Passive Thermal Control in Small Satellites – Up-to-Date Tendencies and Advanced Components

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ABSTRACT

The main idea of this lecture is to present the survey of current tendencies in micro-satellites thermal control concepts that can be rational and useful for posterior missions due to intensive dissemination of satellites of such type. For this purpose the available references and lessons learned by the National Technical University of Ukraine in the course of the elaboration of thermal control hardware for micro-satellites Magion 4, 5, BIRD and autonomous thermal control systems for interplanetary missions VEGA and PHOBOS have been used. The main parameters taken into consideration for analysis are the followings: satellite sizes, mass, power consumption, orbit parameters, altitude control peculiarities and thermal control description. It was defined that passive thermal control concepts are widely used, excepting autonomous temperature regulation for sensitive components such as batteries, high precision optics and some types of sensors. The practical means for realisation of passive thermal control design as multilayer insulation, optical coatings, heat conductive elements and gaskets are briefly described.

INTRODUCTION

Last years conferences on small satellite activities [5, 6], the information about currently elaborated projects and future small satellite missions planned [7] emphasise the actuality of satellites and constant interest in them. The intensive dissemination of small satellites could be explained by their moderate enough cost, short time of elaboration and existing possibility to place the complicated devices like multifunctional equipment and optical systems as payload. According to small satellite classification [7] there are the following conditional groups: nano- and pico- satellites (<10 kg), micro-satellites (10-100 kg), mini-satellites (100 –500 kg), inter-planetary small missions (>500 kg). Table 1 (and Foil_2) presents some technical data for considered small satellite groups.

The certain constrains originating in small satellite designs due to limited mass and power, certain available volume for payload and housekeeping systems produce the set of requirements to each of satellite systems and to the thermal control system as well. The survey of thermal means used in practice in small satellite designs, visible perspectives and difficulties can be useful for better understanding this subject. The main attention at consideration is devoted to small satellite group with mass less and near 100 kg (micro-satellites), where searching of compromise in the power/mass/volume distribution between mandatory housekeeping needs and the payload requirements is actual.

Table 1. Attributes of small satellite groups

Satellite class	Mass	Bus linear sizes	Power averaged
mini	100 ÷ 500 kg	More 1 m	Up to 100 Watts
micro	10 ÷ 100 kg	0.5 ÷ 1 m	Tens Watts
nano	Less 10 kg	Less 0.2 m	Several Watts

The lecture does not contain the analysis of all thermal control principles used and their technical embodiment, as according to [7] (Figure 1, Foil_2) more, than 250 micro-satellites launches have been realised during 1980 – 2002, and each of them has something specific. Additional difficulty is insufficiency of available descriptions of thermal control details.

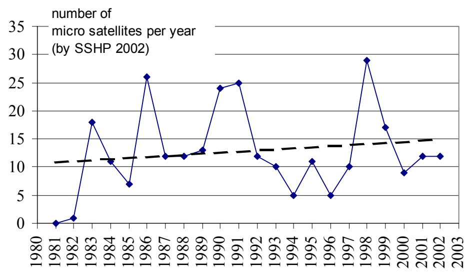

Figure 1. Launches of micro-satellites in years [7]. Dash line is a trend linear approximation of operating micro-satellites

TYPICAL MICRO-SATELLITE THERMAL CONCEPTS

The main principles of space thermal control design, software and hardware used are collected in [8-14, 20] and many other works. Nevertheless, namely the details of thermal control system (TCS) design are mostly important in the case when they are embodied in practically realised projects.

The diverse information about thermal designs is collected by Surrey Satellite Technology Ltd, [6, p.417], Technical University of Berlin [6, p.347], by Design Bureau “Pivdenne” [5, p.437] and other leading small satellite space hardware manufacturers as well. Summarising, it is possible to emphasise the following regularities.

