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ABSORBED HEAT-FLUX METHOD FOR GROUND SIMULATION OF ON-ORBIT THERMAL ENVIRONMENT OF SATELLITE

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ABSTRACT

An absorbed heat-flux method for ground simulation of on-orbit thermal environment of satellite is addressed in this paper. For satellite ground test, high vacuum and extremely low temperature of deep space are achieved by space simulation chamber, while spatial environmental heating is simulated by employing the absorbed heat-flux method. The methodology is explained in detail with test requirement and setup implemented on a satellite. Developed heat-load control system is presented with an adjusted PID-control logic and the system schematic realized is shown. A practical and successful application of the heat simulation method to KOMPSAT (Korea Multi-purpose Satellite) thermal environmental test is demonstrated, finally.

1. INTRODUCTION

On orbit, satellites are exposed necessarily to high vacuum and extremely low temperature of deep space that is 4 K ideally, as well as spatial-environmental heating. The principal forms of environmental heating on orbit are sunlight, both direct and reflected off (albedo) of the earth, and IR (infrared) energy emitted from the earth itself. Thermal control of a satellite on orbit is achieved by balancing the energy emitted by the satellite as IR radiation against the energy dissipated by internal electrical components plus the energy absorbed from the environment. For ground test of satellite, high vacuum and extremely low temperature of spatial environments are attained by space-simulation chamber equipped with liquid-nitrogen-cooled internal shrouds and cryo-pumps of high capability (Kim et al. 1996). However, realization of the on-orbit environmental heating is the very matter of difficulty and delicacy.

Simulation methods for the environmental heat loads are divided into two categories: absorbed-heat flux and incident-heat flux. The latter utilizes a solar simulator as an incident flux generator which emits a radiation spectrum similar to sun. It has been a main stream of the way in producing the environmental heating for a long time since artificial satellite showed up. However, there still exist a couple of issues in adopting the classical method. One is a matter of cost effectiveness and the other is ascribed to technical difficulties. The former results from that high complexities in addition to the very high price of solar simulator itself, are faced in interfacing the solar simulator to vacuum chamber; Moreover to get the enough size of solar beam and to protect reflection and re-radiation occurring from auxiliary equipment within the chamber, the chamber size also needs to be very large. IR radiation of earth still needs to be produced by any separate simulator, too. All of them are directly relevant to facility cost. Technical difficulties are as follows: 1) spectral matching is not exact, though the solar simulation is most realistic in generating solar radiation; 2) in order to produce any localized heating, satellite positioned inside the chamber needs to be direction-controlled because the environmental heat loads are inevitably different from side to side during satellite orbiting. This means that an attitude-control mechanism of satellite must be incorporated.

A replacement of the incident flux method is the absorbed heat-flux method of low cost and low complexity the methodological philosophy of which is described in this paper. In this method test-only heaters are directly affixed to satellite. It is a deficient aspect of the absorbed flux method that the heaters shall be taken off after the test and satellite refurbishment must be performed to keep away from any possibility of contamination which could be caused by heater adhesives. When affixing the heaters on satellite there needs a fairly delicate workmanship, too. After the heaters are installed there should not be any chasms or room underneath the heater strip: Chasms can cause a highly localized hot spot during heating which may result in the breakage of the heater-circuit conductors.

Heat flux to be imposed on heater is determined by analytical calculation and it is managed by a heat-loads control system in practical test. A strong dependency on the analysis in determining the intensity and distribution of environmental heat flux is another drawback of this technique. The analysis needs detailed informations on the orbital parameters such as orbit altitude, inclination, sun day angle, apogee, and perigee, etc. in addition to orbiting attitude and surface radiative properties of satellite. With those parameters and properties TRASYS (NASA 1988), which is a well-known computer program for calculating the thermal radiation environment for a satellite in orbit, produces the radiation conductors and total heating as a function of time or averaged. Once the environmental heat loads are computed by the analysis program the heating is generated by electrical power through strip heaters attached to satellite radiator zones where radiative heat exchange between satellite and spatial environment is primarily active. Heat-loads control system on which an adjusted PID (proportional-integral-differential) control logic was implemented manages the electrical power required. One thing of remark at this time is that an ideal PID routine, by itself can not be properly applied to satellite thermal test, though it is a predominant feedback controller in the control dynamics. The reasons are as follows: 1) Proportional, integral, and differential constant which are essential for feedback routine can not be determined prior to and even during the flight-model satellite test because pre-test of satellite is not possible and through the test temperature deviating from its acceptance level is inevitable for determining the constants but satellite does not allow any kinds of aging caused by the out-of-limit trial; 2) Differently from the linear and isolated-mass system where a relatively simplified modeling with trivial system properties (e.g., mass moment of inertia) enables the momentum behavior predictable, solution of nonlinear partial differential equation is essential for thermal behavior prediction in satellite system which is of nonlinear and non-isolated thermal features coming from the inherent heat transfer mechanism. But this seems not feasible from the viewpoint of practicality and cost effectiveness of test. Thus there remains finding out an optimal controller or PID alternative for the test employing absorbed heat-flux method.

