensors. Both have the same function (redundancy). When the temperature decreases below Tstart, the heating resistor is switched on until the battery box reaches Tstop. 500 mW are dissipated in the resistor. This energy is taken directly from the one of the battery so the battery is also self heating. Tstart is set as -3°C, Tstop is +2°C.
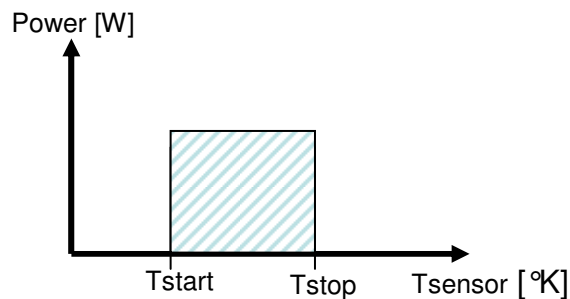
Battery resistor logic scheme

Battery autoheating, when current is taken out of the battery, the reaction is exothermic which generate heat. Measurements have been done and this effect is emulated here by a linear law. Autoheating is about 0.15 W at -40°C and 0.05 at +40°C and is emulated by a linear law.