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OSTM/Jason-2 Mission

Assignment 5: Thermal Control Subsystem

Group 25

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Contents

Acronyms

AMR Advanced Microwave Radiometer. 8

DHU Data Handling Unit. 6

DORIS Doppler Orbitography and Radiopositioning Integrated by Satellite. 8

JASON Joint Altimetry Satellite Oceanography Network. 4

MLI Multi Layer Insulation. 2, 4–7, 10

NASA The National Aeronautics and Space Administration. 10

OSTM Ocean Surface Topography Mission. 4

POSEIDON Positioning,Ocean,Solid Earth, Ice Dynamics, Orbital Navigator. 8

PROTEUS Plateforme Reconfigurable pour l'Observation, les Télécommunications Et les Usages Scientifiques. 4–7, 9, 10

SHM Safe Hold Mode. 7, 9

SSM Secondary Surface Mirrors. 2, 4–6

STA Star Tracker Array. 4, 6

T2L2 Time Transfer by Laser Link. 8

TCS Thermal Control System. 4, 6, 7

1 Architecture

Understanding how the Thermal Control System (TCS) design of PROTEUS was dimensioned is critical to advance in the analysis of the Jason-2 mission. This subsystem was designed to withstand the required maximum thermal loads with margins, relying on both passive regulation, such as radiators, and active regulation through the use of heaters.

OSTM/JASON 2 mission consists of two main components: the Proteus platform and the payload module. The bus system has been equipped with various solutions to regulate system temperature effectively and maintain operational stability. Additionally, in order to ensure payload safety and health, PROTEUS provides thermal control and heater power to the payload in all satellite modes.

It is noteworthy that PROTEUS has been designed to meet the versatile demands of different missions. For this reason, the generic design of this platform must endure a wide range of environments, with orbits spanning altitudes from 500 to 1500 km and various inclinations. These requirements impose strict demands on the Thermal Control System, which must keep all equipment within specified temperature ranges despite significant fluctuations in thermal loads [7].

To address these challenges, a passive thermal control design has been adopted, featuring strategically positioned radiators aided by a software-controlled active heater system.

The thermal control components used on Proteus are:

- External and internal Multi Layer Insulation (MLI)
- Heater lines and thermistors
- Silvered Secondary Surface Mirrors (SSM) Radiators
- Thermal straps and insulating washers
- Aluminium doublers, which spread heat in the radiators.
- White paint on the dedicated cylindrical radiator of the launcher adaptor.

The active regulation uses heaters located on the panels of the platform (or on some specific equipment, such as the battery) and is monitored by the on-board computer. Temperature sensors located nearby the electric heaters send to the computer the difference from the set-point temperature. Consequently, this computer applies the power needed to reach the operational target.

The thermal control of Jason-2 is divided into 5 distinct parts each one as much uncoupled as possible from the others: [9]

1. Battery zone
2. Propulsion panel
3. Main platform zone, excluded Battery and Propulsion
4. Star Tracker Array (STA)
5. Payload Module, including scientific instruments

In order to ensure efficient thermal decoupling between the Proteus platform and its payload, four mechanical links that are situated on the upper faces of four pods act as an interface.

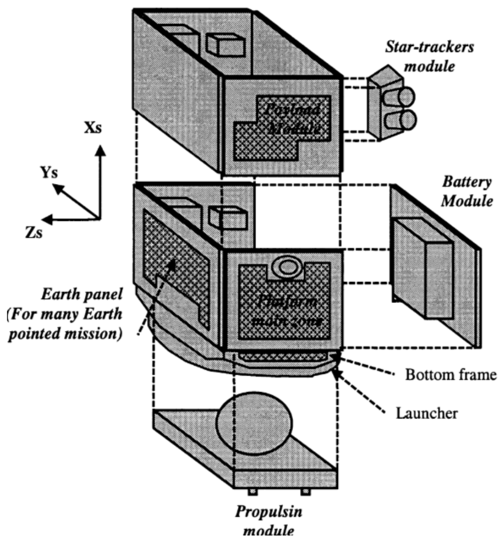

Fig. 1: TCS Components' Design

Figure 2 depicts the payload module of Jason-2, which has a cubical shape with no central structure, mirroring the platform's structure. The cube's panels serve as both structural support and heat rejection surfaces for module thermal control.