On ground test exists a simulation limitation relevant with surface radiative properties of satellite: A hot-case condition of satellite often can not be completely simulated, because thermally-critical surfaces, e.g., SSM (second surface mirror) as a radiator and MLI (multi layer insulation) blanket as a typical radiation insulator, have beginning-of-life (BOL) solar absorptivity while ground-testing but the hot case are practically dominated by end-of-life (EOL) properties in orbit (Gilmore 1994). This test-only deficiency is covered by thermal analysis for the full knowledge on the satellite thermal characteristics in the operational orbit. This is a common shortage the absorbed flux and incident flux method share with.

Objectives of this article is to address the absorbed flux method employed practically on the satellite on-orbit thermal simulation with its system design implementation. Thermal analysis is out of scope in this work and its outcome which was used for the adoption of heating power is simply mentioned. A practical application to KOMPSAT (Korea Multi-purpose Satellite) is demonstrated, additionally.

2. GROUND SIMULATION OF SATELLITE ON-ORBIT ENVIRONMENT

2.1 Satellite Test Requirements

To verify the design and ensure successful operational use in the space environment satellites are to be subjected to extensive ground testing prior to the launch. An US Air Force military standard establishes a uniform set of definitions, environmental criteria, and test methods for military space vehicles, subsystems, and components (MIL-STD-1540B 1982). This also states that the test requirements should be tailored to the specific space program, considering design complexity, state of the art, mission criticality, and acceptable risk. Acceptance tests are required formal tests that are conducted to demonstrate performance to specified requirements and to act as quality-control screens to detect deficiencies in workmanship, materials, and quality. Three tests described in MIL-STD-1540B associated with space vehicle level are thermal cycling, thermal vacuum, and thermal balance. The thermal cycling tests at system level, as in the component level, are primarily environmental screens to expose design, workmanship, material, and processing defects. This test is optional at the spacecraft level and often replaced by thermal vacuum test unless the spacecraft to be tested is the first kind in its development.

The objective of thermal vacuum test is to expose spacecraft to environments which are non-destructive in nature, but yet able to provide assurance of detecting any deficiencies. This test is constituted primarily of system-level functional performance tests between and at temperature extremes. Emphasis is put on component and subsystem interaction and interfaces, and on end-to-end electrical system performance.

Thermal balance test is comprised of dedicated thermal tests conducted during thermal vacuum test to verify the thermal analytic models and the thermal design by way of the functional demonstration of thermal control hardware and software. A successful test and subsequent model correlation establishes the ability of the thermal control subsystem to maintain all payloads and bus equipment within specified temperature limits for all mission phases. Several mission phases such as launch, deployment, sun pointing, science, and safe haven mode, etc. are commonly involved in satellite operation with their specific thermal environments and electrical power configurations. As time of thermal facilities is expensive, a judicious choice of test cases related to mission phases and environments should be made for ground test.

KOMPSAT was inherited from TOMS-EP (Total Ozone Mapping Spectroscope-Earth Probe) and STEP (Space Test Experiment Platform), etc.: i.e., KOMPSAT is not the first kind of satellite development. As mentioned before, test requirement of MIL-STD-1540B are allowed for tailoring if the design complexity of satellite could be decreased by its heritages. KOMPSAT test requirement in its details was described based upon the tailoring philosophy: Four times of thermal vacuum cycles in which one cold and hot balance test is included, are required for KOMPSAT as MIL-STD-1540B requires at least four cycles at vehicle-level acceptance test. Test level of thermal acceptance strictly followed MIL-STD-1540B except for payload sensors which have their own acceptance criteria for specific mission phases.

KOMPSAT was subjected to three times of thermal vacuum cycling followed by one temperature cycle during which CPT's (Comprehensive Electrical Performance Test) and thermal balance tests were performed at the cold and hot extremes: i.e., at the end of three cyclings cold CPT, cold thermal balance, hot CPT, and hot balance test followed and the test ended with returning to ambient condition. Thermal balance test was performed with two extreme cases of environments and satellite configurations of cold BOL Safe Haven Mode and hot EOL Science Mode. Hot case falls under high solar energy absorption at EOL of high absorptivity and such high, yet realistic, levels of equipment usage as an active operation of payloads. Minimal equipment usage, bus voltage, and solar heating such as safe haven mode at BOL are the conditions for cold case.

2.2 Test Setup and Heat Loads Control Criteria

KOMPSAT is a sun-synchronous LEO (Low Earth Orbit) satellite built by the joint-development program of Korea Aerospace Research Institute (KARI) and TRW (Redondo Beach, California, USA). Its flight model has been integrated at KARI and the functional and environmental test have been completed in mid 1999. Aboard a Taurus solid rocket which built by OSC (Orbital Science Corporation, USA), it is supposed to launch from Vandenberg Air Force Base in California, USA in late 1999, and will be placed into its circular operational orbit of 685 km altitude and 98 deg. inclination to facilitate cartography that is its primary mission, ocean color imaging, and ionosphere measurement.

