Experience of passive thermal control of long-term near-Earth small satellite mission

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Abstract: The microsatellite BIRD (Bispectral InfraRed Detection) with mass of 94 kg and overall sizes 0.55 x 0.61 x 0.62 m operates on near-Earth sun-synchronous orbit more than 11 years. The temperature range -10…+30 °C for payload and housekeeping equipment with average power of 35 W and peak power of 200 W in the observation mode (10…20 min) is provided by a passive thermal control system (TCS). The TCS maintains a thermal stability of the payload structure by use of heat transfer elements – grooved heat pipes, thermally jointing the satellite segments. Two radiators, multilayer insulation (MLI) and low-conductive stand-offs provide the required temperature level. An analysis of TCS performance includes the definition of minimal, maximal and average temperatures of satellite units and their comparison with the designed parameters. The elaborated passive TCS successfully keeps the nominal temperature level of satellite components during one-year designed period of exploitation and sequent 10 years.

1. INTRODUCTION

The main features of BIRD (**B**ispectral **I**nfra**R**ed **D**etection, launched on 22 October 2001) are presented in [1], the description of the thermal control system and some summaries of operation – in [1-3]. This satellite is intended to demonstrate in space a new compact infrared imaging sensor. BIRD is a cubic shaped 3-axis stabilized microsatellite without a propulsion system. The satellite bus is designed as a three-box body, and consists of the service segment (SS), the electronic segment (ES) and the payload segment (PS), fig. 1.

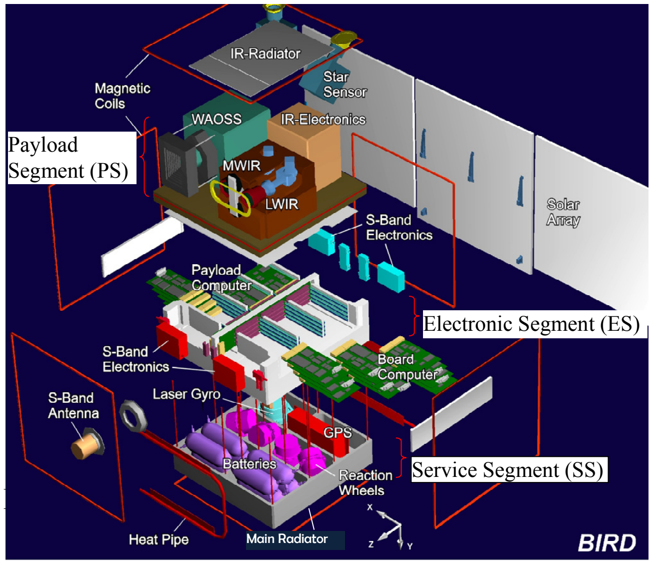

Fig. 1: Scheme of satellite structure and components

The satellite external surface is covered with MLI except the instruments' windows, antennas and two radiators. The satellite operates on the circular sun-synchronous orbit 570 km. Satellite bus has dimensions of 0.55x0.61x0.62 m, a total mass of 94 kg to launch it as piggy-back payload.

Payload instruments are assembled on payload platform and are placed in payload segment. They include multi-spectral sensor optical systems LWIR and MWIR with cooled infrared sensors (operating temperatures -193 ^{o}C and -168 ^{o}C, correspondently) and Stereo Scanner WAOSS (operating temperature +20 ^{o}C). For the first time for microsatellite practice, the heat pipes (HPs) were used. Typically the main heat pipe functions in this application are: heat transfer between segments in the wide temperature range -50…+50 ^{o}C and temperature regulation of payload segment platform.

2. THERMAL CONTROL SYSTEM CONCEPT AND HARDWARE

BIRD TCS is passive and uses the dissipated heat of spacecraft housekeeping equipment and payload to maintain the system on required temperature level. The BIRD TCS is based on the usage of the following thermal control elements:

- MLI for protection of inner satellite content against external light disturbances, coming from the sun and the earth;
- sized radiators (main and IR) with selective optical coating to radiate the heat generated by equipment on appropriate temperature level;
- two heat pipes, which thermally connect the SS and PS to solve the problem of heat transfer between two thermally disconnected segments and reducing the variation of temperature by increasing the thermal mass.

