

## THERMAL DESIGN OF SACI-1 SATELLITE

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***Abstract.** This paper describes the development of the thermal design of SACI-1 satellite, launched in October 1999 by the Chinese rocket Long March 4. To satisfy the temperature requirements, a passive thermal control was designed and implemented on this satellite. A numerical model, based on the lumped parameter method, was implemented and used as a tool during the design process. Several tests were performed on the thermal mockup and the flight model, in the thermal-vacuum chamber, to verify the thermal design performance and check the mathematical model.*

***Keywords:** Satellite thermal control, Saci-1*

### 1. INTRODUCTION

In the space environment, the two main temperature references are the sun, at 5700 K, representing the heat source, and the deep space, at 4 K, representing the heat sink. Theoretically, a body in the space may have any temperature between these two references, depending on the interaction between the body and the space thermal environment.

On the other hand, any component of a spacecraft has its own range of temperature within it can operate. Out of this range, malfunction or permanent damage can occur. A typical range for electronic equipment is from -10 to 50 °C. However some equipment, as battery, are more temperature sensitive and the range is narrower, about -5 to 25 °C. Others, as solar cells, stand a wider range of temperature, typically from -70 to 70 °C.

The objective of the thermal design is to keep all the components of the spacecraft within their temperature limits, during the spacecraft life time. The basic thermal design consists in managing the heat absorbed from the solar radiation and the heat rejected to the space, which balance determines the equilibrium temperature of the spacecraft. Several types of selective coatings can be used to control the absorption and rejection of the heat radiation, by means of their optical properties, namely, solar absorptivity ( $\alpha_s$ ) and infrared emissivity ( $\epsilon_{IR}$ ). A list of the common coatings for space application and their properties is presented by Gilmore (1994).

In a real satellite thermal design, besides the direct solar radiation, other sources of heat impinging to the satellite are considered, i.e., albedo (solar radiation reflected by earth),

IR radiation emitted by the earth and heat generation inside the electronic equipments. The heat input from these sources depends on (i) the satellite orientation related to the sun and earth, (ii) the orbital position and (iii) the satellite operation mode.

The satellite temperature distribution is not only function of the interaction between the satellite and the environment, but also depends on the heat exchange among its components, that can occur by conduction and by radiation. Basically the satellite average temperature is set by the external surface coatings. The internal temperature gradient is adjusted by the inner surface coating and by the conductive path inside the satellite. The components can be thermally insulated or coupled with others, depending on the design requirements.

Generally it is possible to accomplish the thermal control of low orbit, low power and spin stabilized satellite, by using only passive means, that is, by using selective surface coating and by controlling heat conduction path with thermal insulator or thermal bonding. Passive thermal control is preferred due to its high reliability and low weight if compared with the active control.

This paper presents the thermal design of SACI-1 satellite, fully developed by the Thermal Control Group of INPE (Instituto Nacional de Pesquisas Espaciais), from the design conception to implementation on the flight model. The design is based only on passive means following other satellites with similar characteristics as reported by Murthy et al. (1978) and Muraoka and Leite (1994).

## **2. SACI-1 SATELLITE DESCRIPTION**

SACI-1 (Satélite de Aplicações Científicas) is the first Brazilian scientific satellite, designed, assembled and tested at INPE and launched by Chinese rocket Long March 4, as piggyback, in October 1999. Unfortunately the satellite couldn't communicate with the ground stations after launching, and the mission was aborted. Four scientific experiments, selected by Brazilian Academy of Science, flew on board the satellite to study specific phenomena in the near-Earth space environment (Neri & Fonseca, 1998).

SACI-1 main body has a shape of a parallelepiped, with dimensions of 0.38 x 0.38 x 0.46 m, as shown in Fig. 1. It is composed by modular aluminum ribbed frames, stacked horizontally and fastened to each other by means of stud bolts. Fig. 2 presents the distribution of the modules in the satellite. Three extendable panels with solar cells are used to capture energy from the sun, in a maximum rate of 150 W. A fourth extendable panel is used for dynamical purpose only. The total mass is about 60 kg.

The orbit is circular, with an altitude of 778 km and an inclination of 98.5 degrees related to the equatorial plane. The orbit period is about 100 minutes, and the eclipse (period when the satellite is in the shadow of the earth) can last between 32 and 34 minutes. The spin stabilization, with about 6 rpm, keeps the solar panels always pointing to the sun, during the nominal phase.