For ground thermal test the satellite is put into a space-simulation chamber with a cryogenic shroud. The shroud is chilled upto the temperature less than -170 $^{\circ}C$ with liquid nitrogen to simulate the cold sink of outer space. The condition of high vacuum of 10^{-5} Torr or less is achieved by pumping system which are comprised of cryo-pumps, mechanical pumps, and turbo-molecular pumps in series and/or in parallel. Figure 1 shows KOMPSAT spacecraft installed on the mounting fixture of space-simulation chamber whose available diameter/length is 3.6 $\times$ 3.0 m. In order to keep the spacecraft away from unwanted conductive heat sink which can be caused by the fixture, a thermal insulator is inserted in between.

The satellite bus structure consists of a payload module, an electronic equipments module, a propulsion module, and a launch vehicle adapter with two solar array wings. The bus enclosure panels are completely covered with MLI only except for radiator regions. SSM's were adhered to

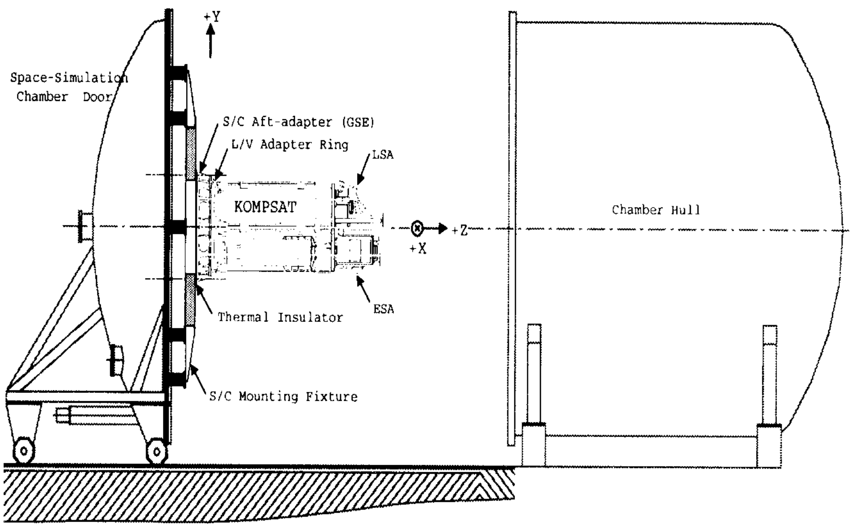

Figure 1. Satellite installed on Space-simulation Chamber for Ground Test

the panel outside as a representative radiator. The MLI insulates the bus from heat flows either into or out of the bus structure, while the SSM allows thermal radiation to space. The SSM's reflect most of the incident solar radiation when they are exposed to the sun and albedo heat loads. SSM of 6 mil thickness has the radiative properties that are IR emissivity, $\epsilon = 0.78$, and solar absorptivity, $\alpha = 0.07$ (BOL) to 0.15 (EOL). Environmental heat loads are attained with strip heaters affixed directly to bus panels which are divided into 17 radiator sub-zones.

Figure 2 shows layout of the bus panels. There can be found that six heater circuits (H1, H3, H6, H9, H12, and H15) were applied to the radiator area of six payload panels, six ones (H2, H4, H7, H10, H13, and H16) to upper equipments module, and five ones (H5, H8, H11, H14, and H17) to lower equipments module. Each sub-zone exchanges radiative heat with the other facing sub-zones of view factor greater than zero and with the electronic equipments positioned inside the panels. On the same panel the sub-zones essentially exchange heat conductively with each other. These conductive and radiative coupling among the heating zones become the primary cause of difficulty in the precise temperature control. This will be discussed in subsequent section.

Table 1 lists simulation heaters, their locations, relevant thermocouples, and the heater control criteria per each test phase practiced through KOMPSAT thermal test. The type, 'T' (copper/constantan) thermocouples are located on the heater or nearby. They are used for supplying temperatures for controller to achieve the target temperature and/or for watchdog in the direct power (or heat) control mode. For providing against single thermocouple fault, at least two control thermocouples are associated to each heater. One is set just over the heater conductors and the other is positioned somewhat

