Pre-Flight Analysis, Test Evaluation and Flight Verification of the Thermal System of Tohoku University SPRITE-SAT

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The microsatellite SPRITE-SAT developed by Tohoku University was launched in January 2009. Regarding the thermal system of SPRITE-SAT, the mathematical model, the parameter determination for the heat transfer coefficients, and the comparison between the estimate and real temperature in flight mode are shown in this paper. The precision of thermal analysis using the simple 7-node model was solved. The estimate error of temperature in orbit is less than 5 degrees Celsius in panels with most of instruments.

Key Words: Thermal Analysis, Thermal Chamber Test, Microsatellite, Sprite

1. Introduction

Tohoku University developed the 50-kg microsatellite SPRITE-SAT, and it was safely launched by Japanese rocket H-IIA-15 as one of piggy-back satellites from the Tanegashima Space Center, Japan. The launch time is 12:54pm JST on January 23, 2009. In the first observation pass at 14:32pm JST, at the ground station of Tohoku University, it was confirmed that the telemetry signals were normal and the sending commands was successful.

Transient luminous events (TLEs) induced by lightning discharges happens about four-million times a day in the world (Fig. 1). However they are familiar natural phenomena, many things are still unsolved. The sprite phenomena were found in 1989, which are huge luminous phenomena happening over thunderclouds by lightning discharges, and the Terrestrial Gamma-ray Flashes (TGFs) was found and reported in 1994. The primary missions of SPRITE-SAT is the observation of these luminous phenomena relating to lightning discharges, and the observed data contribute to solve the mechanism.

After the satellite separation, the SPRITE-SAT was

safely operated for 12 days. As of May 2009, however, the satellite is suffering from a serious trouble. This trouble was firstly confirmed in the night observation (JST) on February 4, 2009. At first, the telemetry signals are not modulated, or the valid telemetry data cannot be obtained. Secondly, the commands from ground station are not executed at all. On the other side, the transmitter and battery are normally working because the carrier signals can be monitored every daytime and night. Under these conditions, the possibility of the central control unit broken is estimated.

In this paper, the mission and system outlines of SPRITE-SAT are shown firstly. Secondly, regarding the thermal system, the mathematical model, the parameter determination for the coefficient of heat transfer, and the comparison between the estimate and real temperature in flight mode are shown. About this satellite, the attitude control system was described in previous paper¹⁾.

The design methodology of micro-satellite thermal control^{2,3)} and the low-cost test method using a thermal vacuum chamber⁴⁾ were shown. In addition to low-cost instruments and structures, the methods of low-cost test and analysis are needed for development of micro satellites. In this paper, temperature estimation in flight mode using very simple mathematical model is carried out, and model parameters are determined using measurements of thermal parameters are determined using measurements of thermal parameters tests. In addition, the thermal measurements in real flight were analyzed. Comparing the estimated values with flight data, it was confirmed that the estimation using the simple model was sufficiently effective.

2. Outline of the Satellite Missions and System

The image of installed mission instruments is shown in Fig. 2, and the observation image is shown in Fig. 3. All

Fig. 1. Transient luminous events (TLEs).

Fig. 2. Mission instruments.

Fig. 3. Image of observation.

Fig. 4. Interior appearance without side panels.

the observations are carried out in the less than ± 20 degrees of latitude in night side.

For sprite observations, the spectrum camera LSI-1 is used for lightning (740-830 nm), and the spectrum camera LSI-2 is used for sprite (762 nm). The field of view is about 29 degrees both, which is equivalent to 342 km in ground distance.

For TGFs observations, the wide-view camera WFC is used for lightning, and the gamma-ray counters (TGC) are used, courtesy of JAXA/ISAS. The field of view of WFC is 134×180 degrees.

Additionally, the VLF receiver courtesy of Stanford University and the extra CCD camera for the purpose of technical verification of star sensor are installed. The deployable mast is used as VLF antenna as well as attitude stabilization.