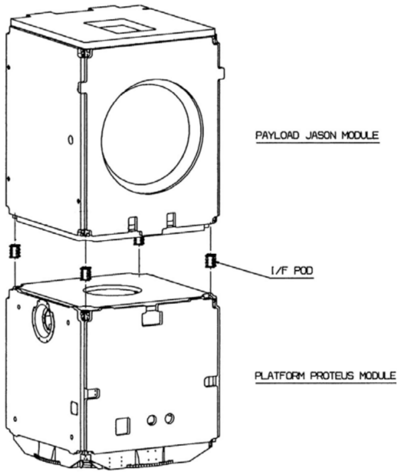

Fig. 2: Interface PROTEUS - Payload

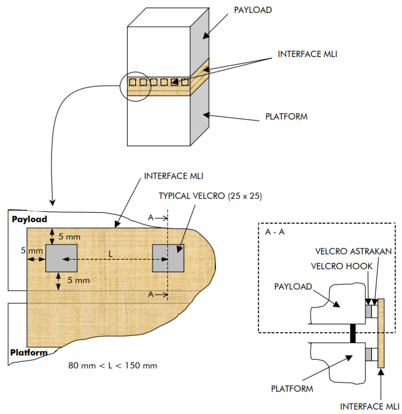

Fig. 3: MLI Interface between Platform and Payload

1.1 Multi Layer Insulation (MLI)

Multi Layer Insulation (MLI) consists of a very lightweight reflective film assembled in many thin layers, typically made of polyimide or polyester films. Typically, MLI can guarantee a reflection of up to 95% of radiation away from the spacecraft, depending on the number of layers. [4]

Multi Layer Insulation is one of the main components of PROTEUS's passive thermal control and can be used both internally and externally on various sections of the spacecraft. MLI is present in the following satellite elements:[9]

- In the battery module, which, due to its specific low and narrow temperature range, is decoupled from the rest of the platform through the presence of an internal MLI coating.
- Inside the propulsion zone, an internal layer of Multi Layer Insulation (MLI) is placed on the tank, on the upper side of the -Xs panel, and on all hydrazine-rich components located on the upper side of the propulsion panel, to limit radiative coupling with the platform.
- The entire main zone of the platform is covered with a layer of MLI, except for radiative surface areas, as well as all external areas of the satellite, excluding those requiring a line of sight with the Sun and Earth.

1.2 Heater Lines and Thermistors

Electric resistance heaters are composed of a polyimide film with etched foil circuits that generate heat when a current is applied. [6]

In the PROTEUS satellite, active thermal regulation utilizes electric heaters positioned on the platform panels and specific equipment. The performance of the regulation depends on the main characteristics of the heating lines, which may vary depending on the components or areas heated. The thermal stability of the lines can be summarized as: [9]

- Approximately 0.3°C on heating lines with low thermal inertia and sensors located near the heaters.
- Approximately 0.5°C on heating lines with high thermal inertia and sensors located near the heaters.
- Approximately 1.0°C on heating lines with high thermal inertia and sensors located far from the heaters.

On the PROTEUS Platform, 11 nominal and 11 redundant heating lines are used, each associated with three temperature acquisition sensors. For active thermal control, another additional 33 acquisition lines are used, along with thermal sensors compatible with these lines. [7]

In particular, the sensors utilized include the Fenwal 526-31 BS12-153, which is a thermistor that has a temperature measurement range from -60°C to +90°C, and the Rosemount 118 MF, a sensor with measurement has a range from -120 °C to +140°C. [7]

1.3 Silvered SSM Radiators

A radiator is a surface with low solar absorptivity and high infrared emissivity that is intended to dissipate excess heat through radiative heat transfer.