Just after the launch the satellite should be without any rotation (or with small residual spin), and with aleatory orientation related to the sun. During the attitude acquisition phase the rotation should increase in a approximate rate of 1 rpm per hour and, at the same time, the satellite should be pointed toward the sun, in a way that, in three hours the sun aspect angle (angle between the spin axis and the sun rays) is less than 20 degrees.

## **3. THERMAL DESIGN SPECIFICATION**

The thermal design was specified to keep all components of the satellite in their respective operation temperature limits, considering all the thermal conditions expected for the satellite

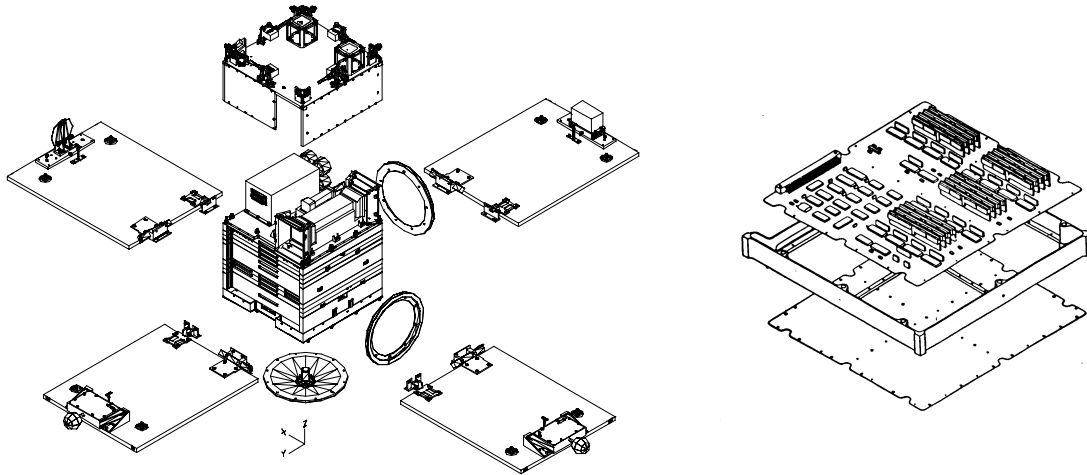


Figure 1 - Exploded view of SACI-1 and the standard modular frame

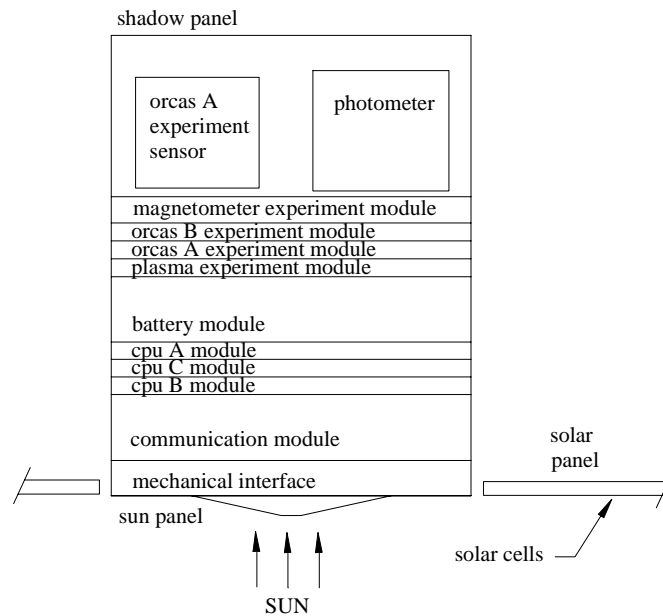


Figure 2 - Distribution of the modules in the satellite

during the whole mission. Table 1 presents the operational temperature limits and the internal heat dissipation of each equipment of the satellite.

The most thermal sensitive equipment is the battery, whose temperature limits is between  $-5$  and  $25$  °C. The most dissipative equipment is the communication module that dissipates  $7.5$  W, in the continuum regime, and  $24.8$  W during a period of 12 minutes, when the satellite is transmitting data to the ground station.

During the nominal operational phase, when all experiments are turned on, the maximum total internal heat dissipation is  $54.9$  W, considering an average value for a period of one orbit. During the attitude acquisition phase all experiments will be turned off, and the total internal heat dissipation is reduced to  $32.2$  W.