The system specifications are shown in Table 1. The total size at launch is 500 x 500 x 494 mm, and the mass is 43.9 kg. The 86-cm deployable mast is installed, and the attitude is stabilized using the 3-axis magnetometer and the two magnetic torquers.

The interior structure is shown in Fig. 4. There is the central pillar structure, and the almost bus instruments are installed on this center structure. The solar cells are mounted on the top panel and the 4 side panels. As shown in Fig. 2, the almost mission instruments (LSI-1, 2, WFC,

Table 1. Specifications of SPRITE-SAT.

Size	at launch	W 500 x D 500 x H 494 mm
	after mast deployment	W 500 x D 500 x H 1354 mm
Weight	total	43.9 kg
Orbit	type	Sun Synchronous Orbit (SSO)
	localtime	1pm at descending node
	altitude	661-km circle (98-min period)
	inclination	98 deg
Attitude Control	type	gravity gradient stabilization
	deployable mast	860-mm size extension tip-mass: 3680g, sleeves: 500g
	sensors	3-axis magnetometers (in GAS*)
	actuators	2-axis magnetic torquers
Attitude and Orbit Sensors	attitude sensors	3-axis sun sensors (6 solar cells) 3-axis magnetometers (in GAS) 3-axis magnetometers (in TAMU**) 3-axis gyro sensors (in TAMU) 3-axis accelerometers (in TAMU)
	orbit sensors	GPS receiver
Power	solar cells	P-type single crystalline silicon 42 series/panel x 5 panels
	batteries	9-cell NiMH (total 3.3AH, 10.8V)
	power generation	23.5 W (avg. in 62-min sunshine per
	power consumption	22.0 W (at observation mode) 12.0 W (at communication mode) 9.5 W (at typical mode)
Communication	uplink	UHF, 1200bps at Sendai station, Japan
	downlink	S-BAND, 0.1W, 9600bps or 1200bps main: Sendai station, Japan sub: Kiruna station, Sweden

* GAS = Geomagnetic Aspectmeter Unit

** TAMU = Tohoku-AAC MEMS Unit

TGC) are installed on the bottom panel, and the CCD camera and the GPS receiver is installed on the top panel. The deployable mast is stored in the inside upper of central structure, and the power control unit and battery unit is stored in the lower part.

The 6 outer panels (top, side, and bottom) are aluminum-honeycomb sandwich panels. The skin thickness is 0.3-mm A7075-T6, and the core is 10-mm A1/8-5056.001P (except bottom) or 20-mm A1/8-5056.002P. The center structure consists of 4 aluminum alloy panels, which are 5-mm A7075-T6. The surface of all the aluminum panels and the skin of honeycomb panels are coated by Alodine (Alocrom) #1000.

The mass distribution of each panel is defined as the total mass of a panel and the installed instruments. The top panel is 1.7 kg, the side panel is 0.7 kg (except -X panel, it is 1.1 kg), the bottom panel is 12.6 kg, and the center structure is 26.4 kg.

3. Mathematical Model of Thermal System

The control of thermal system is passive. There are no active thermal controllers such as heaters. To properly regulate the thermal conditions, the paint and tapes are used.

In the thermal analysis, the satellite is modeled by 7-node structures shown in Fig. 5. Although the real

Fig. 5. Numerical analysis model.

Fig. 6. Surface of panels.

(i: top panel, ii: side panels, iii: bottom panel)

element	α	ε	panel	$A(m^2)$	α	ε
solar cell*	0.68	0.80	top	0.2397	0.463	0.619
aluminum	0.17	0.10	side	0.1862	0.517	0.652
MLI	0.05	0.05	bottom	0.2397	0.118	0.078
Kapton	0.48	0.89				

*power generation efficiency = 0.17

center structure is a rectangular column, which width is 225 mm, it is modeled by one flat board. The top panel (or bottom panel) and the side panel are combined by bolts, and the coefficient of heat transfer is defined by C_{ts} . The top panel (or bottom panel) and the center structure are also combined by bolts but the 5-mm insulation plate made by glass epoxy is inserted between panels, and the coefficient is defined by C_{tc} . Between side panels, the panels are not combined, and there is about 1-mm space. In thermal vacuum tests, these two parameters are determined.