Proposed thermal design is based on the following assumptions. In order to minimize the thermal deformation of assembly of two segments (PS and SS) the use of interface elements with low temperature expansion is required. Typically such materials have low conductivity that leads to the high thermal resistance between PS and SS R_{SS-PS} = 10…20 K/W that may produce a great temperature difference between them at heat transfer. Heat pipes with much smaller thermal resistance and montage flexibility allow minimizing the temperature difference between segments to less than 10 ^{o}C (fig. 2), providing flexible mechanical connection between them.

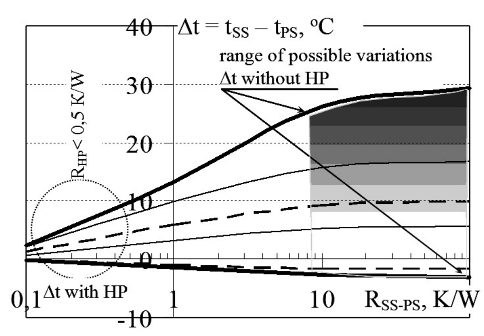

Fig. 2: Predicted temperature difference between mounted sides of satellite segments as function of thermal resistance between them

The technical requirements to parallelism of optical axes of payload optical instruments and to geometric location of focal planes have initiated the selection of payload baseplate design as the assembly of two carbon fibre layer (7), aramid honeycomb structures (3, 6) with embedded between them 3 mm layer of Carbon-Fibre-Carbon (2), having low coefficient of thermal expansion and thermal conductivity of 155 W/mK (fig. 3). Heat pipe 1 is attached to flange 8 which has thermal contact with layer 2. The insets 4 and 5 are thermally connected to conductive later 2 and are used for clamping of instruments to payload platform [1].

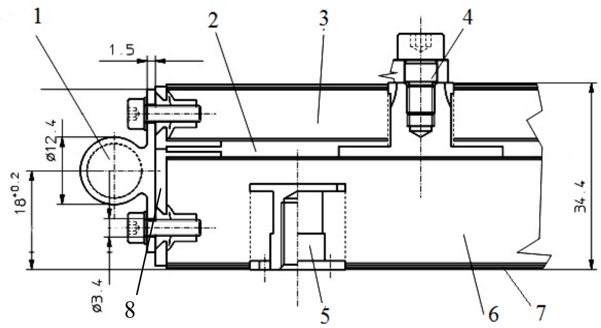

Fig. 3: Scheme of heat pipe thermal interface with payload platform (carbon fibre honeycomb structure). Courtesy of DLR

Another reason of heat pipe application is associated with the possibility to increase the thermal capacity of the satellite main radiator (located in service segment) by thermally connecting segments one to another. At higher thermal mass the impact of periodically inner power generation and external heat fluxes on main radiator produces narrower temperature variation in comparison with segments with separated radiation surfaces.

3. HEAT PIPES QUALIFICATION

The heat pipe with the following parameters has been selected. HP shell is the round extruded profile made of aluminium alloy. Flanged zones are only in evaporation zone (325 x 30 x 1.2 mm) and condensation zone (285 x 30 x 1.2 mm). The direction of heat flux could be from evaporator to condenser and backwards. Two heat pipes are used in TCS.