Table 1 - Equipment temperature limits and their thermal dissipations

Equipment	Operation temper. limits (°C)	Heat dissipation (W)
Communication module	-10 ~ 50	7.5 + 17.3 <sup>(1)</sup>
CPU B module	-10 ~ 50	5.4
CPU C module	-10 ~ 50	3.9
CPU A module	-10 ~ 50	3.9
Battery module	-5 ~ 25	6.0 + 5.0 <sup>(2)</sup> + 2.0 <sup>(3)</sup>
Plasma experiment module	-10 ~ 50	6.3
Orcas A experiment module	-10 ~ 50	1.5
Orcas A sensor	-10 ~ 50	1.5
Orcas B experiment module B	-10 ~ 50	1.5
Magnet. experiment module	-10 ~ 50	6.3
Photometer experiment sensor	-10 ~ 50	3.6
Solar sensor	-30 ~ 70	0
Antenna	-20 ~ 60	0
Solar cells	-85 ~ 75	0

<sup>(1)</sup> Additional temporary dissipation during the visibility period (~ 12 minutes);

<sup>(2)</sup> Additional temporary dissipation during the illuminated period (~ 68 minutes);

<sup>(3)</sup> Additional dissipation at the end of life (EOL), due to degradation of the battery cells.

#### 4. THERMAL DESIGN

Some characteristics of SACI-1, as low power, spin stabilization, and low orbit, are very favorable to the thermal control point of view. With low heat dissipation in the electronic equipments, the conduction path, through the structure, is enough to transfer the heat from the equipment to the satellite external surface, from which it is irradiated to the space. The spinning provides an homogeneous incidence of the external radiation on the satellite, that reduces the internal thermal gradient. Low orbit means short orbit period that, in turn, means short exposition of the satellite to the cold (eclipse) and to the hot environment (sun light), that reduces the variation of temperature during the orbit.

On the other hand, the considerable difference of conditions between operational and acquisition phase makes the thermal design more difficult. As, during the acquisition phase, the orientation of the satellite related to the sun is unknown, the thermal control should be designed to satisfy the thermal requirements at any solar incident angle. Also, the thermal design should accommodate the large variation of the internal heat dissipation between the acquisition phase (32.2 W) and the operational phase (52.9W), that is a source of the temperature variation along the satellite life time.

The first step in designing of the SACI-1 thermal control was to identify the critical cases, in terms of the heat load comprising the internal heat dissipation and the external radiation. Table 2 summarizes the conditions of the coldest and hottest orbit, during the operational phase. Shown cases do not comprise the variation during a orbit (eclipse/sun light) but variations of long period as change in the solar constant and the orientation of the orbit plane related to sun and earth. Also, the thermal design had to consider the conditions during the acquisition phase, when the orientation the satellite related to the sun is unknown and all experiments are turned off (average internal dissipation is 32.2 W).

Table 2 - Conditions of the critical cases during the nominal phase

Item \ Case	Hottest case	Coldest case
Solar aspect angle	0 degree	0 degree
Solar constant	1420 W/m <sup>2</sup>	1326 W/m <sup>2</sup>
Eclipse period	32 minutes	34 minutes
Avg. total internal dissipation	54.9 W <sup>(1)</sup>	50.8 W <sup>(2)</sup>

<sup>(1)</sup> orbit with 12 minutes of visibility and EOL (end of life) battery cells dissipation

<sup>(2)</sup> orbit without visibility and BOL (begin of life) battery cells dissipation

After determination of the critical cases, the next step was to find a combination of external surface coatings that satisfies the temperature limits of the components. This was done interactively, testing case by case and adjusting accordingly the coating, until a combination, that satisfies all the limits for all the critical cases, is reached.

Besides of setting the components temperature in the allowed range, the choice of the external surface coating took into account the optimization of the relation between external absorbed heat flux and internal dissipation. The balance between these two types of heat load is important to avoid high sensitivity of the temperature related to the fluctuation of the internal heat dissipation or to the variation of the external heat flux. For example, if the satellite is insulated from the external environment, a small fluctuation of the equipment heat dissipation would cause a large temperature variation in the satellite. On the other hand, if the satellite is totally painted with black paint, the temperature fluctuation between illuminated and eclipse period would be large.

Another concern in the choice of external coatings was to minimize the impact of satellite orientation, related to the sun, on the satellite temperature. The objective was to avoid undesired thermal behavior during the acquisition phase, when the satellite may have different attitude from the nominal one. The final external surface coating was set as shown in Fig. 3.

