General

JERG-2-310

DESIGN STANDARD

SPACECRAFT
THERMAL CONTROL SYSTEM

July 8 ,2009

Japan Aerospace Exploration Agency

This is an English translation of JREG-2-310. Whenever there is anything ambiguous in this document, the original document (the Japanese version) shall be used to clarify the intent of the requirement.

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CONTENTS

1 General rules

1.1 Purpose

The purpose of this document is to apply a design standard to the thermal control design of the spacecraft (satellite/space probe) developed by Japan Aerospace Exploration Agency (hereafter called “JAXA”). Moreover, this standard is defined in anticipation of being applied not only to JAXA, but also to other organizations and companies.

1.2 Scope

This standard defines requirements for the thermal design, thermal design analysis and thermal tests of the spacecraft system and its components (hereafter called “spacecraft etc.”) and requirements for the thermal interface between spacecraft and launch vehicles etc.

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Additionally, the subject of this standard is the general thermal range, between approximately 100K and 500K. For temperatures outside of this range, i.e. temperatures higher than 500K or lower than 100K, this standard is not required, but can be used just for reference.

2 Related documents

2.1 Applicable documents

The following normative documents contain provisions which, through reference in this text, constitute provisions of this standard.

(1) JERG-2-311 Design Standard to prevent peeling of MLI
(2) JERG-2-211 Design Standard for electrical charge/discharge
(3) JERG-2-002 Standard of General Tests for Spacecraft
(4) JERG-2-141 Standard of Space Environment
(5) JERG-2-212 Design Standard of Wire Derating
(6) JMR-010 Standard for Contamination Control

2.2 Foreign standard etc.

(1) ECSS-E-30 Mechanical Part 1A : Thermal Control
(2) NASA GSFC GETS (ELV)-1 General Environmental Test Specification for Spacecraft and Component
(3) ISO 14644-1 Clean rooms and associated controlled environments
Part1 : Classification of airborne particulates
(4) ISO 14952-2 Space systems – surface cleanliness of fluid systems Part2 : cleanliness levels
(5) NASA TM X-64627 “Space and planetary environment criteria guidelines for use in space vehicle development 1971 Revision”

3 Definition of terms and abbreviation

3.1 Definition of terms

The terms used in this design standard are defined below.

3.1.1 Temperature

(1) Allowable temperature range

The allowable temperature range shows the resistance to the environmental temperature of a component, and classified under the operating phase (maintaining performance and function) of equipment, the non-operating phase (preservation) and turn-on. See section 5.2 for more detail.

(2) Temperature reference point

The temperature reference point refers to the allowable temperature range for a component on a piece of equipment. Generally the position of the temperature reference point is important in the context of interface. The overall temperature level may be determined by the temperature reference point, or in some cases, a selected thermally critical point . See section 5.3 for more detail.

(3) Design margin

This refers to the margin of safety in the area of thermal design required to keep the spacecraft in orbit, certainly within the allowable temperature range as defined in (1) above, and with consideration of any unpredictable in-orbit phenomenon. See section 5.2 for more detail.

(4) Prediction error

Prediction error refers to an error of estimated temperature caused by an error in mathematical modeling and/or thermal analytical tool, and included in the measuring system and thermal input system for the in-orbit temperature prediction. See section 5.2 for more detail.

(5) Range of test temperature

This refers to the temperature range loaded onto the model at the qualification test, acceptance test or PFT (Proto Flight Test).

(6) Design prediction temperature range

This refers to the predicted temperature range as the result of thermal analysis. See section 5.2 for more detail.

3.1.2 Construction equipment for spacecraft system

The components of a spacecraft system comprise part of a larger system, to include the combination of parts, devices and structures, and perform a function independently within the whole operation of a larger system. Further, the components of spacecraft system are classified as being either internal onboard equipment or exposed onboard equipment, as follows:

(a) Internal onboard equipment
Refers to equipment mounted within the spacecraft and not exposed to outer space.
(b) Exposed onboard equipment
Refers to equipment either mounted outside the spacecraft, or when a portion of the equipment is exposed to the outer space and/or installed in an opening.

3.1.3 Tests

The terms concerned with test, used in this design standard, is based on the definition mentioned on “JERG-2-002 Standard of General Tests for Spacecraft” of 2.1 (3). The outline is shown below.

(1) Development Phase Tests

<1> qualification test

The test to prove that the design and production procedure is suitable to obtain the results that
the hardware (and software) satisfy the requirements of the specification.

<2> acceptance test

The test to determine acceptance of goods for supply as flight hardware. The purpose of the acceptance test is to detect any faults in the materials or manufacturing of products which were produced according to the qualified design.

<3> PFT (Proto Flight Test)

Following the qualification and acceptance tests, the Proto Flight Test determines the quality of design and production procedure, as well as the possibility of acceptance as the flight model.

(2) Verification of thermal design and the proof of function and performance tests

<1> thermal balance test

The thermal balance test includes placing the space system or components into a space chamber and simulating the thermal vacuum environment found in orbit. This test confirms and verifies the thermal mathematical model of the component, the validity of the thermal control design and the performance of the thermal control hardware, based on the acquired thermal data.

In this test, the confirmation and verification of the external thermal input conditions is included, providing its own thermal vacuum test used for the development of the thermal control system of the spacecraft.

<2> thermal vacuum test

In a more narrow sense, the thermal vacuum test, where the spacecraft or component is placed into a space chamber and exposed to a thermal environment as if it were in orbit or even under more harsh conditions, is designed to prove the spacecraft or component works normally and satisfies the requirement of function and performance. In a wider sense, the thermal vacuum test is included as part of thermal balance test as described above. The general, wider sense of the term, which includes both the thermal balance test and the thermal vacuum test, is the standard term unless the thermal vacuum test is specified.

3.1.4 Thermal control technology

(1) Thermal control device

The constituent components of thermal control hardware (element, parts etc.) are defined as such until its function is lost when it is further subdivided.

(2) Passive thermal control method

The passive thermal control method is the procedure to control the temperature of the component within the specified range by adjusting the paths of conduction and radiation, and by the selection of geometric form of each surface and thermo-physical property of the spacecraft. The typical thermal control elements used for the passive thermal control method include multi-layer insulation, thermal coating, heat sink and thermal doubler (thermal diffusion) etc. Furthermore, the fixed conductance heat pipe corresponds to this type of thermal control method.

(3) Active thermal control method

The active thermal control method is the procedure to control the temperature using mechanical mobile components or fluid, using electric energy from a heater, changing the component's thermo-physical property, or utilizing another technology to change/control the temperature. The heater and thermal louver is the typical thermal control element used in the active thermal control method. In addition, the variable conductance heat pipe corresponds to this type of thermal control method.

(4) Thermal mathematical model

The thermal mathematical model refers to the mathematical representation of the thermal characteristics of the spacecraft or components, which is used for temperature prediction within the thermal environmental conditions. It is also sometimes called thermal analysis model.

3.2 Definition of abbreviation

The abbreviation used in this design standard is shown as follows.