In Secondary Surface Mirrors Radiators, incident light enters the front surface, passes through a substrate of teflon or glass, and reflects off the coating of the back surface, also known as second surface, which in this case is made out of silver. The substrate serves primarily to provide a smooth, transparent surface to support and safeguard the reflective coating. Typically, a Secondary Surface Mirrors reflects 80% to 85% of incoming light, limiting the entry flux while maximising the emission.[9]

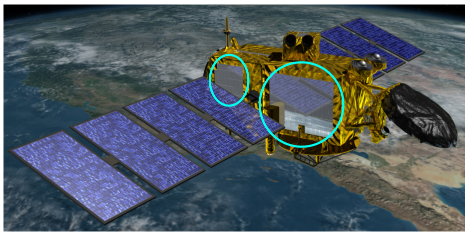

Fig. 4: Silvered SSM on Jason 2

The thermo-optic properties of the system are summarized in Table 1:

	ϵ **Infrared Emissivity**	α_{min} **Solar Absorptivity**	α_{max} **Solar Absorptivity**
SSM	0.76	0.10	0.16
MLI	0.77	0.32	0.49
Solar array (Solar cell face)	0.82	0.75	0.85
Solar array (back face)	0.7	0.92	0.92

Table 1: Thermo-optic characteristics of TCS components [7]

1.4 Thermal Straps and Insulating Washers

The insulating washers allow the overall thermal conductive coupling between the Star Tracker Array and the payload to be lower than 0.04 W/°C. These components also enable the isolation of -Z panel of the battery zone from the rest of the structure.

2 Design Choices

During the design of the TCS for the Jason-2 mission different possibilities were studied and thoroughly analyzed. Due to the volume constraints of the platform, the main objective of the TCS is to simply satisfy thermal requirements despite an equipment layout not necessarily optimized from a thermal point of view. In this paragraph the main motivations for the choices made in the design will be highlighted and further explored. [9]

2.1 Constraints

Even though the subsystem is not fully optimized thermally some critical constraints must be respected in order for the subsystem to function nominally. This constraints can be summarized in the following list:

- The battery must be positioned on the Anti-Earth panel of the spacecraft;
- The propulsion zone must be located near the launcher adaptor, which is strongly affected by solar incident fluxes in some satellite modes;
- The Data Handling Unit (DHU) must be located on the Earth-pointing panel of the spacecraft, which is the least efficient for heat rejection capability;
- There does not exist any possibility to conductively uncouple the launcher adaptor from the rest of the platform and to protect it from solar fluxes.[9]

2.2 Requirements and Objectives

The PROTEUS Thermal Control System has been designed obeying these following general guidelines: [9]

- The passive thermal control is sized to obtain the maximum authorized temperature on the equipments in the hottest case between all phases.
- The active thermal control is sized to withstand the coldest case between all phases.

This allows to minimize the heating power demand for the coldest case in all satellite modes and to effectively ensure the limit temperatures, considering margins.

When designing the TCS specifically for the Jason-2 some objectives were selected in order to limit the changes that could be done from the original general PROTEUS subsystem design. These objectives can be summarized in the following list: [9]

- No significant concept modifications must be done, such as radiators location on the spacecraft.
- No modifications in size and design of heater line components must be done in order to maintain compatibility with the PROTEUS platform.
- Only minor adaptations of thermal control components are allowed, such as a changes of the areas covered by MLI or the removal of distinct temperature sensors.

3 Environment and Temperature Range

Before proceeding with the reverse sizing, it is crucial to define the environmental conditions, understand the worst-case scenarios, both in the hot and cold cases, and determine the temperature ranges allowed by each system.

3.1 Environment

All spacecraft components have a range of allowed temperatures that must be maintained to meet survival and operational requirements during all mission phases. The spacecraft heat exchanges with the environment are shown in Figure 5.