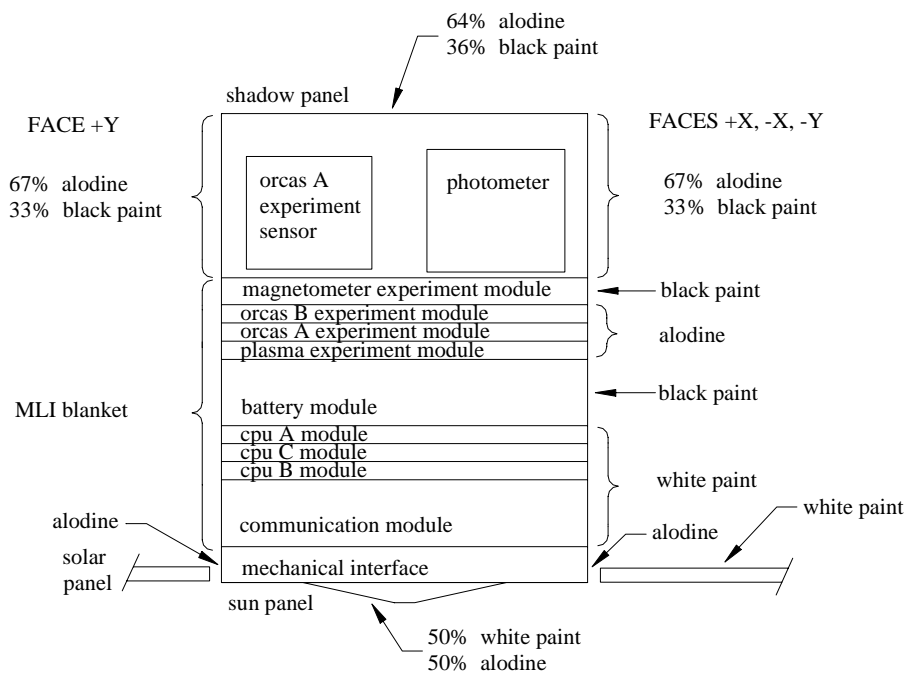


Figure 3 - SACI-1 external surface coatings

The face +Y of the satellite body, where the electrical connectors are located, was covered by a Multi-layer Insulation (MLI) blanket. This blanket protects the connectors and cables from large temperature fluctuation that they could have if they were directly exposed to the space environment, since they acts as small fins.

The shadowed face of the solar panel was painted with white paint to minimize its temperature level, as the efficiency of the solar cells is higher at low temperatures.

Special thermal design was required for the equipments outside of the satellite main body. Some sensors with small heat dissipation, fixed on shadowed face of the solar panel, are insulated from the space environment using MLI blanket, and from the solar panel structure using epoxy-fiberglass composite washer at the fixation points, to avoid large fluctuation of their temperature during the orbit, passing through the illuminated and eclipse period.

Concerning to the satellite internal temperature gradients, the adopted strategy was to minimize them, and to try a design that satisfies the narrower range of temperature limitation, that is from -5 to 25 °C for the battery module. This was done considering that SACI-1 is a compact satellite and it is difficult to separate it in different thermal zones to satisfy, individually, each equipment thermal limitation. Following this orientation, the shields (thin aluminum plates put between modules) were painted with black paint. The properties of the thermal control components, used in the design, are presented in Table 3.

Table 3 - Thermal control materials and its properties

Material	solar absorptivity	IR emissivity	therm. conductivity
Black paint	0.96	0.88	-
White paint	0.14 ~ 0.24	0.80	-
Alodine	0.24	0.06	-
Solar cell <sup>(1)</sup>	0.68 ~ 0.80	0.83	-
MLI Blanket			0.06 W/m <sup>2</sup> /K <sup>(2)</sup>
Fiber glass washer			0.5 W/m/K

<sup>(1)</sup> not thermal control material, <sup>(2)</sup> thermal conductance per area

## 5. MATHEMATICAL MODEL

The thermal analysis was accomplished by using PCTER thermal analysis software (Bastos, 1990), developed at INPE by the thermal control group. PCTER is a set of programs that cover all the steps of the satellite thermal analysis, from geometry processing to temperature presentation.