AT=Acceptance test
CVCM=Collected Volatile Condensable Materials
EM=Engineering Model
FM=Flight Model
ICD=Interface Control Drawing

MLI=Multi Layer Insulation
PFM=Proto-flight Model
OSR=Optical Solar Reflector
PM=Prototype Model
QT=Qualification Test
SLI=Single Layer Insulation
TML=Total Mass Loss

4 General requirements

Every spacecraft is required to have a thermal control design to keep every component of spacecraft within a specified temperature range. To achieve this requirement, it is necessary to clarify thermal interface conditions through the manufacturing, test, launch and operation phases, and to implement and to verify the thermal control design through test or analysis, considering the thermal interface between spacecraft system and its components, as well as the launch vehicle.

5 Design requirements

Each requirement below must be considered for the thermal control design of a spacecraft system, to include:

(1) environmental conditions
(2) temperature conditions of thermal design
(3) thermal interface
(4) thermal control device, material technology
(5) thermal design verification

5.1 Environmental conditions

5.1.1 Fundamental rule

Every spacecraft must meet various thermal environmental requirements at each phase, from production to in-orbit operation. Every spacecraft must also consider the thermal environment during all phases of operation (on the ground, in the earth and moon/planet orbit and interplanetary) in their designed service lives.

On the ground, spacecraft must be maintained within the allowable temperature range, including consideration of temperature and humidity from production to launch. Each spacecraft has its own uniquely defined thermal environment.

During the launch and ascent phases, spacecraft are heated through the fairing(as described in section5.1.3). These heating conditions are specific to each spacecraft and launch vehicle.

Spacecraft fly in orbit of outside the earth’s atmosphere, are heated externally by solar radiation, by albedo (the reflection ingredients of solar radiation from the earth and moon/planet) and by infrared radiation from the earth and moon/planet. These heat inputs are taken under consideration, including potential error and fluctuation, through each spacecraft’s entire designed service life.

5.1.2 Ground environment

Before launch, every thermal environmental condition must be considered for every spacecraft at every phase of production, test, handling, storage, transportation and preparation before launch. If any suitable thermal environmental condition (thermal conditions of batteries etc.) is necessary for spacecraft, a study using ground support equipment shall be performed. Thus, the thermal environmental conditions must be studied in an early stage of a spacecraft's development as possible.

5.1.3 Launch and ascent environment

Monitoring of the thermal environmental conditions and rapid decompression environmental conditions during launch and ascent are important. Changes in the environmental conditions caused by spacecraft itself, however, are not taken into account.

(1) thermal environmental conditions

During the launch phase, the conditions are considered separately (see below) because the thermal environment of a spacecraft is different before and after fairing separation.

<1> There is heating of the spacecraft during the fairing

<2> There is heating of the spacecraft directly after fairing separation

The above thermal environmental conditions must be clearly understood at an early stage of its development as possible.

<1> heating to spacecraft through fairing

Spacecraft are heated by radiation from inside the fairing and as a result, the temperature of the fairing increases. Aerodynamic heating, solar radiant heat, albedo and earth infrared radiant heat increase the temperature of the exterior surface of fairing before fairing separation.

It is necessary to consider the heat exchange (through conduction and radiation) resulting from the interface between spacecraft and launch vehicle. Generally, the temperature record of the fairing interior is shown by the launch vehicle side.

<2> heating to spacecraft directly after fairing separation

After fairing separation, the exterior surface of the spacecraft is heated by the outside environment directly. The spacecraft is heated directly by the free molecular flow of heat by rarefied gas, solar radiant heat, albedo and earth infrared radiant heat. In some cases, it is necessary to consider the heat exchange (through conduction and radiation) resulting from the interface between spacecraft and launch vehicle.

Additionally, with respect to the launch vehicles which use solid fuel motors for the upper-stage propulsion, it is necessary to take into account of the heat conducted from the motor case and the heat from the firing and plume from the upper-stage vehicle.

(2) rapid decompression environmental conditions

It is necessary to consider the blistering of MLI and the reduction of internal gas temperature in the fairing caused by the rapid decompression in the fairing during the vehicle's ascent. In order to prevent the blistering and the separation of MLI, which considerably affects the thermal control system, see section 2.1 (1) below “JERG-2-311 Design Standard to prevent peeling of MLI”.

5.1.4 Orbital environment

5.1.4.1 External thermal input conditions

(1) solar constant

The solar constant is defined as the emission of energy from the unit's surface area vertical to the sunlight per hour, to the outside air at a distance of 1 AU (Astronomical Unit: distance between earth and sun), or 1,366 W/m^2. Though the intensity of solar radiation is in inverse proportion to the square of the distance from the sun, this formula must be adjusted seasonally by about ±3.5 % on average because of the earth's elliptical orbit.

Furthermore, please refer to section 2.1 (5) "JERG-2-141 Standard of Space Environment" for further details of the solar constant.

(2) albedo and infrared radiation

<1> albedo

The albedo is the ratio of sunlight which reflects to the surface of the earth and moon/planet. The earth albedo changes approximately 0.05~0.6 according to the latitude, season and weather condition etc. at the point directly below of spacecraft around the earth. The annual average of albedo over the global surface is 0.3.

<2> infrared radiation

The infrared radiation is the thermal radiation that the earth and moon/planet emit. The Earth's infrared radiation changes approximately 150~300 W/m^2 according to the latitude, season and weather condition etc. at the point directly below of spacecraft around the earth.

The annual average of infrared radiation over the global surface is 237 W/m^2.

<3> application of spacecraft to thermal design

Regarding the application of a thermal design to spacecraft, usually the Earth's albedo and the Earth's infrared radiation are treated as constants in time and in space, and the following value is applied according to the time constant of each component.

Table 5.1.4 Value of the earth albedo and the earth infrared radiation applied to the thermal design

time constant of the component	earth albedo	earth infrared radiation (W/m^2)
Δt < 0.3 hr	+0.30 +30/-0.15	237 +28/-97
0.3 <Δt < 3 hr	0.30±0.10	237 +24/-48
3 hr <Δt	0.30±0.05	237±21

The time constant of Table 5.1.4 is the value determined based upon the relation between the heat capacity of the spacecraft and the thermal resistance for the (internal and external) thermal input. So, if the time constant of spacecraft is known, the estimation of the thermal response of the system shall be possible. If the values from Table 5.1.4 are used for the thermal design of spacecraft, in cases when the tolerance of the Earth's infrared radiation is positive, the tolerance of the Earth's albedo is negative and when the tolerance of the Earth's infrared radiation is negative, the tolerance of the Earth's albedo is positive.

In consideration of components which are sensitive to the change of the external thermal input (such as an external component with a small heat capacity, a small amount of self-heating, or with a large surface area), occasionally it may be necessary to simulate the in time and in space change of the Earth's albedo and Earth's infrared radiation. If so, please refer to section 2.1(5) "JERG-2-141 Standard of Space Environment" to investigate this compensation of external thermal input in detail if necessary.

(3) moon/planetary albedo and infrared radiation

A moon/planet's albedo and infrared radiation with the exception of the Earth and Moon shall be based on current research results.

(4) shade

Shade is defined as the condition of solar light being obstructed by the earth or planet etc. Shade can be classified into two categories: umbra (when solar light is completely obstructed); and penumbra (when the obstruction of the solar light is not complete). When the shade is penumbra for a significant duration, a separate thermal design must be determined separate from the thermal design during the umbra.

5.1.4.2 Orbital conditions

To decide the strength and the direction of the orbital thermal input, the conditions below are must be considered within the parameter of the orbit.