External Heat Exchange

Micro-satellites operate on low Earth circular (450 –1200 km) with wide range of NASA β angles and on high elliptical orbits. This means that satellite sides can be exposed to external disturbances as IR Earth radiation, Albedo, Sun. IR and Albedo can be negligible for high elliptic orbits. The typical maximal incident fluxes for 550 km orbit for flat surface with normal to nadir are q_{IR} ~200 W/m^2, $q_{ALB,\,MAX}$ ~450 W/m^2 (orbit-averaged <150 W/m^2). The eclipse time can reach from 0.5 hour (circular) to several hours for elliptic orbits. Most often used Attitude Control Systems (ACS) are spin attitude control, gravity and 3-axes stabilisation. This means that satellite external light disturbances can be predicted due to exact knowledge of its attitude. Spin attitude foresees the arrangement of rotation axes perpendicular or parallel to Sun direction; ACS for 3-axes stabilised satellites has to be designed to change often enough the satellite orientation along orbit movement to solve the intended tasks (Foil_3).

Thermal Concepts

There are three conditional approaches in thermal scheme design – autonomous concept, centralised concept and combined concept (Figure 2, Foil_4). Autonomous concept foresees the individual thermal control for each device or group of devices. The devices/equipment thermally are disconnected each from other. The centralised concept guesses the tight thermal coupling of all units and the usage of one centralised radiator for external heat exchange. The mostly exploited concept is combined concept where some group of equipment (for example, housekeeping equipment, satellite bus) has common thermal control, several devices (some of payload) are thermally disconnected from satellite bus, and uses autonomous thermal control. The thermal concept defines the way length of heat roaming between device and aimed radiator, and also effective thermal mass of satellite and units. Each of concepts has own peculiarities, depending on satellite orientation and attitude control. For spinning satellite with perpendicular position of rotation and Sun light axes (Figure 2, Foil_4) the external sides can be used as thermal sink as at ratio α_s/ε< 1 the wall temperature will trend to the level of 10 ÷ 20 ^{0}C. The solar cell has similar optical characteristics, and it can be used as thermal sink. For coinciding longitudinal and Sun light axes (Figure 2, spinning and 3-axes stabilised satellites, Foil_4) lateral satellite surfaces can be used for heat rejection. The heat removal can be arranged through one centralised radiator (all thermal lines go to it) or several radiators distributed. The tight thermal connection of all components enlarges the thermal mass of satellite, reduces the temperature non-uniformity throughout the satellite. Assuming that at power of 40 W the area of radiator is only 0.15 ÷0.2 m^2, and the rest of lateral surface should be coated with MLI with area of ~ 0.5÷1 m^2. The satellite radiator temperature is sensitive to steadiness of power rejected, and roughly 1 W of energy variation courses 1÷1.5 K temperature changing.