Table 1. Main parameters of heat pipes for BIRD thermal control system

Shell diameter, m	0.0122	Vapour core diameter, m	0.008
Wall thickness, m	0.001	Heat carrier	ammonia
Length of evaporator, m	0.325	Exploitation temperature, ^{o}C	-50...+50
Length of transport zone, m	0.344	Thermal resistance for exploitation temperature , K/W	less than 0.07
Length of condenser, m	0.285	Transported heat, W·m	not less than 60
Effective heat transport length, m	0.65	Rated heat transfer, W	60
Bending radius, m	0.048	Long-life tests, hours	32000
Shell protection	polishing	Flange material	aluminium alloy
Capillary structure: - groove depth / width	axial grooves	Mass per unit, kg	less than 0.23

/groove opening, mm Quantity of axial grooves	1 / 0.5/ 0.45 30		

The programs of the acceptance and qualification tests are elaborated according to ESA requirements [5]. The heat pipe test programs elaborated by authors for VEGA Project (1984-86), Phobos Project (1986-87), and Mars – 96 have been applied as well. The following main tests have been foreseen: (a) inspection and physical measurements; (b) proof pressure test - leak test; (c) performance testing; (d) burst test; (e) random vibration; (f) storage simulation test; (g) thermal cycles/shock test; (h) aging test (long life test).

Among performance tests are the following:

- definition of thermal resistance of heat pipes
- definition of maximum heat transfer ability
- definition of temperature distribution along the heat pipe length
- definition of priming time of heat pipe after the full dry-out of an evaporation zone
- definition of heat pipe capability to start-up with 80% of maximum heat transport capacity at various vapour temperatures.

In these tests the thermal technical characteristics of heat pipes in the wide range of input heat fluxes (from 5 to 170 W) and constant vapour temperature (t_v) have been checked. Vapour temperature was kept stable at the levels: -50°C, -20°C, 0°C, +20°C, +30°C, +50°C independently of input heat power. At the vapour temperatures –50°C and -20°C the tests were carried out in a vacuum chamber with a nitrogen cooler. For t_v = 0…+50°C the tests have been conducted in air medium with force liquid cooling of condenser zone.

Figure 4 illustrates the heat pipe generalised performance within exploitation temperature range, namely shows the thermal resistance variation R_{hp} as function of vapour temperature and (QM, FM – marking of HPs) function of transmitted power Q. Qualification and flight heat pipes had the similar experimental dependencies $R_{hp} = f(Q)$, and the dispersion of R_{hp} for the same value of Q is less than 0.01 K/W.

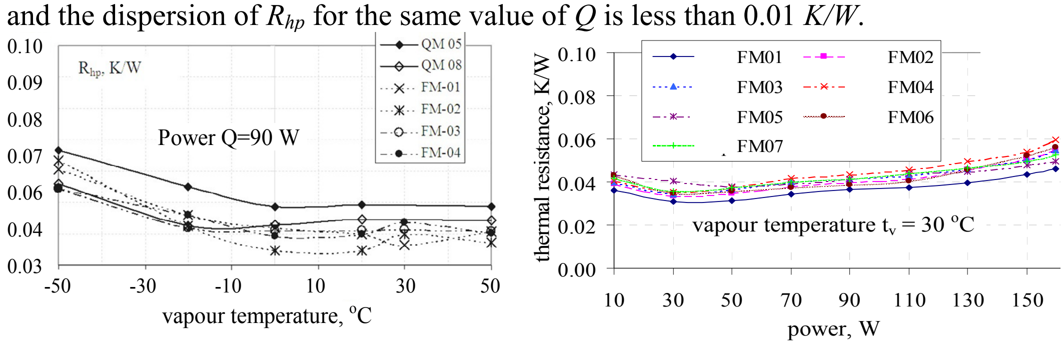

Fig. 4: Heat pipe thermal resistance R_{hp} as function of vapour temperature at constant power, and R_{hp} as the function of transported power Q. Courtesy of NTUU "KPI"

Maximal heat transport rate of heat pipe: $Q_{max} \cdot L_{eff}$ (W·m) = $-2.95 \cdot 10^{-3} \cdot T_v^2 + 1.873 \cdot T_v - 212.4$, where L_{eff} = 0.65 m - effective transport length of heat pipe, T_v – saturation vapour temperature in K. This dependence summarizes the experimental data with deviation of ±10 % for 8 tested heat pipes.