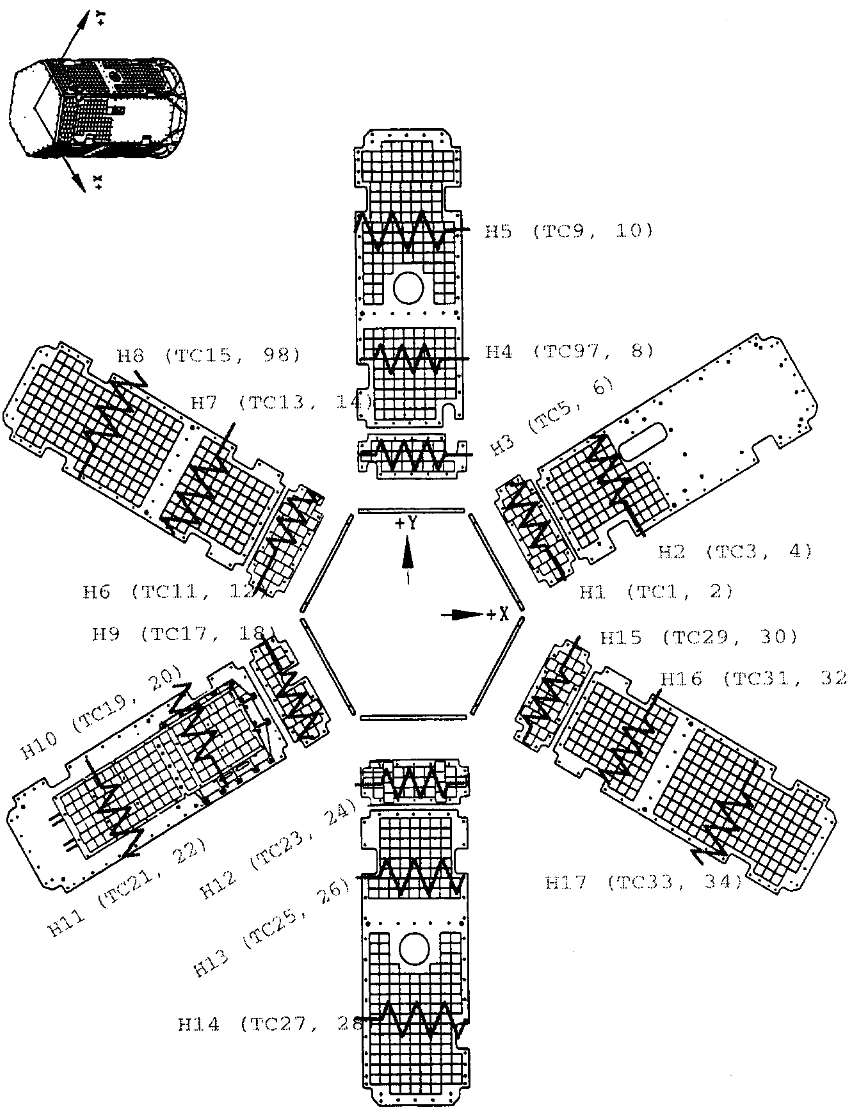

Figure 2. Layout of Satellite Bus Enclosure Panels

Table 1. Control Criteria of Heat Loads (Watts) per Each Test Phase

Test Phases / Flux Control Heaters			Chamber Evac. & Fill Cold Wall	Primary Heater Checkout	Transition to Hot	Primary Heater Checkout	Transition to Hot	Redundent Heater Checkout	Transition to Hot	Redundent Heater Checkout	Cold CPT	Cold T/B	Transition to Hot	Hot CPT	Hot T/B	Prop. Sys. Leak Test	Purge Cold Wall & Return to Ambient
Heater Ckt.# (Pri.)	Location	Control TC's	A	B	C	D	E	F	G	H	I	J	K	L	M	N	O
H.01	+X+Y Payload Radiator	1,2	20±3℃	Varied	Varied	Varied	Varied	Varied	Varied	Varied	Varied	9.1	Varied	Varied	9.0	20±3℃	20±3℃
H.02	+X+Y Upper Equip. Rad.	3,4	20±3℃	Varied	Varied	Varied	Varied	Varied	Varied	Varied	Varied	3.8	Varied	Varied	6.3	20±3℃	20±3℃
H.03	+Y Payload Rad.	5,6	20±3℃	Varied	Varied	Varied	Varied	Varied	Varied	Varied	Varied	8.7	Varied	Varied	7.4	20±3℃	20±3℃
H.04	+Y Upper Equip. Rad.	97,8	20±3℃	Varied	Varied	Varied	Varied	Varied	Varied	Varied	Varied	3.8	Varied	Varied	5.7	20±3℃	20±3℃
H.05	+Y Lower Equip. Rad.	9,10	20±3℃	Varied	Varied	Varied	Varied	Varied	Varied	Varied	Varied	5.2	Varied	Varied	9.6	20±3℃	20±3℃
H.06	-X+Y Payload Rad.	11,12	20±3℃	Varied	Varied	Varied	Varied	Varied	Varied	Varied	Varied	9.2	Varied	Varied	9.1	20±3℃	20±3℃
H.07	-X+Y Upper Equip. Rad.	13,14	20±3℃	Varied	Varied	Varied	Varied	Varied	Varied	Varied	Varied	2.6	Varied	Varied	4.2	20±3℃	20±3℃
H.08	-X+Y Lower Equip. Rad.	15,98	20±3℃	Varied	Varied	Varied	Varied	Varied	Varied	Varied	Varied	7.5	Varied	Varied	12.3	20±3℃	20±3℃
H.09	-X-Y Payload Rad.	17,18	20±3℃	Varied	Varied	Varied	Varied	Varied	Varied	Varied	Varied	8.3	Varied	Varied	6.8	20±3℃	20±3℃
H.10	-X-Y Battery Rad., Upper	19,20	10±3℃	Varied	Varied	Varied	Varied	Varied	Varied	Varied	Varied	4.5	Varied	Varied	7.3	10±3℃	10±3℃
H.11	-X-Y Battery Rad., Lower	21,22	10±3℃	Varied	Varied	Varied	Varied	Varied	Varied	Varied	Varied	6.0	Varied	Varied	9.5	10±3℃	10±3℃
H.12	-Y Payload Rad.	23,24	20±3℃	Varied	Varied	Varied	Varied	Varied	Varied	Varied	Varied	7.9	Varied	Varied	3.9	20±3℃	20±3℃
H.13	-Y Upper Equip. Rad.	25,26	20±3℃	Varied	Varied	Varied	Varied	Varied	Varied	Varied	Varied	2.2	Varied	Varied	2.5	20±3℃	20±3℃
H.14	-Y Lower Equip. Rad.	27,28	20±3℃	Varied	Varied	Varied	Varied	Varied	Varied	Varied	Varied	4.4	Varied	Varied	6.8	20±3℃	20±3℃
H.15	+X-Y Payload Rad.	29,30	20±3℃	Varied	Varied	Varied	Varied	Varied	Varied	Varied	Varied	8.5	Varied	Varied	6.8	20±3℃	20±3℃
H.16	+X-Y Upper Equip. Rad.	31,32	20±3℃	Varied	Varied	Varied	Varied	Varied	Varied	Varied	Varied	3.8	Varied	Varied	5.7	20±3℃	20±3℃
H.17	+X-Y Lower Equip. Rad.	33,34	20±3℃	Varied	Varied	Varied	Varied	Varied	Varied	Varied	Varied	6.5	Varied	Varied	10	20±3℃	20±3℃

