1. Sizing ValiSat's Thermal Control System (TCS)

1.1. Introduction

Below is a simple method used to size the model for ValiSat's Thermal Control System. This tutorial is linked to the ValiSat spacecraft model, which will allow you to better understand the design process and create your own similar satellite model in Valispace.

The first step to consider when sizing the Thermal Control System (TCS) of a spacecraft is the thermal environment, which is generally dictated by its orbit; inclination with respect to the Sun, distance from Earth, eclipse time and as well as other factors like the material of the structure.

First, it is necessary to find the net head load (Q_net) going into or out of the spacecraft during the two worst scenarios; called the hot case and cold case. From the net heat load an equilibrium temperature can be determined and thus active or passive heat control methods can be implemented in order to ensure the spacecraft temperature never exceeds an upper and lower bound. These bounds are generally determined by the functioning temperature of the sensitive electronics on-board.

2. Orbital considerations

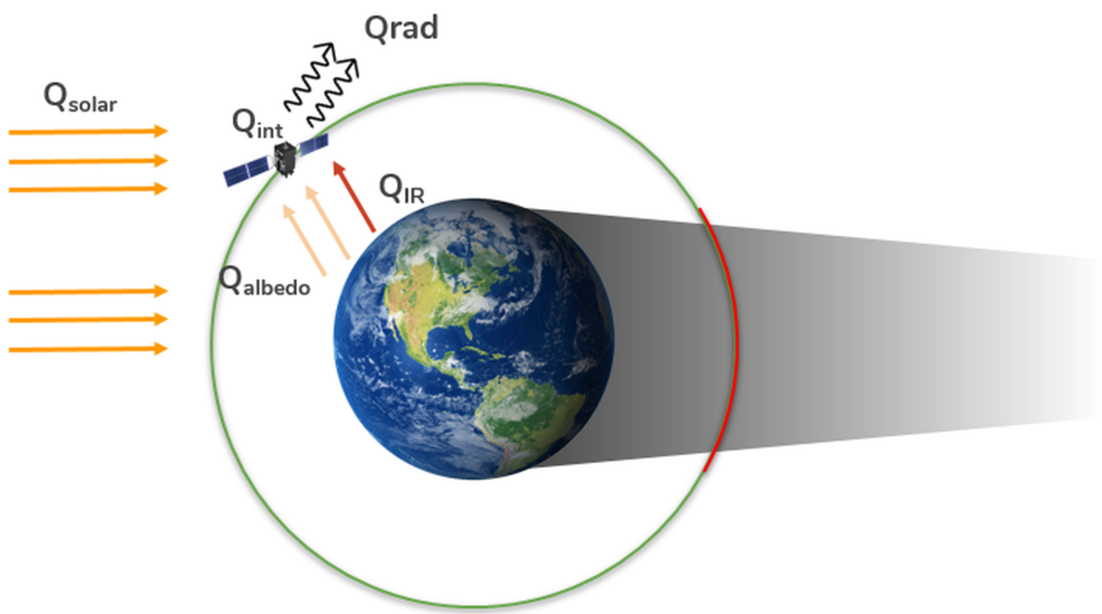
2.1. Net heat

The net heat load of the spacecraft is the sum of all contributions going in and out of the spacecraft at any point during it's orbit. For ValiSat, a circular orbit was assumed with no inclination with respect to the ecliptic.

The net heat load is given by the equation;

$$Q_{net} = Q_{incident} + Q_{albedo} + Q_{IR} + Q_{int} + Q_{rad}$$

where Q_net is the total heat load per second across the spacecraft area. The individual contributions due to the incoming solar radiation (Q_incident), the reflected solar radiation from Earth (Q_albedo), the heat absorbed from the Earth's infrared emissions (Q_IR), the heat load generated internally by the spacecraft components (Q_int), and the heat loads radiated to deep space (Q_rad).



2.2. Hot Case

First the individual heat loads in the hot case must be calculated. The following equations are included in the ValiSat model but can be more accurately modeled with further investigation on the part of the user to include varying attitude, generalised orbits and different spacecraft materials.

2.2.1. Solar radiation

The heat load due to solar radiation (Q_incident) is simply the Solar Constant in Earth orbit (1,370 W/m^2) multiplied by the area of the satellite exposed.

$$Q_{incident} = AF_{\odot}$$

where A is the area of the satellite facing the Sun (assumed to be the constant half of the lateral area of the cylindrical satellite for Valisat).