4. THERMAL CONTROL SYSTEM OPERATION IN FLIGHT

After successful launch with Indian PSLV (Polar Satellite Launch Vehicle, Shar Center, India), the first flight performance of BIRD thermal control system was evaluated in [3]. During 10 years of satellite operation the its orbit has changed from intended sun-synchronous orbit. Due to drift of orbit plane, right ascension of ascending node (RAAN) is changed by ~81°. Stable shaded orbits with shadow duration about 35 min at DOM < 2200 has transformed at DOM 3245 to un-shadow orbits during September 2010 (fig. 5). The orbit inclination i has minimum i = 97,7° in the range of day of mission (DOM) 1000…1500, the orbit period was reduced from 96 to 95 min, the semi axis value – from 6945 to 6901 km, eccentricity – from 0,0019244 to 0,0014283.

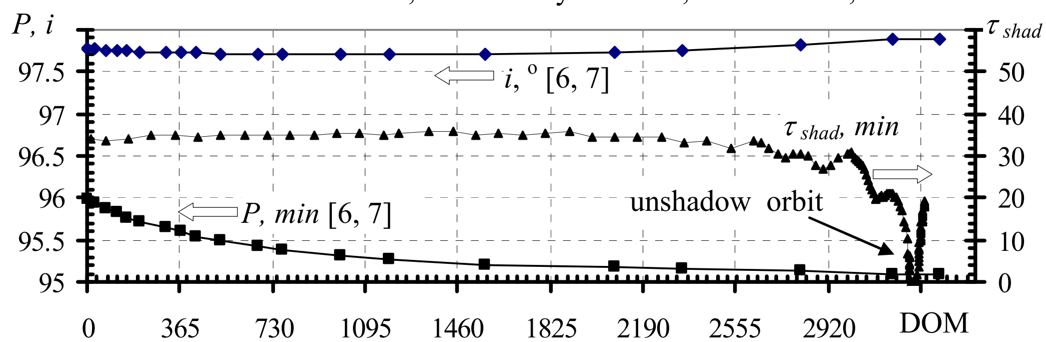

Fig.5: Variation of orbit inclination i, orbit period P on the base of [6, 7]. Shadow time τ_{shad} was complied on the base of sun sensor telemetry

The on-board temperature measurements are performed by 33 temperature sensors of the type AD590. The sensors have been spread over the satellite structure, payload, housekeeping equipment and solar panels (fig. 6) and are interrogated every 30 s.

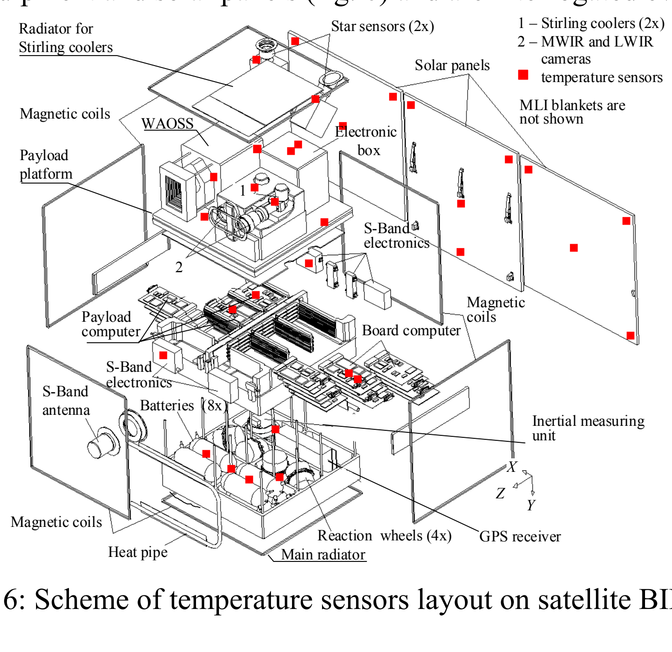

Fig. 6: Scheme of temperature sensors layout on satellite BIRD

The on-orbit temperature variation for the main radiator is inside of ΔT = 1…4 °C, for payload platform – less than ΔT = 2 °C. Payload platform temperature has the similar oscillatory character as radiator one. Sometimes the radiator has the more essential rise of temperature due to satellite manoeuvres. The increasing of S-band transmitter temperature during operation (7 min) makes ΔT= 10…14 ^{o}C.