off of the conductors. The latter is selected as the primary control sensor in the test because the former does give temperature reaction faster than the latter. Accounting for thermal inertia too fast temperature variation does not help the stable heat control. The control or watchdog sensors shall be located in this way. This fact is rationalized as follows: Heaters affixed to radiators take the role of supplying heating power rather than attaining target temperature and moreover radiator panels have no temperature limits. In the power control mode final temperature can not be predicted in advance. The stabilized temperature will be dependent upon heat capacity of the material on which heater is attached and upon the rate of heat exchange with its neighbor. In this case the only thing to be cautious is to protect the heaters against burnout and conductor breakage which can be caused by localized high temperature. The highest occurs necessarily on the heater itself. This is why the sensor are set on the heaters.

Exceptions are for the heaters installed directly on electronic equipment of satellite. They are purposed to simulate heat dissipation generated by electronics, though they are not listed in Table 1. In this case one sensor of the two is located away from the heater. For the flight model satellite, temperatures exceeding the acceptance limit are never allowed in any circumstances. Therefore a sensor over the heater is used for monitoring the localized hottest temperature and the other away from the heater is for observing the more representative temperature of electronic box. It could be practically found that temperature difference between both sensors was less than 3 $°C$ even in worst case. Some of the electronics located outside of bus have thermal control surfaces of silverized teflon ($\epsilon = 0.78$, $\alpha = 0.13$ to 0.23). Their environmental heating should have been also made like as in the

SSM radiator zones.

When applying the heaters, a precaution should be taken against any deterioration of the radiating surface properties. In reality the possibility of property change is additional defect of the present method. Once the radiator area is covered by test heaters the apparent surface shall have as similar IR emissivity to the original surface as possible. To realize this, heaters surface of which is gold-etched were selected. Their surface IR emissivity is 0.78 (Gilmore 1994). Differently from the emissivity, absorption coefficient of the heater surface needs not be matched to the original one because solar radiation is replaced and simulated by the absorbed heat loads in this method.

Starting from the phase A of chamber evacuation and shroud chilling by liquid nitrogen the satellite heads for the first cold extreme (phase B in Table 1). Once the cold extreme condition is achieved the environmental heating is adjusted to achieve the hot extreme condition of satellite (phase C). After this cycle repeats twice more (phase D & E and phase F & G) satellite turns over toward another cold extreme for CPT and cold balance test (phase H, I, and J). During the cold transition phases (B, D, F, and H) On/Off function of flight heaters dedicated to thermal control on orbit are completely checked out. The CPT and thermal balance test are performed again at the hot extreme (phase L and M) and the test ends up with returning to ground environment (phase O). During the cold and hot CPT the satellite undergoes the change of electrical configuration of its equipments, which causes change of heat dissipation. Those changes can make satellite deviate unfavorably from its temperature limit because it was being tested near extreme temperatures. Therefore continuous adjusting action of environmental heating should be taken all through the CPT's.

The simulation powers to be assigned to radiators can be obtained from thermal analysis of satellite in its operational orbit. The orbit-averaged values of cold and hot balance case are direct outcome of TRASYS run. In practicing thermal balance, these values (H1 to H17 at the phases of J and M in Table 1) are to be kept strictly. On the other hand those of vacuum cycling phases are not so stringent. In other words the heating values can be manipulated to make the spacecraft reach fast to hot or cold extremes during the cycling. From the phase B to phase I, and phase K to phase L, the heating values have been generated several times (Kim & Choi 1998).