2.2.2. Earth albedo

The heat load due to the albedo of the Earth depends on the altitude of the orbit. The Earth can be viewed as a sphere that emits the (reflected) light from the Sun. To determine the flux at the surface of the Earth (F_albedo), one must consider that the Earth is seen as a disc of area of pi * R_E to the Sun's rays, but that in reality, this flux is spread across half the total area of a sphere. Thus the flux of reflected light at ***the Earth's surface*** is simply the ratio of these areas times the incoming solar flux.

$$F_{albedo} = F_{\odot} \frac{\pi R_E}{\frac{4}{2}\pi R_E} \equiv \frac{F_{\odot}}{2}$$

Now, after reflection, the flux received across the spacecraft is reduced by the Earth's albedo (approximately *0.3*) and with the inverse square law to the altitude of its orbit. Thus, assuming a circular and constant orbit, the total heat load is given by

$$Q_{albedo} = \alpha A F_{albedo} \frac{R_E^2}{(a + R_E)^2}$$

where, this time, A represents the area of the satellite facing the Earth (also assumed the constant half lateral area), alpha is Earth's albedo, R_E is the radius of the Earth (*6378.0 km*) and a is the altitude of the satellite (*1122.0 km*).

2.2.3. Earth infrared

The extra heat received due to the Earth's infrared emissions is calculated in the same way, using the inverse square law.

$$Q_{IR} = A F_{IR} \frac{R_E^2}{(a + R_E)^2}$$

where F_IR is the infrared flux emitted at the Earth's surface (reference value of *237.0 W/m^2*)

2.2.4. Internal heat load

For an early phase analysis, the internal heat load required to be dissipated by the spacecraft is the peak power load (*660.0 W*) multiplied by an efficiency factor (assumed 1.0). Clearly, this is an over-sizing that should be refined in later designs as ValiSat will only operate with a peak power load for short periods during its mission.

$$Q_{internal} = \eta P_{peak}$$

2.2.5. Radiated heat load

The naturally radiated heat load is perhaps the most complicated. The following calculation assumes that half of the spacecraft radiates to deep space, while the other side faces the Sun. This is approximately true, but does not take into account the eclipse time or potential attitude variations of the satellite, however, this is expected for a 1st iteration design.

$$Q_{radiated} = \frac{A}{2}\sigma T_{desired}^4$$

where T_desired is the *desired* temperature of the inner electronics (assumed to be *293.0 K*).

2.2.6. Total heat loads in the hot case

The above calculations result in the following values for ValiSat heat loads in the hotcase.

Q values	Vali value
Q_incident	*9.799 kW*
Q_albedo	*768.697 W*
Q_IR	*1.228 kW*
Q_internal	*810 W*
Q_radiated	*-90.395 W*
Q_Net	*8.095 kW*

2.2.7. Equilibrium temperature

The equilibrium temperature of the spacecraft in the hot case can be easily calculated using the following equation;

$$T_{eq} = \left(\frac{Q_{net}}{\frac{A}{2}\sigma} \right)^{\frac{1}{4}}$$

Without any active or passive controls, in the hot case, the worst case equillibrium temperature is *339.146580791 K*, which is clearly too hot for electronics (extreme survival range is generally -10 to +40 degC). Thus, thermal control methods must be employed to reduced this temperate into the operational range.

2.3. Design

2.3.1. Multi-Layer Insulation (MLI)

The addition of MLI works to insulate the spacecraft from the external environment. Several models can be used, and mostly they are empirical. The following inputs were chosen for the ValiSat MLI.

Variable	Value
Number of layers	*36*
Mass density	*0.05 kg/m**2* / layer
Absorptivity in the IR	*0.015* *
Absorptivity in the visable	*0.3*
Emissivity in the IR	*0.015* *

*Summary of MLI inputs and reference values. *Equal through Kirchoff's Law*

With the addition of MLI, the new absorbed heat loads of the spacecraft can be calculated.

Radiated heat load with MLI:

In the hot case, the amount radiated by the spacecraft can be calculated using the following formula, taken from M. Diaz-Aguado, *et al.* 'Small satellite thermal design, test, and analysis', SPIE 2006.

$$Q_{radiated,MLI} = \frac{A\sigma(T_1^4 - T_n^4)}{(n-1)(2/\alpha_{MLI,IR} - 1)}$$

where A is the total radiating (non-solar pointing) spacecraft area, T_1 is the temperature of first layer (assumed to be the desired spacecraft internal temperature) and T_n is the temperature of the outer most layer, *n* is the number of layers and \alpha_MLI is the absorptivity in the IR, as this is the regime in which a warm spacecraft will radiate its energy. The number of layers can be adjusted in order to reduce the radiated heat load. At *36.0* layers, the radiated load is just *0.39044549024 W*.