The temperatures of all the elements of microsatellite have evident periodic character, similar to the radiator that deals with nonuniform external thermal condition for satellite main radiator along orbit and tight thermal contact between radiator and other units. Figures 8 illustrates the temperature history during one day variation of main plate (radiator) and payload platform, which thermally connected by heat pipes.

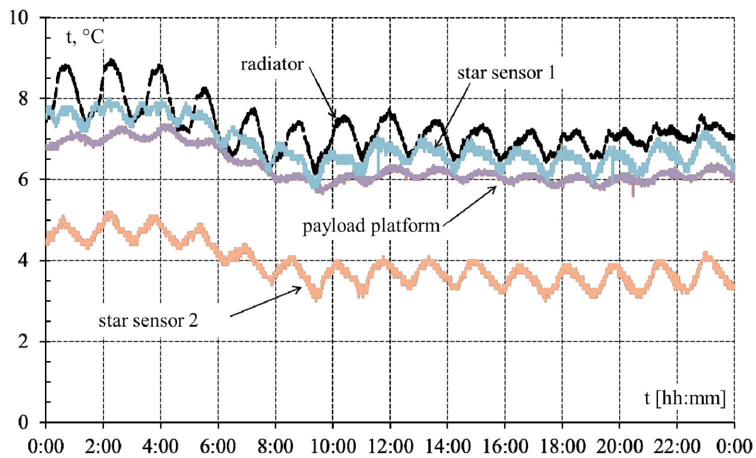

Fig. 7: Temperature variation for payload platform and radiator on 02.01.2002

The solar panels have considerably lower thermal mass and are directly illumined by the sun, therefore they have the widest range of temperature excursions. The central panel, cooled from one side only, has the maximal temperature +75…+90 ^{o}C and minimal -60 ^{o}C in shadow. The panels -X and +X (heat is radiated by two sides) have the maximal temperature +60…+70 ^{o}C and minimal temperature less than -70 ^{o}C .

There is a certain interest to review the satellite temperatures during the whole period of its exploitation. In order to reduce the volume of summarized file, having saved the most important features during every day, the following algorithm has been proposed for temperature and power data processing: to find the maximal $T_{max} = max(T_{\tau 1} : T_{\tau 2})$, minimal $T_{min} = min(T_{\tau 1} : T_{\tau 2})$ and mean integral values for certain period of processing $\tau_2 : \tau_1$. The interval $\tau_2 : \tau_1$ = 24 hours has been accepted as it coincided with telemetry timing output interval. After processing of telemetric data every temperature is presented by three values (min, max, mean integral) at the same time.

Analysis of TCS flight performance shows that it provided the accepted temperature range for satellite equipment. The variation of temperature non-isothermality over the payload platform (difference between the maximal and minimal temperatures of the platform) is within ±2 ^{o}C. The temperature difference between the radiator and payload platform is over the range of -2…+6 ^{o}C (fig. 7), that corresponds to predicted.