Using commercial thermal analysis software, the temperature estimation at plenty of nodes is possible. However, the analysis time is not short, so the result in different setting parameters cannot be calculated quickly. Also, the team members cannot realize the validity of results because of the complexity of model. So, it is important to build a simplified analysis model at the same time. A 7-node simple thermal model is reasonable solution because it can be developed in short period. The center structure is simplified as a same-sized panel of a side panel because the calculation of view factor matrix is

Fig. 7. Thermal vacuum test.

Fig. 8. Photo and CAD model of IR heater cage.

easier compared to a rectangle pillar or a small-size plate. In the flight result, the battery temperature was temporarily increased to very high level. This cause was not the model error but the fault of power charge controller.

The photos of each panel are shown in Fig. 6. The each surface consists of solar cells, aluminum, multi layer insulation (MLI), and Kapton tape. From the area and the heat specification of each element, the solar absorption α and thermal emissivity ε of each panel are calculated in Table 2. The α of solar cells is the effective value in thermal analysis. The specification value by vendor is 0.85, and the efficiency of power generation is 17.0 %.

For the inside of panels which is not exposed to outside, a thermal control tape is used to the top panel and the side panels. The surface of this tape is an aluminum film, and the heat emissivity is 0.035, which is equivalent to aluminum. And, a black paint is used for the bottom panel and the center structure, which heat emissivity is 0.874. The area of center structure is 0.1645 m², and the area of other panel is shown in Table 2.

The time derivative of temperature is expressed by Eq.(1). The suffix i is the number of structure node (1..7).

$$m_i c_{pi} \frac{dT_i}{dt} = Q_i - \sum_{j=1}^n C_{ij} (T_i - T_j) - \sum_{j=1}^n \varepsilon_i \varepsilon_j F_{ij} A_i \sigma (T_i^4 - T_j^4) \quad (1)$$

where m_i is the mass, which is shown in Sec. 2, c_{pi} is the specific heat, which is defined by 879 J/kg.K (equivalent to A7075) in all the nodes. The T_i is the temperature (K), and dT_i/dt is the time derivative of temperature per unit of time (s). The Q_i is the sum of heat input and output (W) to external environment. The C_{ij} is the coefficient of heat transfer (W/K) between nodes, ε_i is the heat emissivity, F_{ij} is the view factor

between nodes, and A_i is the area (m^2). The σ is the Stephan-Boltzmann constant, which is $5.671 \times 10^{-8} \text{ W/m}^2\cdot\text{K}^4$.

4. Parameter Determination of Heat Transfer Coefficients by Thermal Vacuum Test

4.1. Outline of thermal vacuum test

In the development process of SPRITE-SAT, the thermal vacuum test was carried out two times for the flight model as shown in Fig. 7. The tests were carried out at the JAXA Kansai Satellite Office located in Osaka, Japan. In this test, the satellite is stored in the cylindrical chamber, which temperature is about -200°C , and the inside of chamber is coated by black paint. The satellite is covered by the IR heater cage courtesy of JAXA, shown in Fig. 8. The panels of satellite are radiantly heated from six directions with different heat strength. The power (W) for heaters can be adjusted to favorite values from the out of chamber. From the results of this test, the parameters of heat transfer coefficient are determined, and the precision of temperature estimation in orbit is improved. If necessary, the strategy of passive thermal control such as the area of tapes is modified.