2.3 Development of Heat Loads Control System

Environmental heat loads are generated by absorbed-heat flux control system. Its main functions are as follows: 1) real-time supplying of regulated DC electrical power to the heaters; 2) real-time acquisition of the thermocouple-recorded temperatures which are used to PID-control and to watch over whether or not the heater is being operated within the temperature limits specified. The system is comprised of 50 linear-programmable DC power supplies, fuse rack, switching unit which selects remote or manual control of power supply, shunt box which accurately measures currents being impressed, control unit, and data acquisition unit. Number of power supplies should be more than number of heaters to be controlled because a power supply can manage only a heat load variation with time.

Figure 3 shows a schematic configuration of the system hardware developed. Control, acquisition, and interface units correspond respectively to VXI HPIB command module, 5.5-digit multimeter & two 64-channel thermocouple relay muxes, two 60-channel data acquisition/switch units, and two 60-channel DAC's (digital to analog converter) in the system materialized (Kim & Cho 1998). The

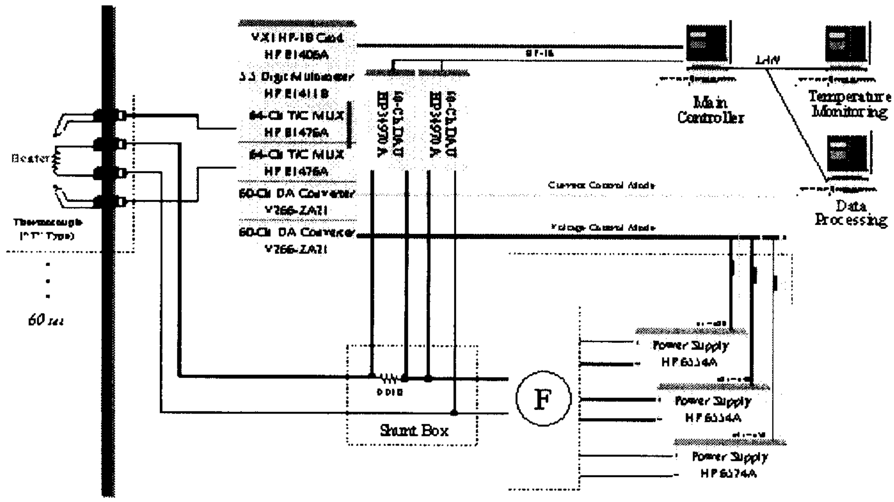

Figure 3. Schematic of Heat-Loads Control System

system was very compact and stable by the employment of VXI bus architecture and very unique in that nowhere could be found the similar system (hardware and software) which automatically controls 50 power supplies.

During the test, overheating or under-heating of the satellite may damage its electronic components which are located near the region heat-controlled. Therefore, design and operation of the control system shall be based on keeping, first of all, the safety of the satellite under test and then on realizing the thermal environment as closely as possible. Delicately predicted maximum allowable powers and various protection functions, whose considerations are imperative for the safety of spacecraft, were implemented on the control system design and especially in setting up the control software. Fusing system is typical of the hardware safety module.

In designing the control software, PID controller was implemented as a baseline for power and temperature feedback control. Equation (1) shows PID controller expressed in the time domain.

$$U(t) = K_p e(t) + K_i \int e(t)dt + K_d \frac{de(t)}{dt} \tag{1}$$

where U(t) and e(t) are controller output and error (error = set-point - measurement); K_p, K_i, and K_d represent proportional, integral, and derivative parameters, respectively. The output of PID controllers will change in response to a change in measurement or set pont. It is very well known that with only a proportional controller (the first term of righthand side in the above equation), an involvement of offset (deviation from set-point) is inevitable. Integral action, the second term of righthand side in the equation (1), was included in controllers to eliminate this offset. Integral action gives the controller a large gain that results in eliminating the offset, but induces an additional phase lag. Derivative action, the last term of righthand side in the equation (1) adds phase lead and was used to compensate for the lag introduced by the integral action. With the derivative action, the controller output is proportional to the rate of change of the measurement or error. Derivative takes action to inhibit more rapid changes of the measurement than proportional action. When a load or set-point change occurs, the derivative action, however, can cause the controller gain to move the wrong way when the measurement gets near the set-points.

In the application of the PID controller to practical satellite, it is impossible to set up optimal control parameter of each heater prior to test. The reasons are as follows:

(1) Each heater shall have its own proportional bandwidth, integral reset time, and derivative action since each zone to be controlled has respective heat capacity, mass, and thermal conductance, etc. This variety needs tuning of the control constants by the number of (3 x number of heaters); If we can test the satellite prior to its acceptance test, those parameters might be found through an exhaustive control experience, but pre-test of the satellite is never allowed.

(2) Heating zones to be controlled are necessarily coupled with each other neighboring zone in satellite test. This coupling makes each zone thermally non-isolated system. Moreover the heat transfer process is non-linear due to its conduction and radiation nature. There is no proven method to determine the control parameters in the nonlinear and non-isolated system. Although there exist a number of recommendation for the proper tuning of PID controller, they are confined to an isolated or linear system (Bennett 1991). For the present application where each heater affects to others in nonlinear manner, such ideal methods can not be applicable any more.