The cover of front satellite surface with optical selective

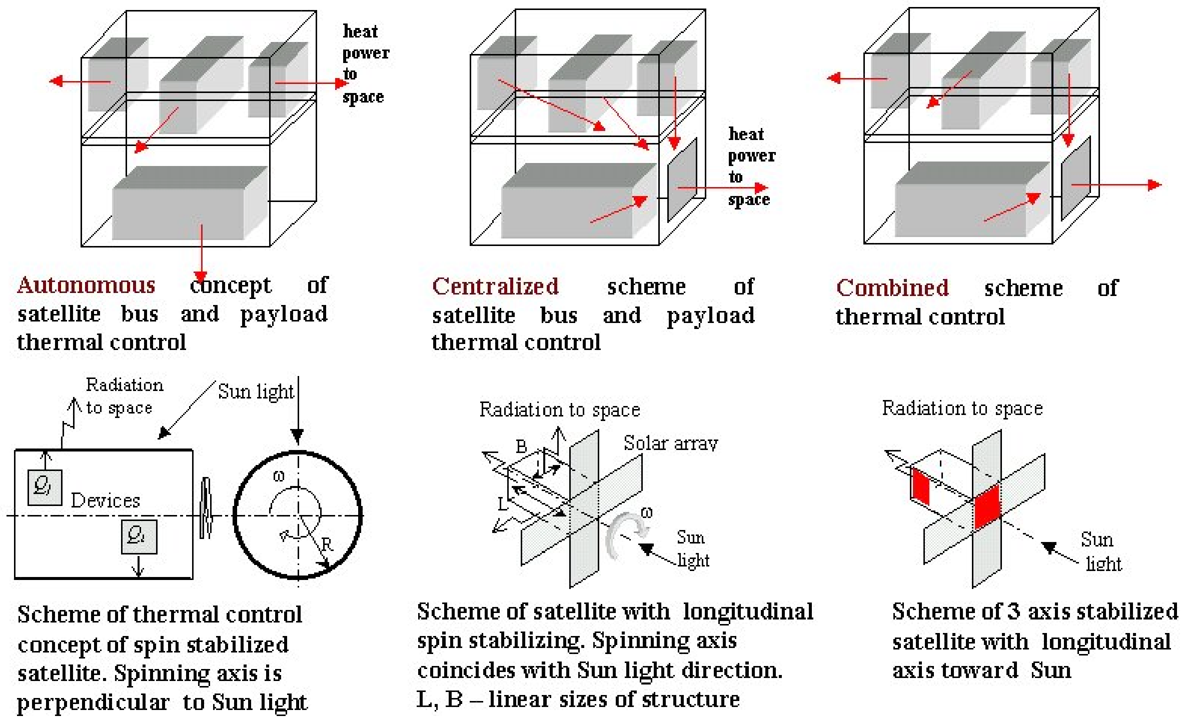

Figure 2. Conditional thermal concepts of micro satellites

coating (OSC) with α_s~0.2, ε ~ 0.85 allows to reach the radiator temperature level of 0 ÷ 30 ^{0}C. Predominant design foresees non-hermetic shell of satellites that requires conductive and radiative means for a heat transfer.

Inner Thermal Tasks

Typical requirements to satellite inner components are collected in the Table 2 [12] (and Foil_5).

Table 2. Typical thermal requirements

Component name	Temperature ^{0}C	Peak power, W
Electrical equipment	-10 to +40	Max 10 per unit
Batteries	-5 to +15	Up to 20
Consumable gas	+9 to +50	-
Microprocessors	-5 to +40	Up to 20
Bearing mechanisms	-5 to +40	-
Solar cells	-60 to +55	-
Solid-state diodes	-60 to +90	-
Orientation sensors	-5 to +45	Max 5
Optics	21	Max 10
Payload	Individually	< 150
Satellite total		250, average 30÷60

The most delicate devices are batteries, optic instrumentation and individual payload. Requirements to average heat generation inside the satellite are in the range of 15÷40 W. This power is produced mainly by housekeeping equipment, heat peak generation coincides with payload operation and can reach 200 W; typical housekeeping heat generating components are board computer, transmitter, ACS and batteries. If payload operates constantly, the power generation does not change abruptly.

The most temperature sensitive units are battery sets, which should be exploited over the range of -5÷ +15 ^{0}C. More details about satellite component operating temperatures are presented in [8-14]. Overwhelming majority satellites are built as non-sealing objects, and heat transfer inside is realised by conduction of structure elements and radiation (black surface coating ε ~ 0.85). Heat pipes and high conductive carbon materials are effective means to improve the temperature uniformity.

Satellite Structure Concept

Two tendencies are noticeable. The first: payload and housekeeping are developed and arranged by one institution. In this case it will provide the thermal solution for the whole satellite. The second tendency: the multymission satellite bus with changeable payload is proposed to consumer. That means that payload should be accommodated into satellite thermal surroundings. The solar arrays may coat the satellite outer surface or could be deployable. After the opening they can not be re-oriented. The modern micro-satellite will carry the high precision optics, which is sensitive to satellite structure geometrical stability, and this requirement can essentially influence on structure design. Some the interesting details about micro-satellite thermal arrangement one can find on web sites of satellite developers and producers [15 – 18].