The 1st test was carried out in October, 2008, but the correct temperature could not be sufficiently obtained because the trouble happened in connectors of thermocouples sensors. However, because the temperature of satellite was lower than expected from the partial good data, the area of the tapes of panel surface was adjusted, and the temperature in orbit is improved to increase by about 3°C . After this modification, the 2nd test was carried out in November, 2008. The total 36ch of thermocouple sensors are put to the satellite surface, and the total 17 ch of same sensors are put to the IR heater cage. The total 10 different phases of heater power combinations are applied. In the analysis of this paper, the results of the phase-7 (heated from the top panel and the 4 side panels) and the phase-8 (heated from the top panel and the 2 side panels) are analyzed, and the model parameters are determined.

4.2 Data analysis and results

Firstly, the view factors from each panel to IR heaters are calculated using the software ANSYS(R), which is usually used for the thermal and structure analysis. The results are 0.37 from top panel to its front heater, and 0.41 from side panel to its front heater. Also, when the side heater next to front heater is visible from panel, the view factor is about 0.01.

From the temperature difference between heater and panel, the heat input (W) to satellite can be calculated. Also, from the temperature difference between panel and chamber, the heat output (W) of satellite can be calculated. The temperatures of two and more points in each panel or heater are measured, and the average value is used for analyses. When the temperatures are in the balance state, the heat input and output between the satellite and the external environment are balanced, that is, the total heat is

Table 3. Example of thermal balance calculation.
(phase-7, top panel)

panel (i)	external (j)	T_i (degC)	T_j (degC)	$\mathcal{E}$	F	Q (W)
top	heater	13.4	109.0	0.836	0.37	38.2
	chamber		-194.2	0.810	0.61	-27.8
total (W)						10.4

Table 4. Results of thermal balance calculation.

panel	phase-7		phase-8	
	heater on	Q (W)	heater on	Q (W)
top	Yes	10.4	Yes	11.4
side (+X)	Yes	-2.6	Yes	8.6
side (-Y)	Yes	-3.7	Yes	6.5
side (-X)	Yes	-0.3	No	-12.2
side (+Y)	Yes	-3.0	No	-13.5
bottom	No	-0.7	No	-0.8
total		-0.1	total	0.0

Table 5. Results of real and estimate temperatures.

panel	phase-7					phase-8				
	epoch temp. (degC)	balance temp. (degC)	balance temp. (degC)			epoch temp. (degC)	balance temp. (degC)	balance temp. (degC)		
	real	real	estimate	error		real	real	estimate	error	
		(i)	(ii)	(i - ii)			(i)	(ii)	(i - ii)	
top	3.4	13.4	11.9	1.5		13.3	14.6	11.7	2.9	
side (+X)	4.6	4.7	2.4	2.3		4.9	20.9	23.6	-2.7	
side (-Y)	2.0	-0.3	-0.1	-0.2		0.8	15.5	19.9	-4.4	
side (-X)	3.7	3.3	7.6	-4.3		2.6	-11.7	-12.6	0.9	
side (+Y)	2.4	0.4	1.5	-1.1		-0.9	-19.1	-14.6	-4.5	
bottom	11.6	5.5	5.0	0.5		5.6	6.6	6.1	0.5	
center	6.8	8.7	7.5	1.2		8.6	9.1	8.5	0.6	

* balance temp. is measured 3h51m or 3h43m after epoch temp. in phase-7 or 8.

Table 6. Results of parameter determination.

determined parameters	phase-7	phase-8	average
Cts (W/K)	0.197	0.267	0.232
Ctc (W/K)	0.658	0.941	0.799

zero. The heater surface and the chamber interior are coated by black paint. The heat emissivity of chamber is fixed to 0.81, and the emissivity of heater is adjusted for the total heat to be zero.

The calculation example of the top panel in phase-7 is shown in Table 3. The balance temperatures of panel, heater, and chamber are 13.4, 109.0, and -194.2°C respectively, then the total heat is 10.4-W input. For all the panels in phase-7 and phase-8, the total heat is calculated as shown in Table 4. It is confirmed in phase-8 that the heat is inputted from the heater-on panels and outputted from other panels.