(1) orbital altitude : altitude of apogee and perigee (or semi-major axis and eccentricity)
(2) orbit inclination
(3) annual variation of orbital plane sun angle (so to called β angle)
(4) annual variation of shadow cover rate (or shade time)

5.1.4.3 Attitude conditions

The attitude conditions determine the incident direction of each surface of spacecraft of the orbital heat input. The attitude conditions are defined by the orbital heat input of the solar light, as well as the albedo and infrared radiation of the Earth or lunar/planet. These are expressed by the following:

(1) The solar orientation based on the spacecraft coordinate system; and
(2) The direction for the earth or moon/planet based on the spacecraft coordinate system.

With regard to these items, the time variation profile for each attitude control phase or the duration of the control phase and the amplitude range for the duration is defined. The attitude control phase is mostly divided as follows.

(a) each phase of the position capture
(b) each position at each intermediate orbit (transfer orbit of stationary spacecraft)
(c) steady position
(d) recapture at the abnormal position (in case of defined) (safe hold)

5.1.5 Thermal environment for spacecraft

5.1.5.1 Special quality of thermo-optics

The special quality of thermo-optics for spacecraft is defined here and is an important factor for forming the thermal environment of spacecraft.

The special quality of thermo-optics for the skin of spacecraft is the primary factor to determine the heat exchange between spacecraft and space. Specifically, the surface temperature of spacecraft is determined by the balance between infrared radiation from the spacecraft to space and conversely, the absorbed amount of incident heat flux of orbital heat input from space to the spacecraft. So, in the design of thermal control of spacecraft etc., the values of the following optical characteristics of each surface must be selected properly.

(1) solar light absorptivity (α_S)
(2) infrared emissivity (ε)

Further, for the complex synthetic aspect, for example an aperture plane with thermal blanket (MLI) and cavity, it is necessary to define the equal surface characteristics converting to a simple surface below. In determining these values, it is necessary to clearly define and understand the conditions for which they will be used..

<1> effective solar light absorptivity (α_{Seff})
<2> effective infrared emissivity (ε_{eff})

The following are the considerations in the design of thermal control.

(1) solar light absorptivity (α_S)

Solar light absorptivity α_S (θ) is defined as the ratio of energy absorbed by the body relative to the energy of the total solar light radiation incident to an arbitrary body with an angleθ. The incident angleθ is defined as the angle from the normal direction of the surface of the body. The α_S (θ) is defined as the following formula, where the spectral radiant intensity of solar light is $Js(\lambda)$, the spectral absorption factor of the body for the direction ofθ is $\alpha(\lambda,\theta)$. Here, λ is the wavelength of the light.

$$\alpha_S(\theta) = \frac{\int_0^\infty \alpha(\lambda, \theta) \cdot J_S(\lambda)\, d\lambda}{\int_0^\infty J_S(\lambda)\, d\lambda} \quad (1)$$

Generally, the solar light absorptivity is measured on the condition that the angle of incidence is within 5°and the range of wavelength is 0.25~2.50 μm corresponding to 96% (the range of wavelength of 0.115~100.00μm is 100 %).

Furthermore, it is necessary to occasionally consider a range over 25μm of wavelength.

(2) infrared emissivity (ε)

Generally, the infrared emissivity has the following 3 kinds of characteristic values:.

<1> whole hemisphere emissivity $\varepsilon_H(T)$
<2> spectral hemisphere emissivity $\varepsilon_H(\lambda, T)$
<3> vertical spectral emissivity $\varepsilon_N(\lambda, \theta, T)$

The whole hemisphere emissivity $\varepsilon_H(T)$ is defined as the ratio of the energy emitted from the surface of a body with T representing surface temperature to hemispherical space with unit time and unit area relative to the energy emitted from the surface of a black body with the same surface temperature as hemispherical space with unit time and unit area, and is expressed with following formula.

$$\varepsilon_H(T) = \frac{\int_0^\infty \varepsilon_H(\lambda, T) E_b(\lambda, T) d\lambda}{\int_0^\infty E_b(\lambda, T) d\lambda} = \frac{\int_0^\infty \varepsilon_H(\lambda, T) E_b(\lambda, T) d\lambda}{\sigma T^4} \quad (2)$$

Here,

$E_b(\lambda, T)$	:	spectral hemisphere emissive energy of black body	$[\mathrm{W}/(\mathrm{m}^2 \cdot \mu\mathrm{m})]$
$\varepsilon_H(\lambda, T)$	:	spectral hemisphere emissivity	
σ	:	Stefan-Boltzmann constant 5.67 X 10^{-8}	$[\mathrm{W}/(\mathrm{m}^2 \cdot \mathrm{K}^4)]$

Generally, the infrared emissivity is measured at a temperature of 300K and with a range of wavelength within 2.5-25μm. Furthermore, it is necessary to occasionally consider a wider range of wavelength.

Additionally, this value is temperature dependent. It is generally understood that the value usually obtained is around the normal temperature. However, it is necessary to consider this temperature dependence when the surface temperature is not regarded as normal.

Generally, attention must be paid to the value of hemisphere emissivity of the dielectric (insulator) surface when it is smaller than the value of vertical emissivity. With respect to the general radiating surface controlled by the emissivity, the characteristics of heat radiation is the dielectric surface., However, as the value of emissivity obtained by the simple procedure is mostly vertical emissivity, it is necessary to understand properly the origin of the value to be used for the sizing design of the radiating surface.

(3) transmissivity (τ)

The transmissivity (τ) is the ratio of penetration energy to whole incident energy on the surface with permeability for incidence and radiation. There are two kinds of characteristic values as follow:

<1> sunlight transmissivity ($\tau_{S:}$ transmissivity of sunlight spectrum)

<2> infrared transmissivity ($\tau_{IR:}$ transmissivity of black radiation spectrum of generally 300K)

The reflectivity (ρ) is also used, except in terms of the characteristics about the radiation described above. This characteristic is defined for each solar light and infrared radiation and is equal to absorptivity and emissivity. The following relation comes to existence between reflectivity, absorptivity and transmissivity in terms of radiation and incidence at the same wavelength.

(reflectivity) + (absorptivity) + (transmissivity) = 1

In terms of reflectivity, as the ratio of ingredient of diffuse reflection and of regular reflection occurs, this value is defined as the specularity. Regarding this value, the following two kinds are also defined according to incidence and radiation spectrum.

<1> solar light specularity (ρ_S)

<2> infrared specularity (ρ_{IR})

Here, the definition of specularity is shown as the following equation.

$$(specularity) = \frac{(ingredien\ to\ \ fregular\ reflection)}{(reflectivity) = (ingredient\ of\ \ diffuse\ reflection) + (ingredient\ of\ \ regular\ reflection)}$$

As mentioned above, the thermo-optical properties on each surface change due to degradation and pollution from the thruster exhaust. It is therefore necessary to pay attention to the range of fluctuation (mainly concerning solar light) caused by the degradation and the pollution during its life time, in addition to its fabrication tolerance. Moreover, it is necessary to consider the dispersion between the various measured samples of α_S and ε_H etc.