FUTURE POTENTIAL THERMAL CONTROL TASKS

The following tasks have been distinguished for satellite thermal control: to cool the high heat-generating components. The future problem is associated with growing heat dissipation of modern board computer processors reaching

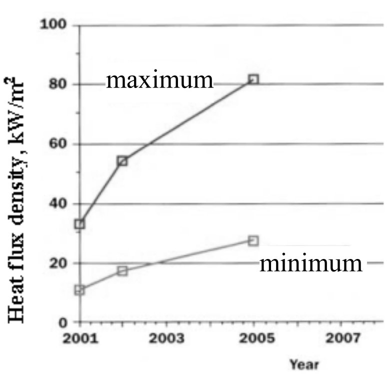

Figure 3. Predicted rate of heat flux density for space electronic components [22]

15÷30 W with density up to 100 kW/m^2 [22], (Figure 3 and Foil_6) to provide thermal and geometrically stable mounting places for devices (illustration on Figure 4, [23] and Foil_6); to ensure the conditions for operation of payloads with several temperature levels.

Other important direction is miniaturisation of space equipment that leads to reducing of sizes of

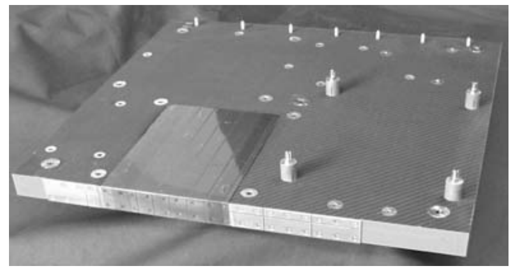

Figure 4. Thermo- and geometry stable payload baseplate of micro-satellite BIRD [23] provides the geometrical stability of mounting places and supports, parallelism of optical axes (± arcmin); provides the comfortable temperature regime for optics (+15 ÷ + 20 ^{0}C) and heat removal. Courtesy of DLR

all the components and increasing in heat flux densities of thermal control components.

SOFTWARE FOR THERMAL DESIGN

The wide set of the following products is available on the market (Foil_7):

1. Method of Lumped Parameters: ESATAN (ThermXL), SINDA/G/FLUINT, TRASSA

2. Finite-Element and Finite Difference Methods: NASTRAN, COSMOS, ANSYS, FLOTHERM, TAK2000

3. Radiation Heat Exchange (internal and external): ESARAD, TERMICA, TRASYS, RadCAD, SSPTA, OAZIS

4. Specific software is very useful for thermal contact conductance definition, thermoelectric cooler design and heat pipe design.

The following peculiarities and problems can appear during thermal model preparation:

1. Information retrieval and definition of accurate thermal properties: thermal conductivity, heat capacity, density, uniformity of properties, temperature dependence of properties

2. Information retrieval and definition of accurate optical properties and their changing during exploitation

- emittance, diffuse reflectivity, specular reflectivity, transmissivity in IR spectrum
- absorptance, diffuse reflectivity, specular reflectivity, transmissivity in solar spectrum

3. Definition of true values of thermal contacts in joints
4. Reasonable simplification of thermal structure of simulated object, preparation of the thermal load functions, choice of heat transfer process functionalities
5. The trace ability of mechanical and structural design changes in course of project and adequate modification of thermal model.

THERMAL HARDWARE

Main thermal control components did not vary abruptly. For micro-satellites they are mainly passive: multilayer insulation, optical coating and finishing, thermal conductive lines, thermal isolators, heat storage, heat pipes, electrical heaters. Conditional classification of hardware is presented in the Table 3.