From the initial temperature, the balance temperature, and the elapsed time, the parameters of heat transfer coefficients C_{ts} and C_{tc} are determined. The elapsed time is 3h 51m and 3h 43m in phase-7 and 8 respectively. Using the batch state estimation method, when the square of difference temperatures between the real and estimate in balance state are minimum, the parameters are defined as optimum values. The real, estimate, and difference (error) values are listed in Table 5. Also, the determined parameters in each phase and the average values are listed in Table 6. The temperature error after determination was

Fig. 9. Statistics of thermal range in single period.

4.5 °C in maximum. In the temperature estimation in orbit, the average values $C_{ts} = 0.232$ W/K, $C_{tc} = 0.799$ W/K are adopted.

In this test, the settings of heater were adjusted by trial and error, and the valid data for analyses were limited. Therefore, the data from only 2 phases of 10 were analyzed. In failure phases, when the bottom panel was heated, the jig for the install of satellite into chamber was directly heated, and the analysis becomes very difficult. In other failure phases, when the temperature difference between initial and balance values is very small, the parameter determination becomes difficult because the sensitivity of measurements to parameters are very small.

5. Flight Data Analysis

Using all the real-time data observed in visible passes and all the downloaded data measured in invisible time, the temperature history of 98-min single period is statistically analyzed as shown in Fig. 9. The target is all the data in flight since the satellite separation time on Jan. 23, 2009 UTC until the final observation because of hardware trouble on Feb. 4, 2009 UTC. The statistics method used in this paper is original.

The procedures of analysis are as follows: 1) the poor data over $\pm 60^\circ\text{C}$ are omitted, 2) the 98-min single period is separated into 2-min segments, and the average and the standard deviation σ are calculated in each segment, 3) the poor data over $\pm 4\sigma$ are omitted and the calculation of 2) is carried out again, 4) the poor data still remain, so the data is sorted in ascending order, and the 10% data in maximum and minimum values are omitted respectively. The maximum and minimum data of remaining data after 4) are defined by MAX and MIN in Fig. 9.

It was confirmed that the correct measurements do not exist in the out of $\pm 60^\circ\text{C}$ in this case, so the wrong data happening instantaneously are omitted. The timing of temperature measurement is not stable, so the number of sampled data in each 2-min segment is different. The total number of data is 56042 in each node, in which the points in segment are from 253 to 4815, except for the side panel using 4-time number of data.

The temperature range of top panel (i) and side panels (ii) is comparatively large, which are $-21.5^\circ\text{C} \sim +49.6^\circ\text{C}$ and $-23.2^\circ\text{C} \sim +40.2^\circ\text{C}$ respectively. The differences ΔT of average temperature are 52.3°C and 49.7°C respectively. In both histories, the temperature starts decreasing around at the eclipse beginning, which is 62 mins after sunshine beginning.

The temperature range of bottom panel (iii) and center structure (iv) is small compared to the top and side panels. The ranges are $+1.9^\circ\text{C} \sim +22.9^\circ\text{C}$, $+7.9^\circ\text{C} \sim +26.8^\circ\text{C}$ respectively. The differences ΔT of average temperature are 11.4°C and 6.0°C respectively.

The range of battery temperature is $12.8^\circ\text{C} \sim 50.8^\circ\text{C}$. To keep the battery performance, it should be kept in the range of $0^\circ\text{C} \sim 30^\circ\text{C}$, and the high temperature will break the battery. The reason of this high temperature is that the design of battery charge system is not sufficient. The avoidance way of this high temperature was found in flight operation. The power must be moderately consumed to some extent, and the charging current must be suppressed.

6. Estimation of Temperatures in Flight Mode

The estimation of temperatures in flight mode is carried out, and it is compared to real temperatures of flight data. The heat income is the sum of the solar radiation Q_s , the earth infrared emission Q_l , and the albedo Q_A . The each heat is expressed by the following equations.

$$\begin{aligned} Q_s &= AG_s \alpha \cos \theta_s, \quad Q_l = Aq_l \varepsilon F_e \cos \theta_e \\ Q_A &= AG_s a K_a \alpha F_e \cos \theta_e \end{aligned} \quad (2)$$

where A is the area of panel, α is the solar absorptivity,

Fig. 10. Simulated thermal history of flight mode.