5.1.5.2 Internal generation of heat by a spacecraft

The internal generation of heat by a spacecraft is caused by the operation of the each component. The component generating the internal heat is roughly classified into electrical and electronic components, propulsion components, solar cell panel and other components, according to its heat source. The following points regarding spacecraft components must be considered for the thermal control design.

(1) electrical and electronic component

Included in this group are the components of the power supply system including battery, communication system and attitude control system which require actuators. Heat is generated by converting electrical energy to thermal energy during its operation. Additionally, in case of the battery, the endothermic generation of heat caused by chemical reaction is added.

Generally, the caloric value of equipment changes according to the operating mode of component. The operating mode of each component is determined by the operating mode of the spacecraft. Therefore, the following must be considered with respect to electrical and electronic components on the spacecraft: .

<1> operating mode of spacecraft (what component, how it operates)
<2> caloric value of each component by operating mode

The specific points of concern for each item are shown below.

<1> operating mode of spacecraft

Though the component to be operated is determined at each phase during mission operation, there are multiple alternatives about what component is operated, because of its redundant systems and multiple options for the mission operation. Among them, the operating mode is considered as the worst case, selecting all combined mode including the worst conditions of high/low temperature.

The following constraint conditions determine the operation alternatives of each component.

(a) whether component is to be operated at the same time or not to be operated at the same time
(b) whether there is a condition requiring a component to be used with the redundant system
(c) whether there are constraints of mission operations (such as parallel operation, operating duty and constraints on operation sequence etc.)

<b> caloric value of each component in operating mode

As the caloric value of each component changes according to various conditions, the following items shall be defined.

(a) classification of operating mode and caloric value of each mode
(b) correlation equation in case the generation of heat depends upon temperature and timing etc.

(2) propulsion component

The secondary propulsion system used for attitude control and trajectory control is another component for consideration of heat generation. The secondary propulsion system is classified to the tank storing the propellant and the plumbing (including devices) transporting propellant to the thrusters.

With respect to the thermal design for the tank and plumbing (including devices), it is necessary to consider maintaining the temperature of the propellant within the allowable range, to avoid a rise or drop in the partial temperature distribution.

The thrusters are classified into two categories: chemical propulsion and non-chemical propulsion (including electrical propulsion). Chemical propulsion creates propulsion by generating the gas by chemical reaction or combustion, such as a liquid propellant thruster. An electrical propulsion system, such as an ion engine, creates propulsion by converting the electrical energy into kinetic energy.

These components are primarily mounted on the outside of the spacecraft, and function as the thermal interface design of the thermal control system design following its confirmation of the feasibility of the exhaust heat design component.

The following items must be considered in terms of propulsion components.

<1> pollution [The plume of thruster exhaust] affects thermal control material
<2> excessive heat inflow to spacecraft at soak back

(3) solar cell panel

The solar cells attached to the solar cell panel supplies the electric power converted from the
incidental solar energy for the spacecraft. During this process, all of the energy not converted to electric power becomes thermal energy.

In the thermal design of solar cell panel on orbit, careful consideration must be given to the significant temperature changes in the external thermal environment caused by the trajectory conditions of solar rays during periods of sunlight and shade and the efficiency of power conversion of the solar cell during those periods.

The temperature level of the solar cell panel in-orbit changes according to the extent of the power conversion of the incidental solar energy, and depends on the heat balance of the incidental energy to the solar cell, the radiant energy in space and the generation of electric power.

Please refer to section2.1(6) “JERG-2-212 Design Standard of Wire Derating” regarding the thermal design of the wire harness.

(4) other components

<1> A rotating component (e.g. an antenna with a driving motorized mechanism)

For a rotating component, which has a driving motorized mechanism for rotating on the spacecraft, it is necessary to consider to the following items.

(a) The temperature stability pertaining to the driving motorized mechanism while it is rotating.

(b) The existence of temperature dependence pertaining to the torque performance of the driving motorized mechanism

(c) The secure exhaust of heat from the driving motorized mechanism (exothermic body)

(d) others

<2> refrigerator

The refrigerator used in order to cool down the sensing port of the optical sensor etc., is classified into several categories according to its type of cooling mechanism. The issues which must be address when mounting it to a spacecraft include the mechanical environment during the launch and ascending phases and measurements to control vibration and the heat exhaust of the driving mechanism (e.g., the compressor etc.) while in-orbit. The refrigerator's cooling mechanism must reduce the temperature to the extent possible.

In terms of thermal control design, the following must be considered:

(a) The exhaust heat must be designed to be compatible with the mechanical environment
(i.e. a secure exhaust heat pass etc.)

(b) There must be built-in protection against thermal disturbance to the cooling portion (cold head)
(including measures to mitigate the presence of heat)

(c) There must be measures against thermal deformation to maintain alignment

(d) There must be temperature stability of the refrigerator body for protection against the thermal environment.

(e) others

<3> observation equipment

In case of mounting observation equipment on spacecraft, the following items must be considered with respect to the thermal design because the performance of observation equipment depends to a degree on the amount of thermal noise from the sensor itself. This includes the following considerations:

(a) The temperature compensating for heat fluctuation while the observation equipment operating

(b) The stability of the internal thermal environmental temperature of the observation device

(c) The measures designed to mitigate thermal deformation and to maintain alignment

(d) others

5.2 Thermal design temperature conditions

5.2.1 Classification of temperature range

The temperature range of component is classified below.

<1> design estimated temperature range

<2> maximum estimated temperature range

<3> allowable temperature range

Fig. 5.2 shows mutual relationship of temperature range above mentioned. The details are described hereafter.

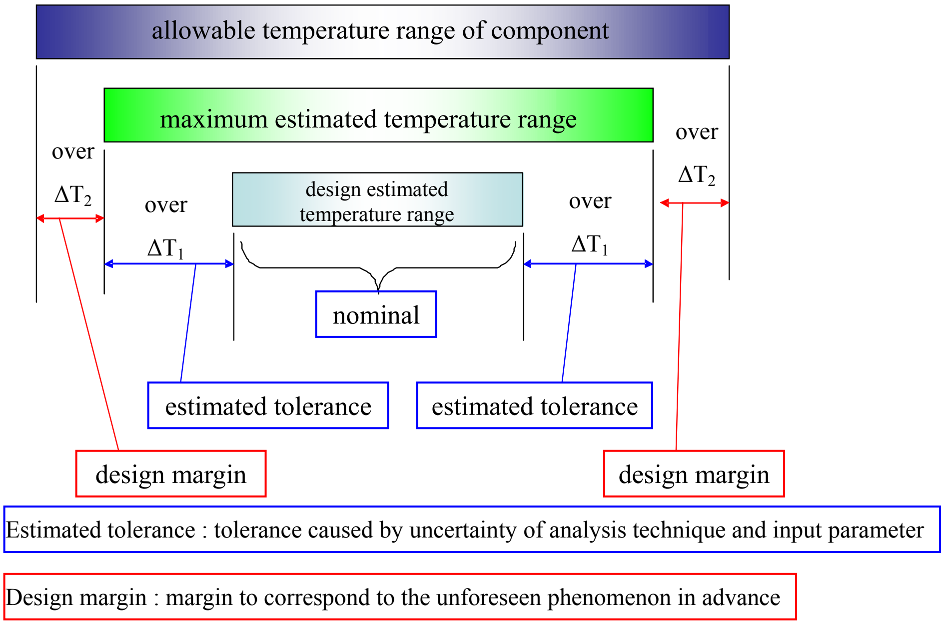


Fig 5.2 Mutual relationship of allowable temperature range and estimated temperature range

5.2.2 Design of the estimated temperature range

The design of the estimated temperature range is the nominal temperature range considering the heat input fluctuation etc. caused by seasonal variation, operation and degradation of thermal control devices. Specifically, it means the temperature range obtained by thermal analysis considering the following factors.