Table 3. Classification of thermal hardware for micro-satellites

Name of hardware	Frequency of application
Passive means to regulate the satellite structure temperature level	
Radiating surfaces and finishing	Often
Multilayer insulation	Often
Heat transport means to manipulate with satellite integral heat capacity	Not often
Semi-passive and active means to regulate the satellite structure temperature level	
Heaters; Louvers	Not often
Heat storage; Variable conductance heat pipes	Very seldom
Changes of satellite orientation; Change of on-off of device	Possible
Hardware for individual devices installed on micro-satellite platform	
Passive means to regulate the device/equipment temperature level	
Radiating surfaces and finishing	Very often
Multilayer insulation	Often
Heat conductive lines (flexible lines, heat pipes)	Often
Low conductive supports and standsoff	Very often
Contact conductance means	Very often
Semi-passive and active means to regulate the device temperature level	
Heaters; Heat storage	Often
Thermal switches, Thermoelectric coolers, Stirling coolers	Seldom

Typical multilayer insulation (MLI) consists of 20 ÷ 30 inner layers: of less than 0.06 mm aluminised Mylar, innermost and outermost layers of 0.25 or 0.5 mm aluminised Kapton [11]. The effective emittance depends on discontinuity of MLI blankets, averaged temperature and pressure, pressing of blankets. The summary on effective emittance is presented on Figure 5 [11] (and Foil_8). Recommended value of effective emittance for draft estimation in [11] makes 0.03 with tolerance ±0.02. This value should be ascertained experimentally.

Optical coating and finishing, optical selecting

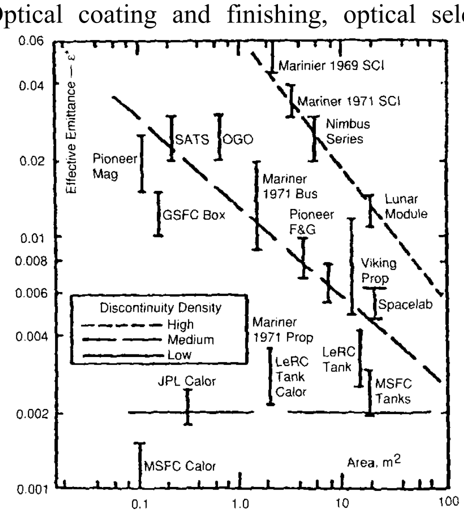

Figure 5. Effective emittance of MLI blankets as function of area and discontinuity by Stimpson & Jaworski [11]

coating can provide the value of ratio emittance/solar absoptance $0.1<\varepsilon/\alpha_s<10$. Typical optical painting for low temperature radiator has $\alpha_s \sim 0.2$, $\varepsilon \sim 0.85$. Figure 6 (and Foil_9) shows the optical properties for some finishing and

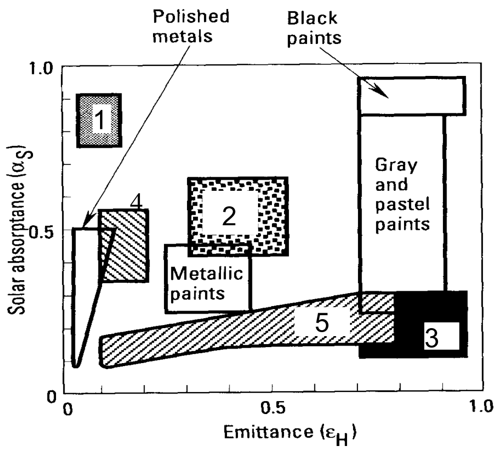

Figure 6. Values of emittance ε and solar absorptance [11]. Zone 1 - selective blacks (solar absorbers); 2 – sandblasted metals and conversion coatings; 3 – white paints and second surface mirrors; 4 -bulk metals (unpolished); 5 –dielectric films on polished metals. Ideal reflector ε=0; α_s=0 - in left bottom corner; ideal painting ε=1; α_s=0 – in right bottom corner; ideal solar absorber ε=0; α_s=1 - in left upper corner; ideal black painting ε=1 α_s=1 – is in right upper corner

coatings [11].