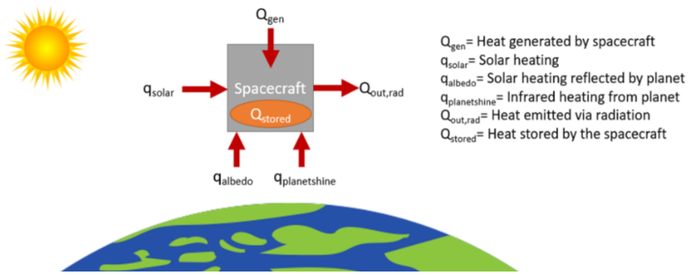

Fig. 5: Heat Exchanges between the Environment and the Spacecraft.

In order to dimension the system for any possible scenario, the two potential worst-case scenarios have been considered:

- **Hot Case**: The satellite is currently operating in nominal mode, with internal power consumption at its highest. Furthermore, it is in the sun's field of view, necessitating consideration for all three types of radiation.
- **Cold Case**: During an eclipse, the satellite is only exposed to infrared radiation since it is hidden by the Earth's shadow. Additionally, the spacecraft is assumed to operate in Safe Hold Mode, signifying minimal internal energy consumption and consequently reduced heat production.

3.2 Temperature Ranges

The platform's thermal control system aims to keep all the equipment inside their specific temperature ranges. Below are two tables outlining the temperature requirements for both the platform and the scientific payload Thermal Control System. As shown in the tables the components with the most pressing requirements belong to PROTEUS, so the critical hot temperature is assumed to be $30°C$ while the critical cold temperature is assumed as $0°C$. These temperature take into account the acceptable ranges of all the components excluding the batteries and the attitude sensors, since these components are thermally decoupled from the rest of the system.

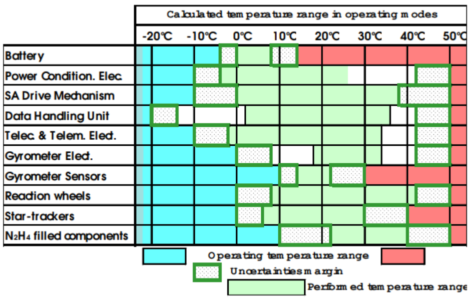

Fig. 6: Proteus Platform's Temperature Ranges [9]

Instrument	**Lower [°C]**	**Upper [°C]**
T2L2 [3]	-40	40
POSEIDON [2]	-5	35
DORIS [8]	-5	5
AMR [10]	-270.5	477

Fig. 7: Payload Instruments' Temperature Ranges

4 Reverse Sizing

Once the range of operative temperatures has been defined the reverse sizing is completed in order to size the thermal controls components, which are the heaters and the radiators. A first approximation of the sizing has been done considering a single-node model in steady-state conditions.

4.1 Thermal heat sources

As shown earlier, the main external sources of heat affecting the satellite are the solar, albedo, and infrared fluxes. The solar flux can be modelled as:

$$q_{solar}(r_{sc}) = q_0 \left(\frac{r_{Earth}}{r_{sc}}\right)^2 \left[\frac{W}{m^2}\right] \tag{1}$$

where $q_0 = 1367.5\ W/m^2$ is the solar flux at 1 AU and r_{sc} is the distance of the spacecraft from the Sun. The albedo flux has been modelled as:

$$q_{albedo} = q_{solar} \cdot a \cdot cos(\theta) \left(\frac{R_{Earth}}{R_{orbit}}\right)^2 \left[\frac{W}{m^2}\right] \tag{2}$$

where $a = 0.3$ is the albedo factor of Earth [7]; θ is the irradiance angle between the normal of the panel considered and the Earth and has been assumed 0 to account for the worst-case scenario. The infrared flux is modelled as:

$$q_{IR} = \sigma \cdot \varepsilon_{Earth} \cdot T_{Earth}^4 \left(\frac{R_{Earth}}{R_{orbit}}\right)^2 \left[\frac{W}{m^2}\right] \tag{3}$$

where ε_{Earth} is the emissivity of the planet and T_{Earth} is the temperature of Earth, which radiates like a black body at an equivalent temperature of $255K$ [7].

4.2 Spacecraft Modelling

The spacecraft has been modelled as a single-node. The emissivity of each panel has been assumed equal to the -Z surface, as it is the only one in which the radiators are not included in the value. The main body areas and optical properties of each panel are summarized in Table 2.