(a) The seasonal variation and short-range variation of external heat source strength.

(b) The variation of heat input by the change of the spacecraft's posture

(c) The changes in the operating condition of component which brings about changes in the output of heat (including degradation of a component's heating element)

(d) The deterioration of thermal control devices through age

(e) The temperature changes to the thermal properties of materials uncertainties as a result of contamination etc. which must be addressed using results from flight testing

(f) The change of the radiation boundary temperature (i.e. the temperature of a body while releasing radiation which is exchanged with a component) and the conductive interface temperature (i.e. the temperature of a body while releasing conducted heat which is exchanged with a component)

5.2.3 Maximum estimated temperature range

The maximum estimated temperature range means the in-orbit maximum estimated temperature range which includes a predicted error of ΔT_1 to become the “design estimated temperature range”.

5.2.4 Allowable temperature ranges

The allowable temperature range is classified below according to the operating condition and service condition of the component.

(a) The operational temperature range (i.e. temperature range for maintaining a component’s efficiency)

This is also the temperature range to satisfy its specifications and performance.

(b) The temperature range for maintaining a component’s function

This temperature range refers to a-drop in-orbit performance drops below its specification, but the components still functions and satisfies its performance requirements without any malfunction nor degradation during operation.

(c) inactive temperature range (preserved temperature range)

The temperature range not create a malfunction or permanent degradation during inactivity while in orbit.

(d) Turn-on temperature range

The temperature range able to turn on the components and does not create a non-recoverable degradation while in orbit.

5.2.5 Estimated tolerance ΔT_1

5.2.5.1 Meaning of estimated tolerance ΔT_1

Estimated tolerance ΔT_1 means the temperature tolerance considering the uncertainty caused by analysis technique and input parameter for analysis.

Furthermore, estimated tolerance ΔT_1 is able to be reviewed after evaluating quantitatively the tolerance using thermal balancing testing etc. and assessing its validity.

(1) uncertainty caused by analysis technique

<1>uncertainties of mathematical modeling: definition of node, approximation of albedo/infrared radiation of earth or moon/planet, approximation of form of radiation plane/flow channel of heat conduction, modeling of MLI

<2> uncertainties of analysis tool: uncertainty of Monte Carlo method/numerical integration
method

(2) uncertainty caused by input parameter for analysis

<1> measurement error: uncertainty of thermal environment simulation, uncertainty of method of measurement and measuring instrument

<2> uncertainty regarding the calorific value of a component
<3> fabrication tolerance: uncertainty of the MLI's effective emissivity and conduct type
<4> uncertainty regarding the amount of degradation due to age: uncertainty in the amount of material degradation

5.2.5.2 Procedure to set up estimated toleranceΔT_1

The empirical orientation method is mainly used to establish estimated tolerance ΔT_1 because of the uncertainty of analysis method and its parameters.

(1) empirical orientation method

Considers orbit, position, thermal control method and similarities based on past results of spacecraft etc., ΔT_1 establishes a value that is even based on the past experience.

(2) probability method

Performing error analysis for each parameter and searching for temperature error regarding probability, as defined below.

ΔT_1 = (absolute sum of errors) ・・・・・worst evaluation

or

ΔT_1 = (RSS of errors) ・・・・・to be set up so that errors can be addressed completely independently

5.2.6 Design margin ΔT_2

5.2.6.1 Meaning of design margin ΔT_2 setting

Design marginΔT_2 is set up to consider speculation using a demonstration test model while including phenomena difficult to quantify.

5.2.6.2 Procedure to set up design margin ΔT_2

For the design margin ΔT_2 based on experience to be set up, the following items must be considered.

(a) levels of past results (factoring in new developments, quality of past results)
(b) criticality in case of deviating temperature range
Criticality caused by deviation
(c) risk tolerance limit for project

5.2.7 ΔT_1 and ΔT_2 of component thermally controlled in active type

5.2.7.1 Approach to thermal control

For defining ΔT_1 and ΔT_2 of a component (thermally controlled in active type) and ΔT_1 and ΔT_2 of a component (thermally controlled in passive type), the standards for consideration are described below.

Concerning to the heat input Q (which includes self-heating) to a spacecraft in-orbit, the temperature of the component must be maintained within the allowable temperature range by setting up the thermal coupling using conductivity between the component and its surroundings.

Therefore, for the following formula, T is set up within the specified temperature range within a variable range of Q by selecting the fixed value Q properly. Here, T_0 is the temperature limit (This is also a variable value).

$$Q=G\Delta T=G(T-T_0) \qquad (5.1)$$

Fig. 5.2.1 shows the conceptual diagram of passive type thermal control.