Heat transfer inside the satellite is realised by means of conduction through structural elements and special conductive lines (Foil_10). Aluminum alloys are used as bus, conductive lines and heat capacity storage (ρ = 2700 kg/m^3, λ = 150 W/m·K, c_p = 900 J/kg·K). Beryllium is used as heat transfer and heat storage (ρ = 1850 kg/m^3, λ = 180 W/m·K, c_p= 1850 J/kg·K). Conductive lines can be fabricated from flexible cooper strings [3, 4] as geometrically adjustable thermal conductors (thermal resistance less, than 2 K/W). They allow to conjugate the elements with unknown exactly layout and permit to reduce the torque on clamping points.

Other very effective heat conductance lines are the heat pipes. For space application they are mostly presented by ammonium axially grooved heat pipes produced by extrusion, though other variants of structures (as metal felts, screens) are used as well. Typical extruded profile has the diameter more, than 8 mm, thermal resistance less, than 0.1 K/W.

Selecting the type of capillary structure and liquid heat transfer media one can match these heat transfer instruments over the wide temperature range from –190 to +100 ^{0}C. Dimensional configurations of heat pipes are presented on Figure 7 [25].

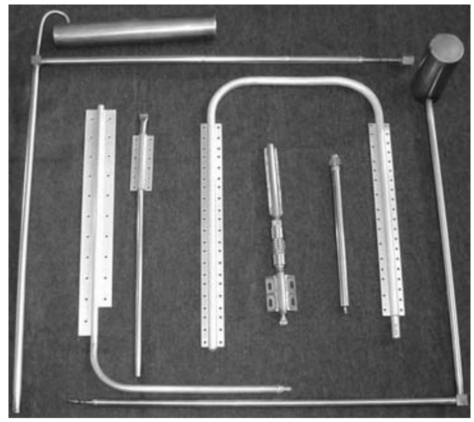

Figure 7. Examples of heat pipe configurations made of metal fiber capillary structure and stainless steel shell [25]

The illustrations of heat pipe application in micro-satellite thermal design are small satellites Magion 4, Magion 5 [1] and BIRD [2]. Scheme of heat pipe configuration for BIRD satellite thermal control system is presented on Figure 8 (and Foil_11). The aim of heat pipes in both cases is to

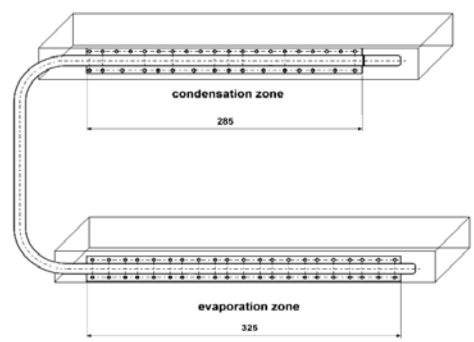

Figure 8. Scheme of heat pipe arrangement in BIRD satellite [2]. Heat is transferred from evaporator (payload) to condenser (radiator) in measurement phase of operation and vice versa in standby mode

transfer a heat between remote satellite zones (distance between front and back compartments for Magion satellites is 0.5 m; distance between payload and main radiator for BIRD satellite makes 0.3 m).

For future micro-satellite missions one of heat pipe modifications will be useful, namely micro heat pipes by circumcircle diameters from 1 to 6 mm. They are constant conductance heat pipes, have the wire wick or grooves as liquid transfer media; material of shell - copper and silver; typical circumcircle diameter within 1 ÷ 6 mm; length - up to 100 mm, heat carrier: alcohol, water; maximum transfer heat flux 2 ÷ 10 W/cm^2; thermal resistance of HPs: less, than 0.5 ÷ 10 K/W. Micro-heat pipe array dimensions are: width 20 ÷ 40 mm, 110 mm long with thickness of 1 ÷ 3 mm. Other type of new heat pipe technology, so-called loop heat pipes (LHPs), allows to operate independently of gravity forces. Distance between heat input and output zones can reach several meters at transfer power up to 100 W. Also the connecting lines in LHP are flexible (thin wall tubes by diameter of 2 ÷ 3 mm), that allows to apply this device for thermal coupling of moveable parts. Alternative solution for heat transport on the distance less, than 0.3 m and for heat spreading is based on the use of variety of graphite materials with effective conductivity more, than cooper [21].