<u>Temperatures of main satellite units.</u> Designed limits (minimal and maximal) and obtained minimal and maximal temperatures are the following. Optical Box of infrared

camera -15/ 15 °C and -9.9/ 22.9 °C; Electronic box of infrared camera: -40/ 45 °C and -2/ 44.4 °C; Optical camera WAOSS: -15/ 30 °C and -10.1/ 33 °C; Payload platform: -20/ 30 °C and -2.7/ 32.1 °C; Star sensors 1, 2: -25/ 30 °C and -6.9/ 32 °C; System of information processing SBC_A, SBC_B:-15/ 40 °C and 11/ 46.5 °C; Board computers 1,2: -20/ 60 °C and -2.5/ 37.9 °C; Transmitters 1, 2: -20/ 60 °C and -3.4/ 56.1 °C; Main radiator: -15/ 35 °C and -2.6/ 40 °C; Batteries: -10/ 25 °C and -0.6/ 48.0 °C; Side solar panel +X: -120/ 100 °C and -94.7/ 76.1 °C; Side solar panel –X: -120/ 100 °C and -96/ 73.4 °C; Central solar panel –Z: -120/ 100 °C and -78.1/ 88.9 °C. Most of time the instrument's temperatures were within the design limits, used for elaboration of payload and housekeeping equipment.

a)

b)

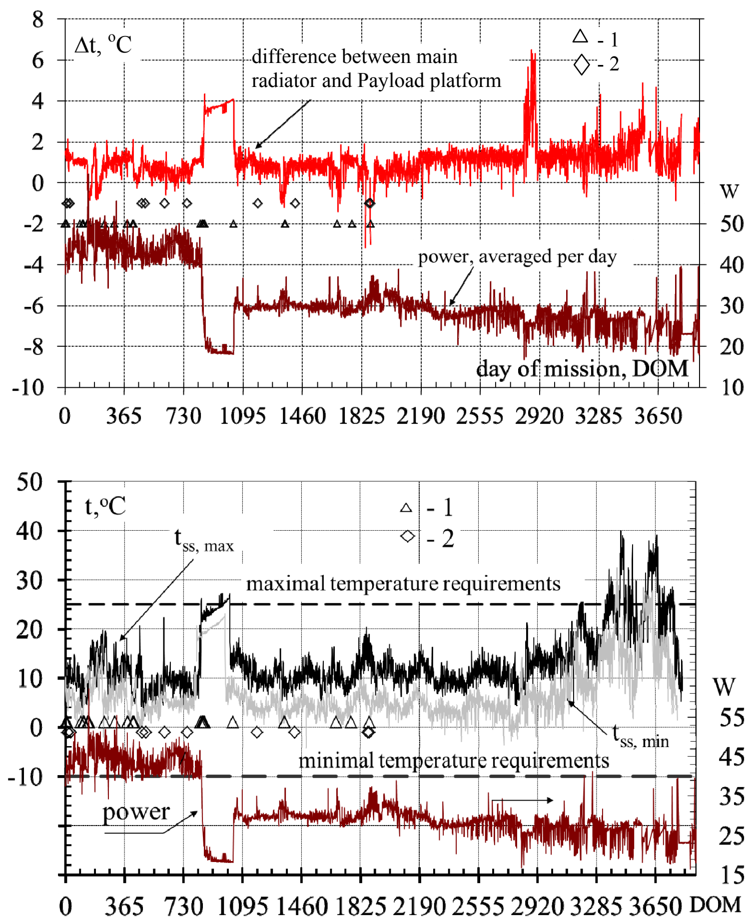

Fig. 8: Average temperature difference between SS and PS platform (a) and main radiator minimal and maximal temperatures (b) during BIRD flight operation. Right axis – power consumption. 1, 2 – technical events on satellite

5. CONCLUSIONS

During the period of 2001-2012 y.y. of space exploitation of microsatellite BIRD the TCS held the design characteristics in near-Earth orbit space conditions. The passive principle of thermal control system has been applied for low sun-synchronous orbit. It is based on the creation of centralized heat transfer system, in which the most of satellite components are thermally attached to the main radiator and have the similar to radiator temperature.

For the first time in space practice of microsatellite design two axially grooved aluminium HPs have been used as the heat transfer line on distance about 0.5 m between two compartments of microsatellite. Elaborated heat pipes and other elements of thermal control system kept their characteristics in ground tests and at long-term space exploitation. The temperature of satellite components was kept inside of designed temperature limits.

6. REFERENCES

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