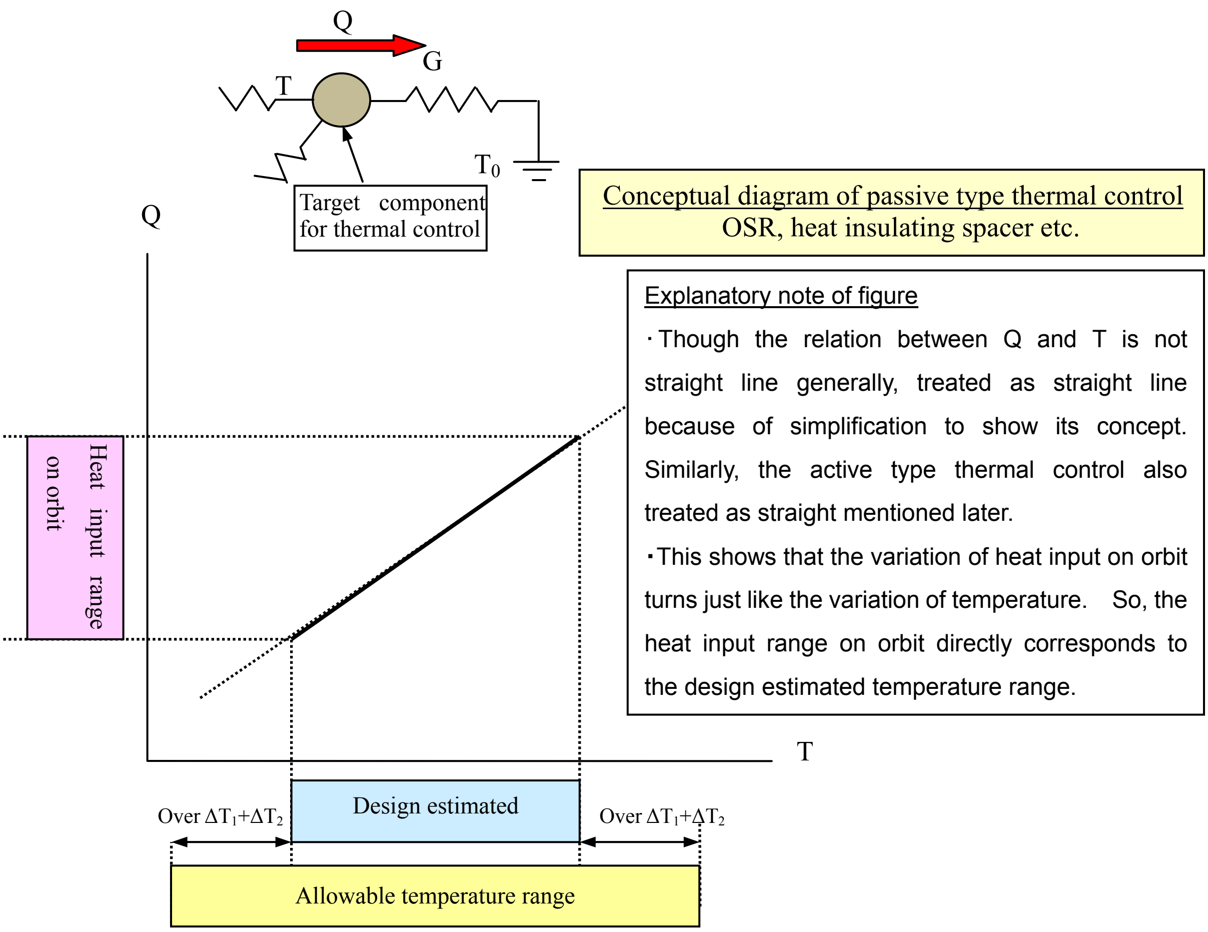


Fig 5.2.1 Conceptual diagram and design margin of passive type thermal control

5.2.7.2 Requirements for ΔT_1 and ΔT_2

With respect to the aforementioned formula at section (5.1), T shall be set up within designated temperature range by regarding Q or G, or both as variable.

The sample of G as variable is thermal louver and a variable conducting type of heat pipe (VCHP) etc., with Q as variable is thermal control by the heater.

The following diagram shows the active type thermal control concept and design requirement.

(1) conceptual diagram of active type thermal control (in case of Q as variable [part 1])

when the spare control power is available only to the lower limit of the estimated design temperature range.

The sample diagram below is a thermal control heater to maintain the temperature of lower limit.

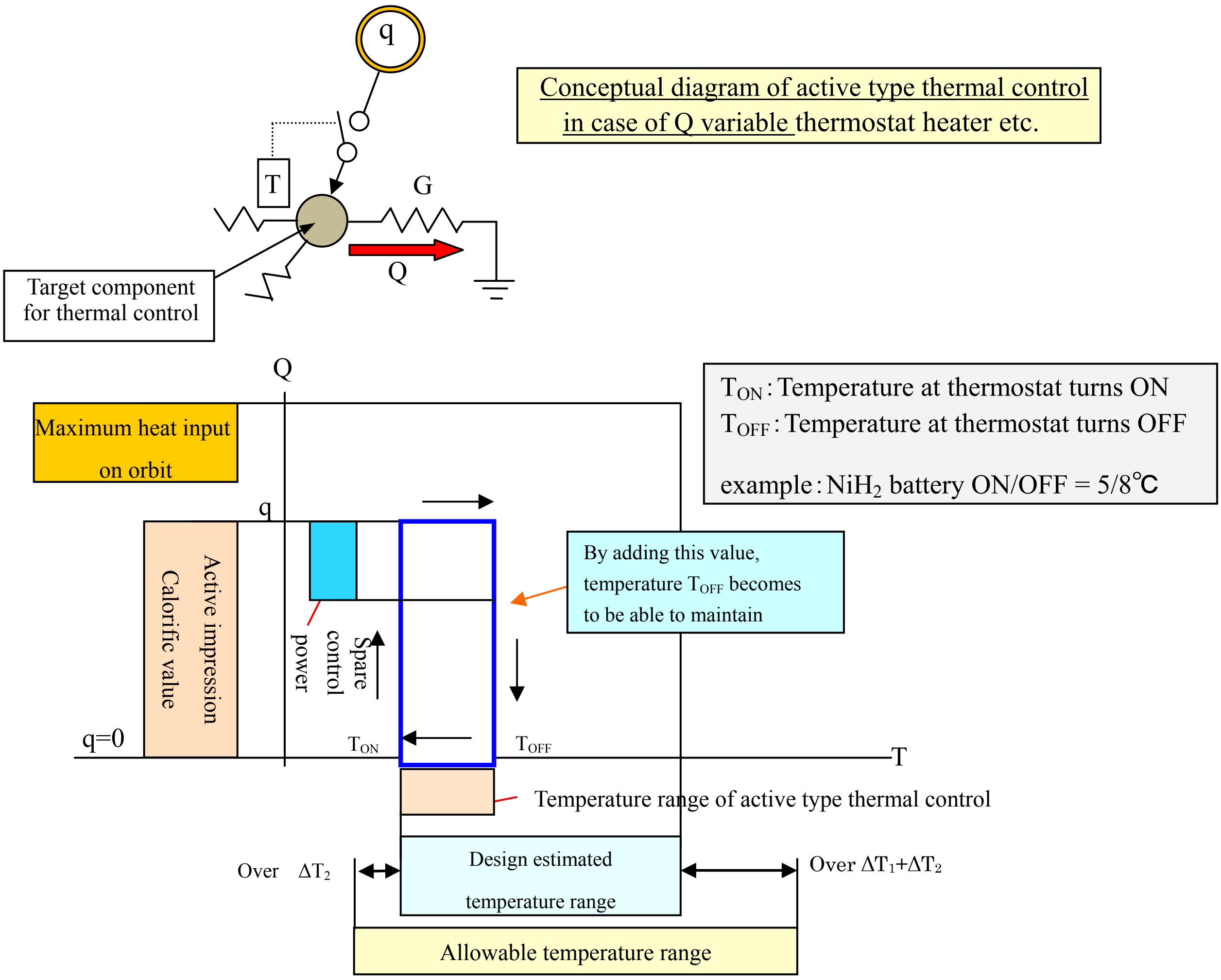


Fig. 5.2.2 Conceptual diagram and design margin of active type thermal control (Q is variable)

Explanatory note of figure

(1) This is conceptual diagram in case of thermal controlling the lower limit of allowable temperature range by active type thermal control. Specifically, this case corresponds to the thermal control of a battery mounted to a high capacity geostationary satellite and propulsion system (tank and plumbing) with a high allowable lower limit as compared to other components.

(2) For such a component, usually the size of the radiating surface is measured based on the time of maximum generation of heat (maximum heat input) for the upper limit temperature. Conversely, the lower limit temperature is necessary to maintain a minimum generation of heat with a heater.

Design requirement

Though the upper temperature limit is designed as a passive type thermal control, for the following describes the lower temperature limit.

(1) Concerning the requirement for setting up the heater capacity, it should be larger than both the amount of calories able to rise from heater OFF temperature to over ΔT_1℃ with the value multiplied to account for the safety factor and heater capacity necessary to maintain the heater OFF position temperature.

(2) At the lower temperature limit with active type control, the designed estimated temperature range should be secured to the margin over ΔT_2℃ of the allowable temperature range. (Further, in some cases, ΔT_2℃ secures its margin through the heater capacity.)