Thermal contacts between satellite structural elements, thermal contact of high heat-generating components with cooling means or substrates could be improved by application of the different types of interface pads, greases, gap fillers, and encapsulates. Typically these means are placed between contacting surfaces that is presented on illustration 9 [26] and Foil_12.

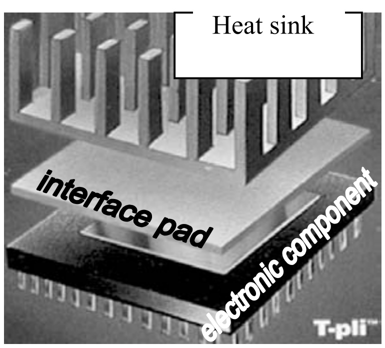

Figure 9. Scheme of interface pad application to improve the contact conductance [27]

Typical conductivity of pads is 1 ÷ 6 W/m K. They can be applied with not high pressure (typically 10÷30 psi or 0.7÷2 bar), and allow to reduce the contact resistance in coupling joint [27]. Other variant to cool the heat-generating microelements is to use the conformable gap fillers, which allow to connect the heat-generating component with thermal shield and to rearrange the heat flux. Scheme of conformable pad application is shown on Figure 10 (and Foil_12).

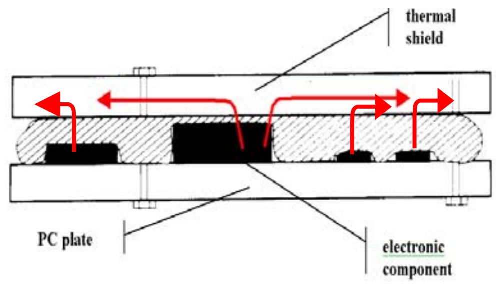

Figure 10. Cooling of electronic component with conformable pad, connecting the electronic component with cooling thermal shield

Potential problems in space application of interface pads, gap fillers, greases, encapsulates and adhesives associate with risk of mass losses and subsequent spacecraft and payload contamination. Other important task is to verify the stability of component characteristics under space environment factors and preliminary control of technical efficiency of selected application.

Also important passive thermal control means are the low conductive supports and standoffs. They are used for thermal isolation of device (or part of device) from satellite. Mainly such measure is necessary for devices with autonomous thermal control either for devices, or their parts, which have other temperature level, different from satellite insides. Typical materials for these supports are a wide variety of low-conductivity materials, including fibreglass, stainless steel, titanium and plastics. The choice of material is dictated by the conductivity, temperature range, thermal expansion and mechanical properties. Typical variant of support design, used many time in space for temperature range of 170 ÷ 350 K for thermal discoupling of 2 kg device is shown in Figure 11 (and Foil_13). Four such supports provide the thermal resistance more, than 400 K/W and practically disconnect device from satellite bus [4].

Such effective thermal control means as the louvers, radiators with changeable optical properties, unfolded radiators, radiators - variable conductance heat pipes, heat storage, thermal switches, controlled electrical heaters have been not considered in this lecture.

The louvers and radiators with variable emittances are more typical for larger satellite with mass more, than 100 kg. Heat storage, thermal switches, thermoelectric cooling and controlled electrical heaters are the attributes of autonomous thermal control technology. More details about above-mentioned instruments one can find in special references [11].