Table 7. Comparison of real and estimated temperatures.
average temperature (degC)

	real (I)	estimate (II)	error (I - II)
top panel	13.4	-3.8	17.2
side panels	10.6	-6.0	16.6
bottom panel	11.2	6.7	4.5
center structure	17.2	13.7	3.5

and ε is the infrared emissivity. The θ_s, θ_e are the angles between the normal vector of panel and the solar or earth direction. When the angle is over 90° , it is fixed to 90° . The G_s is the direct solar flux, q_i is the earth infrared emission per unit area, a is the albedo rate. These values are changing in seasons, but $G_s = 1402 \text{ W/m}^2$, which is the average between Jan. 23, 2009 to Feb. 4, 2009, $q_i = 237 \text{ W/m}^2$, $a = 0.3$ are used in this analysis.

When the altitude of satellite is 668.7 km, the visual radius of earth ρ is 64.8° . Assuming that the satellite is flat plate, $K_a = 0.664 + 0.521 \rho - 0.203 \rho^2 = 0.994$. (see Ref.[5]). The view factor of earth is $F_e = \sin^2 \rho = 0.819$.

According to the power generation efficiency, the generation power by solar panels is 34.5 W in sunshine. Therefore, the generation heat by instruments Q_e is defined by 21.8 W under the assumption that the heat is constant both in 62-min sunshine and 36-min eclipse. In numerical analyses, it is assumed that this heat is generated in the center structure.

The radiation cooling of panels Q_{sp} is defined by the following equation.

$$Q_{sp} = \varepsilon A \sigma (T^4 - T_{sp}^4) \quad (3)$$

where T is the panel temperature, and T_{sp} is the 3-K space temperature.

Combining all the input and output heats mentioned above, the total heat Q is expressed by the following equation.

$$Q = Q_s + Q_l + Q_A + Q_e - Q_{sp} \quad (4)$$

The mathematical model of satellite is same as Sec. 3, and the heat transfer coefficients defined in Table 6 are used. The orbit is circular, and the altitude is 668.7 km, and the inclination is 98.0° . The angle between the

right ascension of descending node and the solar direction is 8.9° according to the orbital elements as of Feb. 4, 2009. The attitude of satellite is assumed to the spinning motion before the mast extension, and the angular velocity is 2.0 deg/s in each axis. The moment of inertia is 1.379, 1.375, and 1.118 kg/m^2 in x, y, z axis respectively, which are real measurements.

The result of temperature estimation for 3 periods is shown in Fig. 10. Also, the comparison of average temperature of panels between the estimates and the flight data is shown in Table 7. The error of estimation is $4.0 \sim 10.1^\circ\text{C}$ in external panels, and 2.5°C in center structure. The reason of error is the difference of measuring position in addition to the uncertainty of mathematical model. In flight data, the measuring position is the edge of panel because there are solar cells in the center of panels, but the center of panel is defined as measuring position in numerical estimation.

7. Conclusion

The microsatellite SPRITE-SAT developed by Tohoku University was launched in January 2009. Regarding the thermal system of SPRITE-SAT, the mathematical model, the parameter determination for the heat transfer coefficients, and the comparison between the estimate and real temperatures in flight mode are shown in this paper. The precision of thermal analysis using the simple 7-node model was solved. The estimate error of temperature is maximum 4.5°C in ground test, and 3.5 to 17.2°C in flight.

The estimation error was large for top and side panels because the weight of top and side panels with few of instruments is very light, which is 0.7 to 1.7 kg . On the one hand, the error was less than 5° for bottom and center panels with most of instruments.

This development method and the flight data are very important resources for the future development of microsatellites.

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