There are three modes for heater control, i.e., 1) Wattage control mode, 2) temperature control mode, and 3) temperature difference (ΔT) control mode. The first is for supplying target heat flux to the zones to be controlled. During the Wattage control mode if the zone being controlled reaches a temperature limit, the control mode is switched to the temperature watchdog mode for the safety of the zone or equipment box. Thus the temperature control mode is used not only for attaining target temperature but also for bounding the temperature within the limit. The third,Δ T control mode is for controlling net conductive heat flow: With ΔT set to zero, unwanted heat flows coming from or entering into satellite through test appendages can be blocked off. In fact, the Wattage control is performed through currents adjust with known heater resistance. In this mode the required power could be obtained only with PD controller. For the adequate heater control in temperature-relevant modes (temperature, watchdog, and Δ T), there was introduced a new dynamic control variable,

$$\Delta T_{lag} = \Delta T_S \times (|e(t)| + 2)/100 \tag{2}$$

where ΔT_lag and ΔT_s represent control lag variable and static temperature constants, respectively. The static temperature constant has a preset value of 0.5 and can be decreased or increased for the acceleration or deceleration of control speed. The control lag constant has the feature that it decreases as the error goes down and vice versa: e.g., once target temperature is exactly achieved becomes 0.01. It was utilized in practical controller as follows: With one set of predetermined PID values (Proportional bandwidth = 5 $\sim$ 20 %, Integral reset = 0.05 $\sim$ 0.1 sec^{-1}, and Derivative action = 20 $\sim$ 60 sec) the temperature control starts; At every feedback action the error is compared to the dynamic control lag. If the error is less than the lag constant no feedback action is taken and controller waits until the error gets greater than the lag constant. This artificial control delay, by lagging the active PID control as the measured value approaches the target temperature, has the effect for the compensation of thermal inertia through allowing the controlled zone to be sufficiently affected by prior control action. With this dynamic control variable all the control channels could be controlled with the maximum(worst) temperature overshoot less than 0.3 °. Accounting for the thermocouple accuracy, measurement errors which can be caused by environmental noise, and test requirement which says the control tolerance of ± 3 °, the maximum overshoot was allowable without any tease.

Functional verification of the flux control system has been repeatedly performed under ambient and under vacuum chamber environment. The verification test which was as similar as satellite thermal cycling and balancing test has been also performed. Test mockup, which was of stainless steel structure with a configuration of various mass and surface area distribution, was initially maintained at 20 °C and then lowered to -10 °C. It was lifted up 43 °C again and the similar cycling repeated. The capability gaining target temperature and target heat loads was maximized under the vacuum and cold environment, which is compared to the ambient or hot environment, due to the absence of buoyancy effect and by the quick removal of residual heat contained in the region of out-of-direct heat control (Kim & Cho 1999a).

As such, the hardware and software of control system has been thoroughly design-verified through its functional demonstration and thus proven as a proper simulator for the spacecraft environmental heat-loads.

3. PRACTICAL APPLICATION TO SATELLITE GROUND TEST

The test heaters and instruments worked satisfactorily all through the on-orbit environmental test of satellite. Heat loads could be generated and controlled, just as wanted, by the absorbed-heat flux control system. The test has started on April 9 and ended on April 20, 1999.

Figure 4(a) depicts history of the heating power managed from the beginning to the end of thermal test. Heaters applied to upper region of equipment enclosure panels are selectively plotted. Test timeline of three vacuum cyclings, cold CPT, cold balance, hot CPT, and hot balance test followed by returning to ambient condition can be macroscopically found in the figure and an exhaustive control activity through the vacuum cyclings and CPT's is also found there. Thermal hardware functions such as thermostat and heater actuation as well as thermal control software of satellite were verified during the vacuum cycling. Temperature extremes of vacuum cycling are based upon worse-case analytic prediction for at least one component in each zone. Typically the satellite is divided into manageable zones and the test temperature limits are specified for each zone: For the present satellite, platforms where electronic boxes are laid together are typical of those zones. A variety of components, often tested to different temperature extremes during component acceptance, must be accommodated during system-level thermal testing. The approach taken during vacuum cycling is to drive as many components as possible to their acceptance temperature extremes, with the constraints that any component should not exceed its component-level extremes to avoid over-stressing. These have caused such exhaustive control activities during the vacuum cycles

Through CPT's the end-to-end electrical test has been comprehensively performed. Likely as vacuum cycling, the environmental heat loads should have been generated many times to make as many components as possible undergo the cold or hot extremes during the cold and hot CPT. While, as mentioned before, the heat loads during cold and hot balance test should have been kept strictly following the values given in the Table 1. Their constant heating can be observed around 15 April