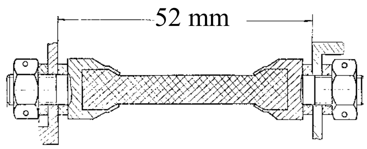

Figure 11. Scheme of low conductance standoff [4]. Insulating material – fibreglass with epoxy, attachment elements is made of stainless steel

Among advanced thermal technologies one can mark the following means, which are very close to be used in practice [11]:

1. Variable – emittance coating technology (variation of emittance $0.2 \div 0.8$)
2. Micro louvers (change of effective emittance in 2 times)
3. Mini two-phase heat transfer devices
   - mini heat pipes: diameters $1 \div 2$ mm, length $60 \div 100$ mm, heat power transferred $1 \div 10$ W
   - mini capillary pump loops: diameter $1 \div 2$ mm, length $300 \div 1000$ mm, heat power transferred $3 \div 100$ W
4. Mechanical thermal mini switches (thermal resistance change 100:1)
5. High performance C-C composites with conductivity of $400 \div 1100$ W/m K, which can be used either as the structure elements or heat transport elements.

GROUND TESTING AND VERIFICATION OF THERMAL CONTROL DESIGN

Verification of thermal design should be conducted prior the launch. For micro-satellites the vacuum chamber with inner sizes about 2-m length and 1.5 m in diameter is often enough for arbitrary satellite layout inside of the chamber. The vacuum chamber of pressure 0.0013 Pa (10^{-5} Torr) or less, equipped with cooled by liquid nitrogen screen simulates the thermal condition of outer space (Figure 12).

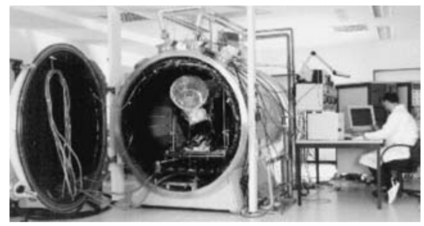

Figure 12. Space simulation chamber of DLR's Space Center in Berlin with volume of 3.2 m^3 [28]

The Earth, Sun heat fluxes are simulated by aiding heaters, installed on the back side of radiating surfaces, on the base of absorbed fluxes knowledge. Sometimes the Earth radiation is simulated by radiative heat sinks with regulated temperature. The thermal balance test of satellite system for cold and hot cases of exploitation should confirm or precise the accepted parameters of most of thermal control components and approve used thermal concept.

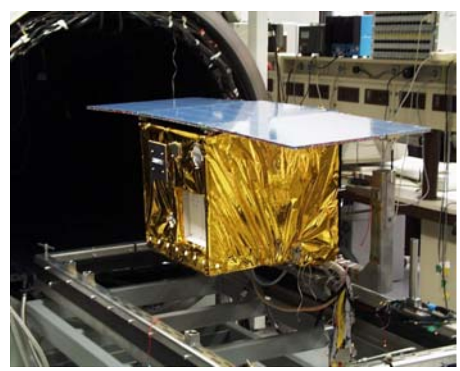

Figure 13. Installation chamber of micro-satellite BIRD into vacuum for thermo-vacuum testing. Courtesy DLR

Exploitation temperature boundary for satellite components should be enlarged by ± 11 ^{0}C to compensate the uncertainty with calculations, design and environmental conditions. Among possible recommendations, based on satellite design and flight performance experience, the following ones could be useful:

- Keep in mind the possibility of unpredictable situation with satellite orientation with respect to sun lights and with variation of heat generation value inside the satellite. They can shift satellite temperature level essentially
- Safe the battery in comfortable temperature range during charge-recharge mode and in a shadow period
- Degradation of optical surfaces is real process, which is very important for front directed toward the Sun surfaces
- Not all the factors can be simulated and estimated in the ground tests and by calculations. Design your equipment for the wider temperature range.

CONCLUSION

Summary of information concerning present-day thermal control concepts conformable to micro-satellite group (mass less, than 100 kg) is surveyed and analysed. The description of conventional thermal hardware used in micro-satellite design is briefly presented. The base for survey was the lessons learned at the National Technical University of Ukraine “Kyiv Polytechnic Institute” during the elaboration of thermal control hardware for micro-satellites Magion 4, 5 (launched in 1995 and 1996) and BIRD (launched in 2002), autonomous thermal control systems for interplanetary missions VEGA (launched in 1984), PHOBOS (launched in 1986) and other public data.

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