Surface	Area[m^2]	Emissivity	Absorptivity
-X	0.783	0.017	0.013
+X	0.783	0.017	0
-Y	2.040	0.017	0.043
+Y	2.040	0.017	0.037
-Z	3.105	0.017	0.002
+Z	3.105	0.017	0.334

Table 2: Areas and thermo-optic values of the main body[1]

The radiative power of the sources can be calculated as:

$$\begin{cases} Q_{solar} = A\alpha_{sc}\, q_{solar} \\ Q_{albedo} = A\alpha_{sc}\, q_{albedo} \qquad [W] \\ Q_{IR} = A\varepsilon_{sc}\, q_{IR} \end{cases} \tag{4}$$

where α_{sc} and ε_{sc} are the absorptivity and the emissivity coefficients related to the surface.
The model used considers surface +Z pointed towards Earth and for this reason, it will be the surface used to compute the heat power due to albedo and infrared fluxes, while surface -X is used to calculate the solar flux as it is considered the worst-case scenario. Surfaces +Y and -Y are perpendicular to the solar panels and for this reason, these surfaces will never point towards the Sun. Conversely, the heat emitted by the spacecraft through radiation can be divided into:

$$\begin{cases} Q_{em} = \sigma\varepsilon A_{sc}(T_{sc}^4 - T_{space}^4)\,[W] \\ Q_{rad} = \sigma\varepsilon A_{rad}(T_{sc}^4 - T_{space}^4)\,[W] \end{cases} \tag{5}$$

where Q_{em} is the heat emitted by all the surfaces, the total amount is given by their sum while Q_{rad} is the heat emitted by the radiators. Since the radiators are directly positioned on the spacecraft surface(mainly on ± Y axis[5]) the total surface will be reduced accordingly iteratively, $A_{sc,new} = A_{sc,tot} - A_{rad}$.

4.2.1 Satellite Hot Case

In the hot case all possible heat sources are considered and the satellite is in full operation so the internal power consumption is at its maximum and it is the sum of the heat dissipated by the PROTEUS and the payload[7] .

$$Q_{solar} + Q_{albedo} + Q_{IR} + Q_{int,max} = Q_{em} + Q_{rad,hot} \tag{6}$$

The hot case occurs when the satellite is in orbit, in nominal mode, during the solar phase. The total thermal power exchanged is resumed in the following table, divided in its components.

Incoming				Outgoing
$Q_{solar}[W]$	$Q_{albedo}[W]$	$Q_{IR}[W]$	$Q_{int,max}[W]$	$Q_{em}[W]$
13.92	290.85	12.695	550	96.52

Table 3: Heat fluxes in the hot case

From Equation 6 the heat flux dissipated by the radiators can be found, $Q_{rad,hot} = 770.95W$ which correspond to a total surface for the radiators of 2.12m^2. This value is satisfactory since it is quite close to the real one which is 2.08m^2[5]. Given a density area of 8 kg/m^2, the total mass of the radiators would be 16.96 kg.

4.2.2 Satellite Cold Case

In the cold case only the infrared flux is present since the satellite is in eclipse phase and it is in SHM so the dissipated internal power by the platform is lower[7] and the payload is not active.

$$Q_{IR} + Q_{int,min} + Q_{heaters} = Q_{em} + Q_{rad,cold} \tag{7}$$

In this equation the only unknown is the heat that must be produced by the heaters in order not to go below the minimum acceptable temperature. The heat components are resumed in Table 4

Incoming		Outgoing	
$Q_{IR}[W]$	$Q_{int,min}[W]$	$Q_{em}[W]$	$Q_{rad,cold}[W]$
12.695	180	63.62	508.16

Table 4: Heat fluxes in the cold case

The heat required by the heaters through this equation should be $379.08W$, which is quite large. This is probably due to how the problem was modelled, since the spacecraft is considered as single-node and it does not take into consideration that different parts of the satellite require different amount of heat: only the critical components need to be kept hot while most of the parts can go below $0°C$.

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