The model is based on the nodal or lumped parameter method, in which the satellite is divided in a finite number of isothermal regions, called nodes. These nodes are connected to each other by conductive and radiative couplings. They can receive thermal load from external radiation or internal heat dissipation. The thermal balance at node  $i$  is given by:

$$m_i c_{pi} \frac{dT_i}{dt} = \sum_{j=1}^N C_{ji} (T_j - T_i) + \sum_{j=1}^N R_{ji} (T_j^4 - T_i^4) + Q_i, \quad i = 1 \dots N \quad (1)$$

The term  $m_i c_{pi}$  represents the thermal capacitance of the node  $i$  and  $T_i$  and  $T_j$  are the absolute temperature of the nodes  $i$  and  $j$  respectively.

The conductive couplings  $C_{ji}$  are determined considering unidirectional fluxes. The radiative couplings  $R_{ji}$  are calculated by Gebhart method (Gebhart, 1961), supposing all

surfaces are diffuse, gray and isothermal. When the node  $i$  is representing an electronic equipment, the thermal load  $Q_i$  is the heat generated inside the equipment when it is turned on. When the node  $i$  is representing an external panel,  $Q_i$  is the thermal radiation absorbed by this panel from external source. The temperature is calculated at each node  $i$  by solving the differential equation (1) using the Crank-Nicholson method.

The satellite was represented by 70 isothermal nodes. Each module was idealized as an open box, and each lateral side and the bottom side are represented by a node, as shown in the Fig. 4. The node 1 represents the printed circuit board, including the mass of the electronic components. It is assumed that the heat generated in these components is distributed homogeneously on this node. The nodes 2 to 5 are subject to the external radiation from the sun and earth, that hits the lateral faces of the satellite.

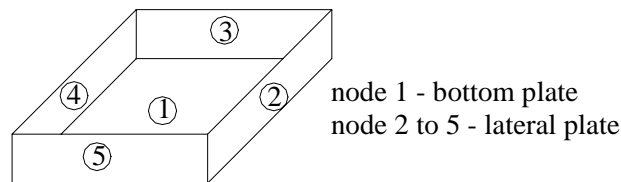


Figure 4 - Nodal breakdown of each module

The external heat loads incident on the satellite are the direct solar radiation, albedo and earth radiation. The intensity of each type of radiation is calculated for each external surface of the satellite at discrete positions in the orbit. The intensity is proportional to the shape factor between the surface and the heat source, that is calculated by contour integral. The calculations consider the solar constant from 1326 to 1420 W/m<sup>2</sup>, the average albedo factor of 0.35 and the average earth temperature of 250 K.

## 6. PREDICTED TEMPERATURES

The developed model was used to determine the temperature distribution for coldest and hottest cases of the operational phase, described in Table 2, and for some representative conditions of the acquisition phase. As in this phase the satellite can have any orientation related to the sun, the analysis was done for five values of the solar aspect angle (0, 45, 90, 135, 180 degrees), and considering that all experiments are turned off (internal heat dissipation of 32.2 W).

The temperatures predicted for the seven cited cases, considering the final design configuration, are presented in Figs. 5 to 9. Figures 7 and 9 merge two cases in one diagram. The term “radiator” in the diagram means the border of the module that is represented by the nodes 2, 3, 4, and 5 in the sketch of the Fig. 4.

For operational phase all the equipments have a margin larger than 5 °C between predicted temperatures and equipment operational limits. The battery temperature is kept near the hot limit in the nominal phase to avoid temperature below the lower limit during the acquisition phase.

For acquisition phase, the main concern was the low limit. The reason is the low internal heat dissipation as all experiments are turned off. Among the analyzed cases the most critical is when the solar aspect angle is between 0 and 45 degrees (Fig. 7). As in these cases the solar energy is already available (sun hits the solar cells), the experiments can be turned on to avoid low temperatures in sensor bay radiator and shadow panel (the prediction consider that the experiments are turned off). For solar aspect angles from 90 to 180 degrees, when

solar energy is not available, the temperatures are in comfortable range, as shown in Figs. 8 and 9.