Example for reference

DRTS NiH_2 battery

・ allowable temperature (-10~25[35(*1)]℃) thermostat : ON/OFF = 5/8 (±1℃)

(*1) at electric discharge and 3 hours after charge start

・ in-orbit data

・ At trickle charging, it shall be maintained within the thermostat control temperature range [5~8℃]

・ At discharge, the temperature increase above the temperature of thermostat OFF (all heaters OFF)

[approximately 18℃ at DC arc jet operation, approximately 10℃ at maximum shade in equinox]

(2) conceptual diagram of active type thermal control (in case of Q as variable [part 2]) when the spare control power is available for both the lower and the upper limit of the estimated design temperature range..

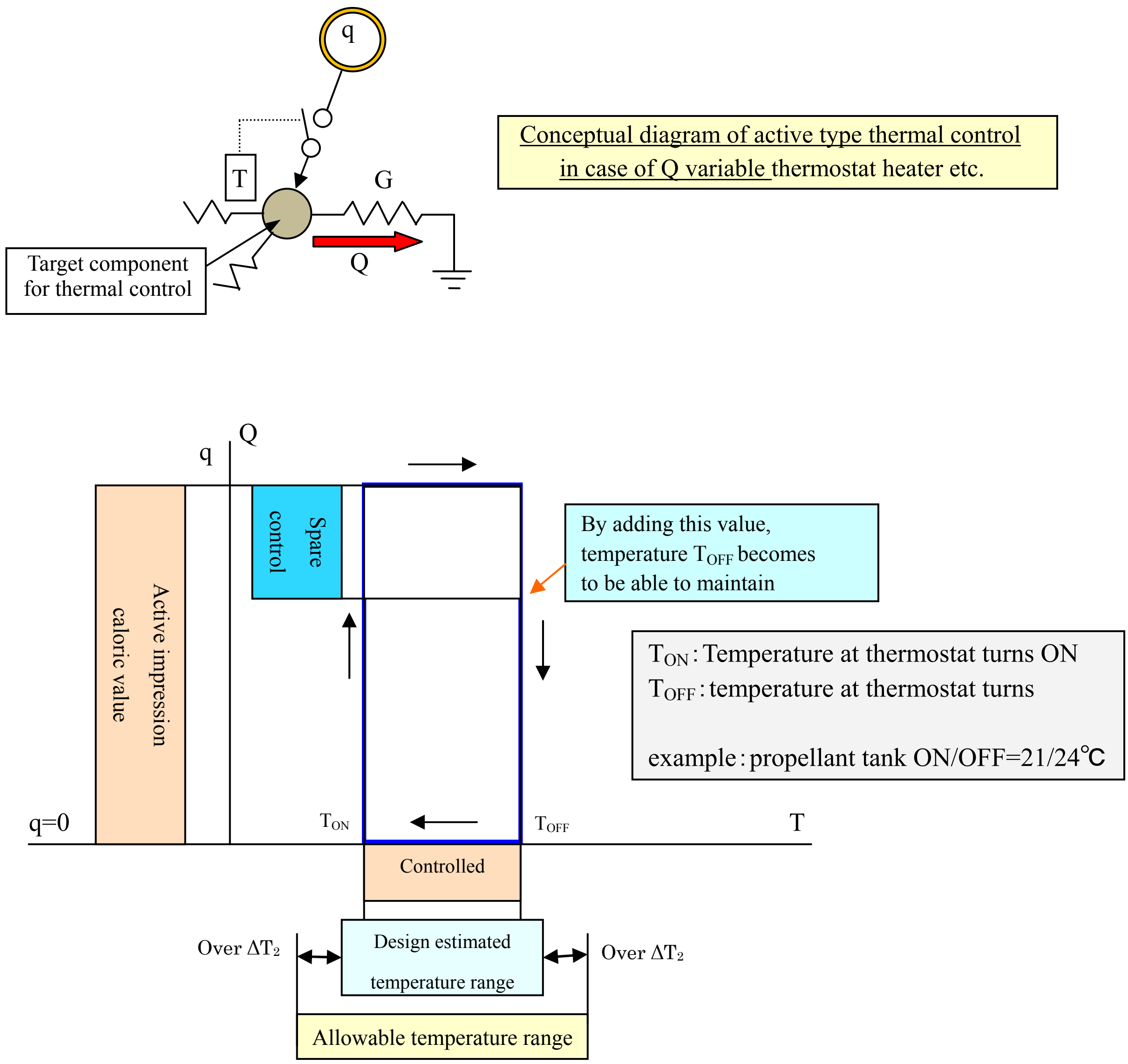


Fig. 5.2.3 Conceptual diagram and design margin of active type thermal control (Q is variable)

<u>Explanatory note of figure</u>

(3) This is a conceptual diagram when there is thermal control to the upper and lower limit of allowable temperature ranges by an active type of thermal control.

Specifically, for example, this is the case for the thermal control component for a battery and optical sensor with a narrow allowable temperature range. The component that the allowable temperature range is narrower than the design margin (upper and lower limit) of the passive type of thermal design.

<u>Design requirement</u>

The upper temperature limit, principally the value over $\Delta T_1+\Delta T_2$℃ from upper limit of design temperature, should be secured (secure of radiating surface), when the active type thermal control of heater etc. is turned to OFF.

The design margin of lower limit temperature is set up as follows.

<1> Concerning to the requirement for setting up heater capacity, it is the principal that it should be larger than both the amount of calories able to rise from heater OFF temperature to overΔT_1 ℃ and the value multiplied to account for the safety factor and heater capacity necessary to maintain the heater OFF position temperature.

<2> At the lower temperature limit with active type thermal control, the design estimated temperature range should be the secured margin over ΔT_2℃ to the allowable temperature range. (Further, in some case, ΔT_2℃ secures its margin through the heater capacity.)

(3) conceptual diagram of active type thermal control (in case of G as variable [part 1]) when the spare control power is available for both the lower and the upper limit of the estimated design temperature range