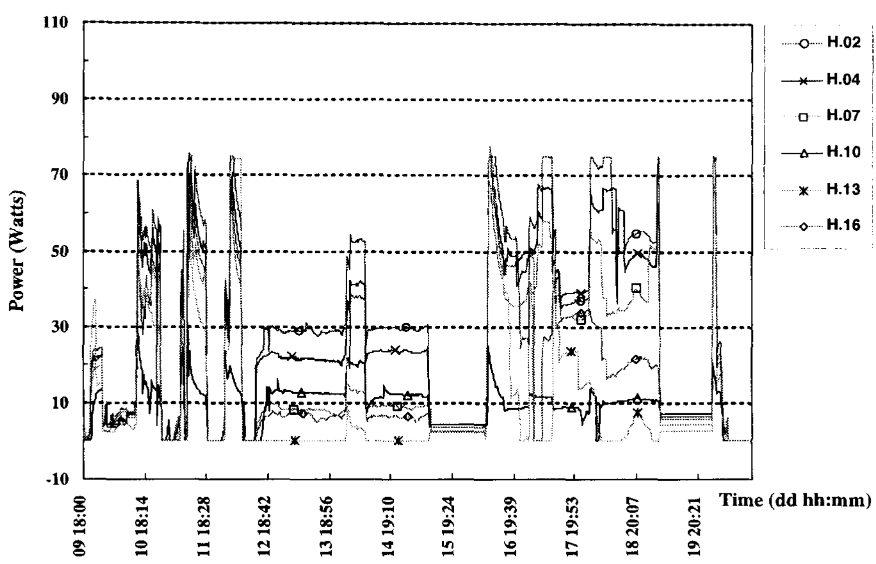

Figure 4(a). History of Heating Power

for cold and 19 April for hot case in Fig. 4(a).

Temperatures of radiator panels corresponding to the heat loads variation are shown in Fig. 4(b). Each thermocouple, TC.04, TC.08, TC.14, TC.20, TC.26, and TC.32 are associated in Fig. 4(a) to the heaters, H.02, H.04, H.07, H.10, H.13, and H.16, respectively. A tendency approaching to constant temperatures can be found during cold and hot balance period in the Fig. 4(b). Ending state of the balances indicates that the energy being emitted by satellite as IR radiation has been balanced to the sum of the energy being dissipated by satellite internal components and the energy being absorbed by environmental heating. Thermal equilibrium or constantly cyclic temperatures of satellite electronic components could be obtained at the end state of cold and hot balance.

During the balance period the satellite, with its specific electrical configuration, was to be exposed to thermal environment expected on orbit. The balanced state was however, just an orbit-averaged one because the environmental heating was also put from the orbit-averaged prediction. In real operational orbit satellites experience a transient thermal environment varying with time. But the transient simulation is so difficult and costly on ground test that the averaged condition is mostly simulated with orbit-averaged heat loads. Nevertheless, such balanced temperatures as obtained from the test can supply very useful data to post-test model correlation and be employed for macroscopic evaluation of satellite thermal design. Conclusively, from the presented thermal tests, a number of suggestions to be reflected on model correlation were drawn and fed back to the thermal analysis engineering.

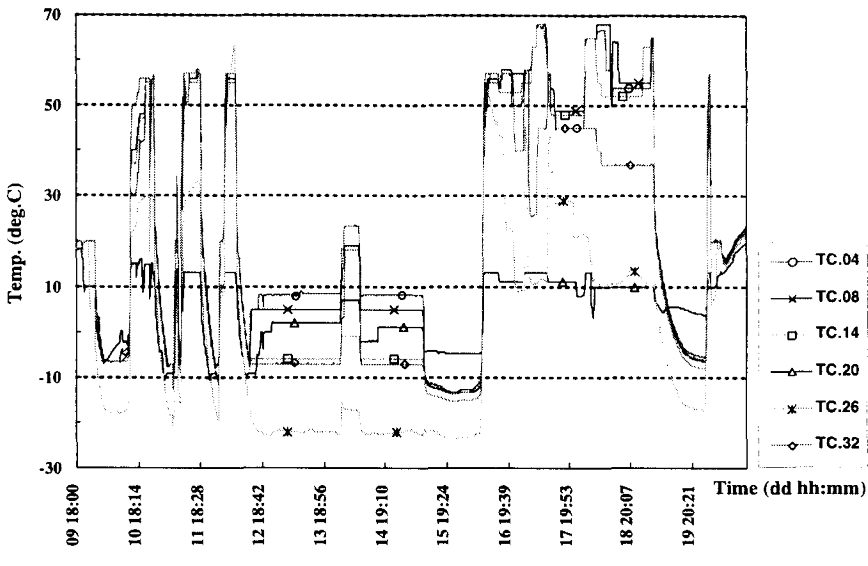

Figure 4(b). Radiator Temperatures Corresponding to Heat Loads

4. SUMMARY AND CONCLUSION

Absorbed heat-flux method for the ground simulation of on-orbit thermal environment of satellite was described in detail. Reviewed and discussed are contents as follows:

1) Methodologies for on-orbit environment simulation were introduced.

2) Thermal test requirement and setup implemented on KOMPSAT were presented.

3) Schematic of the absorbed heat-flux control system was depicted and its detailed control logic was explained.

4) Practical application of the absorbed flux method to satellite was demonstrated: Satellite thermal test has been successfully performed under the on-orbit environment simulated by employing the absorbed heat-flux method.

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