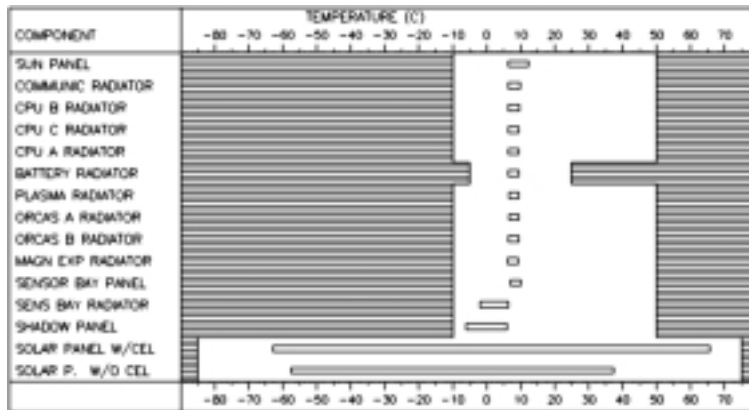


Figure 5 - Predicted temperatures for the cold case of the operational phase

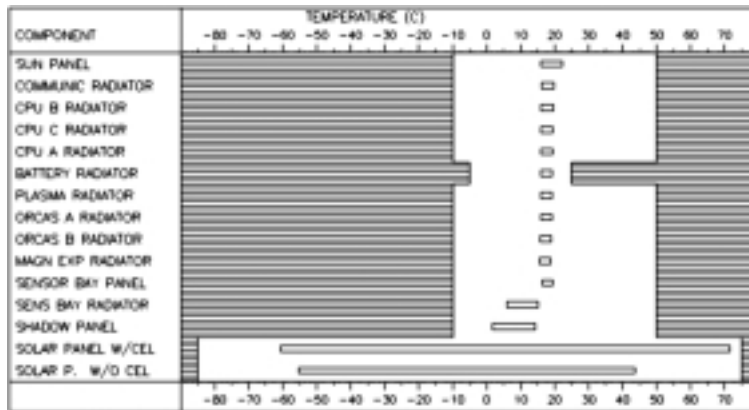


Figure 6 - Predicted temperatures for the hot case of the operational phase

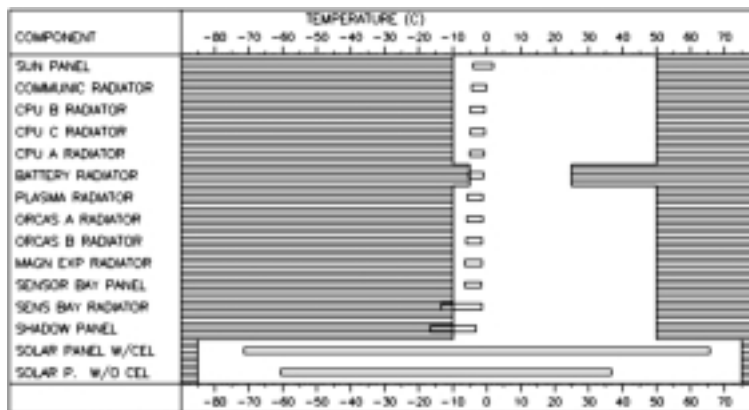


Figure 7 - Predicted temperatures for solar aspect angle of 0° and 45°, for acquisition phase



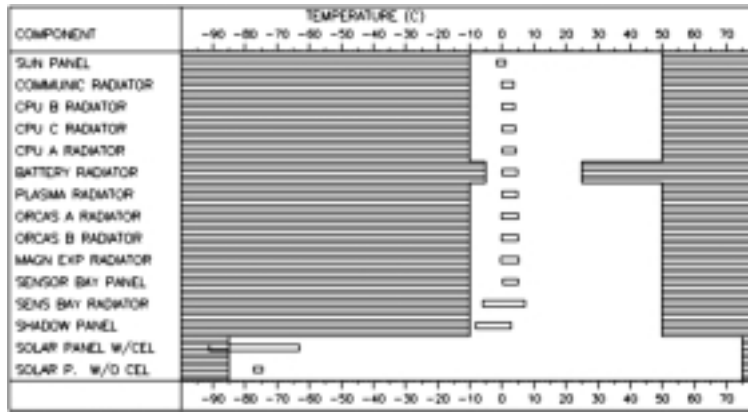


Figure 8 - Predicted temperature for solar aspect angle of 90°, for acquisition phase