Absorbed heat loads with MLI:
The absorbed heat through the albedo and incident contributions are simply altered by the addition of a reduction factor that represents the absorption (alpha_MLI,v) characteristics of the MLI, as shown below.

$$Q_{absorbed,MLI} = \alpha_{MLI,v}\left(Q_{albedo} + Q_{incident}\right)$$

Absorbed IR heat load with MLI:
The same is true with the infrared contribution which becomes;

$$Q_{IR,MLI} = A\alpha_{MLI,IR}F_{IR}\frac{R_E^2}{(a+R_E)^2}$$

Thus, the new heat loads experienced across the spacecraft are given in the table below.

Q values after addition of MLI	Value
Q_absorbed (Q_incident + Q_albedo)	*3.17 kW*
Q_IR	*18.415 W*
Q_internal	*810 W*
Q_radiated	*0.39 W*
Q_net	*3.999 kW*

Now, the new worst case equilibrium can be calculated, as before. The temperature is *284.319801382 K*, which is in a much more acceptable range for the electronics onboard.

Assuming a simple weight per layer of *0.05 kg/m**2* then gives the total mass required for the MLI of *38.8511967011 kg*.

2.3.2. Radiators

Clearly, with a net heat load across the spacecraft, radiators will be required to dissipate excess heat. A few simplifying assumptions are made: that the radiators operate in the worst case at a temperature of T_r = *313.0 K* and are orientated away from the Sun (toward deep space and the Earth).

The following method is taken from the Space Mission Analysis and Design handbook (page 454-456). It should be clear that if one assumes the spacecraft is thermally isolated from its environment due to the addition of MLI then for a radiator, the heat loads and in out must obey the following

$$Q_{internal} + F_{absorbed,r} A_r = \epsilon \sigma_r T_r^4$$

where F_absorbed is the flux absorbed ***across the radiator*** *(*sum of albedo, *148.396568832 W/m**2* and infrared, *171.39369792 W/m**2),* A_r is the required area of the radiator and epsilon_r is the emissivity of the radiator in the infrared (assumed *0.92*). Knowing the hottest case operational temperature, this equation can then be solved for the radiator area.

As with the MLI, a simple weight per meter squared of *3.3 kg/m**2* is known, which gives the total mass required for the radiators as shown in the table below.

Variable	Value
Radiator Area	*4.478 m**2*
Radiator Mass	*14.778 kg*

Summary of the radiators

2.4. Cold case

Now that it is clear that the spacecraft can easily survive the hotcase, it is necessary to check that with the maximally sized radiators the spacecraft doesn't exceed a minimum temperature. In the cold case, it is assumed that no heat is received from the Sun either directly or via albedo of the Earth (this is not true for all generalised orbits). The net heat loads are then as follows:

Q values after addition of MLI	Value
Q_absorbed (Q_incident + Q_albedo)	*0 kW* (no sunlight)
Q_IR	*18.415 W*
Q_internal	*610 W*
Q_radiated (through MLI)	*-6.708 W*
Q_net	*621.707 W*

It is simple to see that except for the internally generated (nominal) power, the heat loads across the spacecraft in the cold case are negligible, meaning that ValiSat is essentially thermally insulated from the environment by the MLI. Now, by re-arranging the previous radiator balance equation to solve for the cold case temperature of the radiator, one attains:

$$T_r = \left(\frac{Q_{internal} + F_{r,IR}A_r}{\epsilon_r \sigma A_r}\right)^{\frac{1}{4}}$$

Thus, by inserting the radiator area just calculated, the minimum equilibrium temperature can be determined as *277.11305985 K*. Even with a margin applied, this is above the minimum temperature of the electronic components onboard.

Therefore, with ValiSat in this specific orbit and assuming a nominal power load during the eclipse, no heater is required to stay above the minimum desired temperature of 260K. However, as the design progresses, this value must be monitored in order to ensure that if the internal power generation dips during the eclipse time that a minimum temperature is always maintained via use of a heater or similar. For now, no active heating methods are required for ValiSat.

2.4.1. Harness (Heat pipes, etc.)

Finally, in order to account for the mass of the system (heat pipes if required at a later stage, nodes, mounts etc.) a *2.0 percent* of the total dry mass of the spacecraft is applied. Thus the TCS harness mass is *15.6255308503 kg*, to be added on top of the total mass of *53.6292856244 kg* for the passive MLI and radiator components.