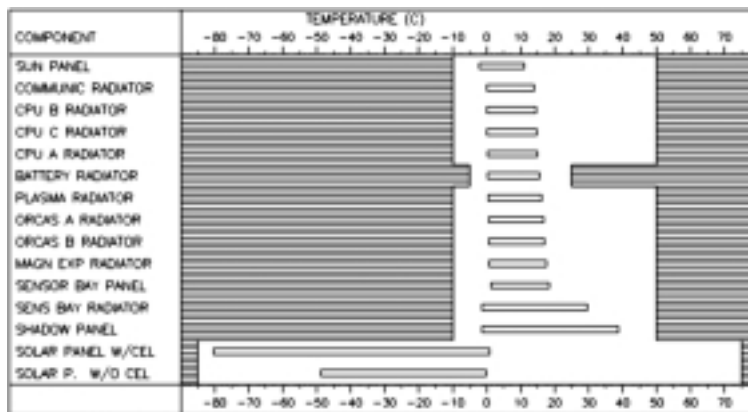


Figure 9 - Predicted temperature for solar aspect angle of 135° and 180°, for acquisition phase

## 7. TESTS

The performance of the thermal design was verified by submitting a mockup to a Thermal Balance Test (TBT) in the space simulation chamber at LIT (Integration and Test Laboratory of INPE). A mockup was constructed using the satellite real structural parts but without the electronic components. The real PCB's (Printed Circuit Board) were replaced by bare PCBs with some small metal blocks fixed on them to simulate the thermal masses. Also skin-heater was installed on the bare PCB to dissipate the equivalent heat of the electronic components during the test. The box type electronic equipment was replaced by aluminum box of equivalent mass with skin-heater.

The mockup was placed at the center of the space simulation chamber. The chamber was evacuated at a pressure around  $10^{-6}$  torr and its internal walls, painted black, were cooled at  $-180^{\circ}\text{C}$ , with  $\text{LN}_2$ , to simulate the deep space. The external heat loads were simulated using skin-heaters, fixed on the satellite external surfaces.

Two steady state conditions and one transient conditions were simulated during the TBT, representing the coldest and hottest cases of the operational phase. In steady state condition, the orbit average heat load was applied on the mockup. In the transient condition, the variation of the heat load along the orbit was simulated, considering the eclipse and illuminated period, and also the status of each equipment (on or off).

The thermal behavior during the test was as expected, indicating the suitability of the adopted design. The mathematical model presented good correlation with the experimental results, as shown in Table 3 for steady-state condition, with maximum discrepancy of  $3.7^{\circ}\text{C}$ .

The flight model was also subjected to a thermal test named Thermal Cycling Test (TCT). Although the main objective of the TCT was to check the satellite operation, the measured temperatures served to check if the thermal design was correctly implemented and to evaluate the mathematical model.

Table 3 - Comparison between predicted and measured temperatures (°C).

Module	Hot case			Cold case		
	measur.	predict.	differ.	measur.	predict.	differ.
Sun panel	23.6	23.0	0.6	13.5	12.4	1.1
Communic. radiator	23.1	22.2	0.9	12.0	11.4	0.6
CPU B radiator	22.9	22.0	0.9	12.4	11.3	1.1
CPU A radiator	22.0	21.8	0.2	11.6	11.3	0.3
Battery radiator	21.3	21.7	-0.4	11.2	11.1	0.1
Plasma exp. radiator	21.3	21.3	0	11.5	10.9	0.6
Orcas A exp. radiator	19.7	20.9	-1.2	11.0	10.7	0.3
Orcas B exp. radiator	20.7	20.8	-0.1	11.3	10.6	0.7
Magnet. exp. radiator	20.1	20.3	-0.2	10.9	10.3	0.6
Sensor bay panel	16.9	20.6	-3.7	8.0	10.6	-2.6
Sensor bay radiator	8.6	9.0	-0.4	0.7	0.6	0.1
Shadow panel	2.8	4.3	-1.5	-4.5	-3.5	-1.0

## 8. CONCLUSION

A passive thermal control system was adopted for SACI-1 satellite. The main design feature was the selection of the external surface coating to manage the heat exchange between the satellite and the environment and to get an appropriate temperature distribution for all components of the satellite. A mathematical model, based on the lumped parameter method, was developed to predict the temperatures as a function of the satellite surface coatings, the incident external radiation profile and the internal heat generation. A mockup of the satellite was submitted to a test in the space simulation chamber and the results showed a satisfactory performance of the thermal design. The test results was also used to qualify the mathematical model considering its good correlation with the experimental data. Unfortunately the thermal design couldn't be checked with flight data because the mission was aborted after launching, due to technical problem with the satellite.

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