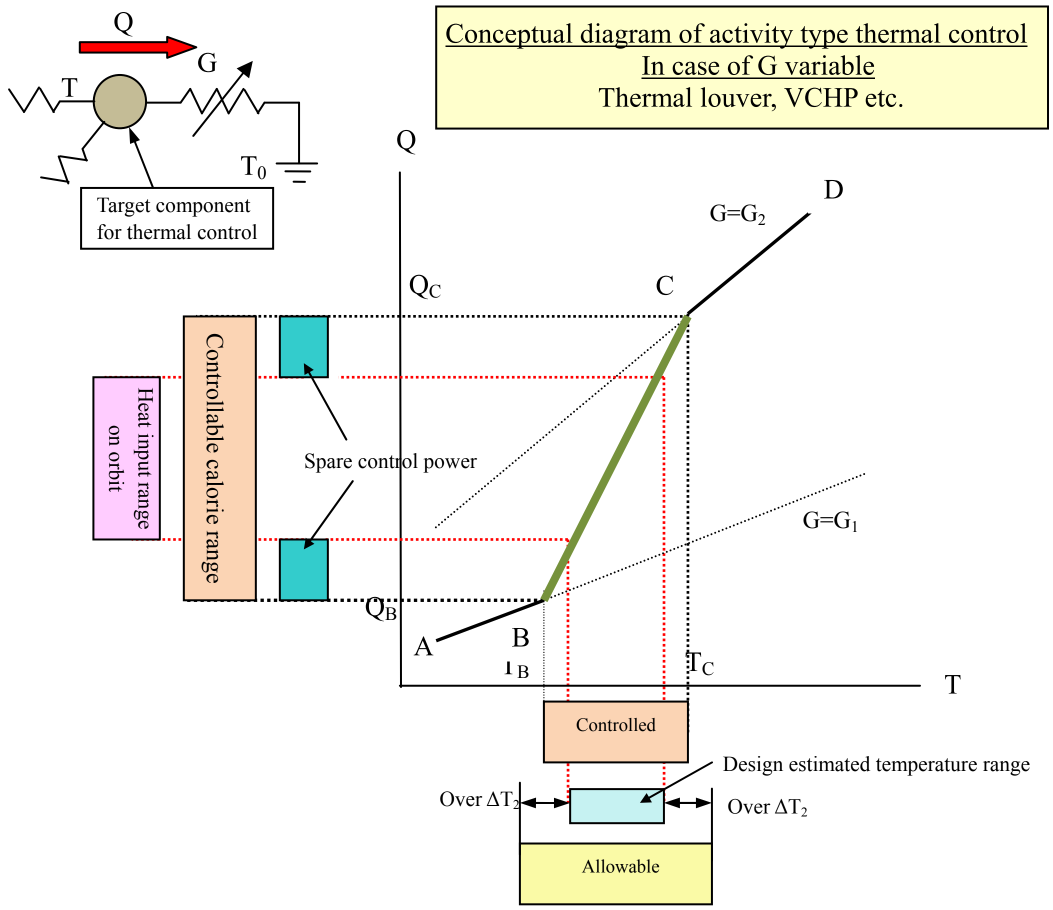


Fig. 5.2.4 Conceptual diagram and design margin of active type thermal control (G is variable)

Explanatory note of figure

・ The following is an explanation of the figure as an example of the thermal louver. (in principal)

・ In the range between line segment AB, the blade is closed and the radiating surface is constant with minimum emissivity (for example ε=0.1).

・ In the range between line segment CD, conversely the blade is hull open and the radiating surface is constant with maximum emissivity (for example ε=0.8).

・ In the range between line segment BC, ε changes from 0.1 to 0.8 according to the temperature of radiating surface.

・ When the calorie variation is from Q_B to Q_C, the radiating surface is maintained within the controlled temperature range from T_B to T_C.

・ At above figure, as the heat input range on orbit is between Q_B and Q_C, it indicates being maintained within the controlled temperature range.

・ Because of the factors that the temperature specified point and controlling sensing point is different and/or setting tolerance of temperature of radiating surface and divergence angle of the blade, the range of estimated design temperature of the component does not necessarily correspond to the heat input range on orbit.

Design requirement

<1> In principle, the spare control power (calorie) shall be the calorie to be able to change the temperature over ΔT_1℃ with the calorie from the upper and lower limit of design estimated range.

<2> The estimated design temperature range shall be a secured margin over ΔT_2℃ from the allowable temperature range.

(4) conceptual diagram of active type thermal control (in case of G as variable [part 2]) when the spare control power is not available for both the lower limit and the upper limit of the estimated design temperature range

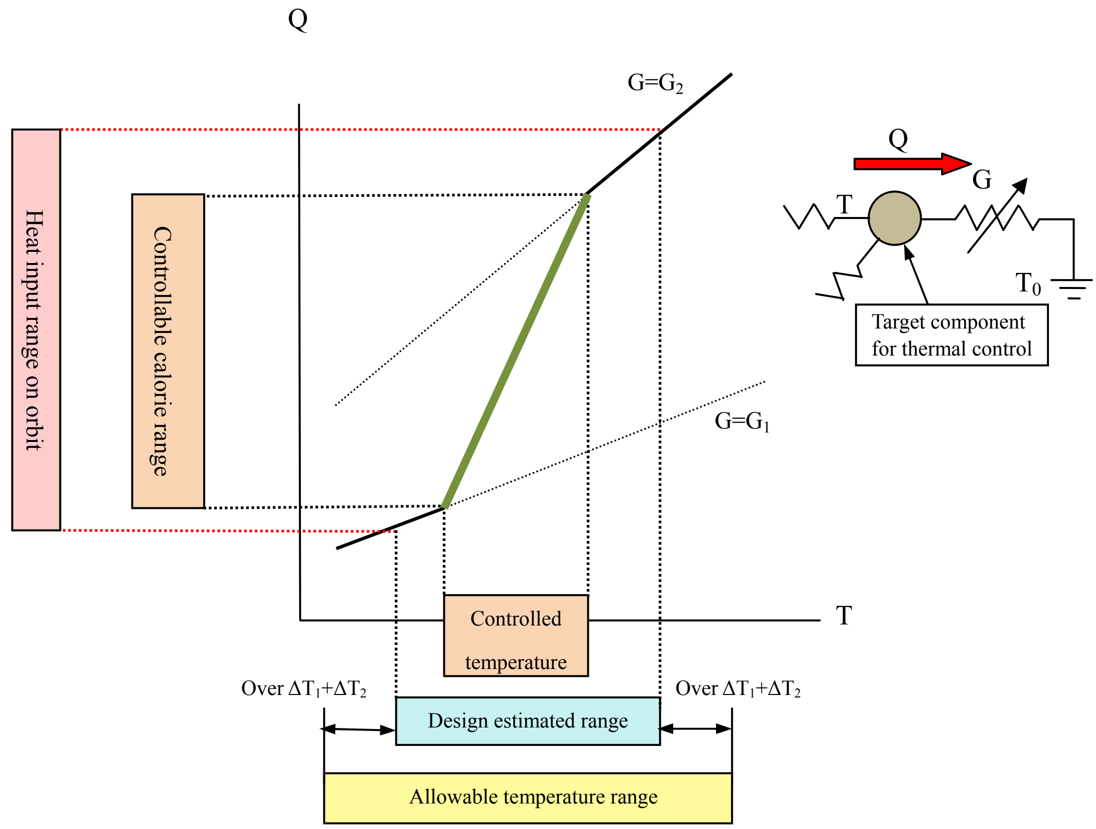


Fig. 5.2.5 Conceptual diagram and design margin of active type thermal control (G is variable)

Explanatory note of figure

- Same as Fig. 5.2.4, explanation of the figure using the example of the thermal louver. (in principal)
- Although in the previous figure, the heat input range on orbit was within the controllable calorie range of the louver, as compared to this figure which was over the range.
- So, upper temperature limit is passively thermal controlled under the fixed value of ε=0.8 blade full open, lower temperature limit is also passively thermal controlled under the fixed value of ε=0.1 blade close.
- Therefore, at the upper and lower limits, there is similarity for the procedure to secure the margin of temperature using passive type thermal control.

Design requirement

<1> Because the heat input range in-orbit is over the controllable heat input range, the procedure using the passive type thermal control is applied. That is, the estimated design temperature range shall be secured over the temperature margin from allowable temperature range of prediction error of ΔT_1℃ and temperature margin of ΔT_2℃.

<2> Additionally, when there exists spare of control power at the upper or lower limits of the estimated temperature shown in Fig. 5.2.4, the active type thermal control shall be applied.

5.2.8 Thermal control of thermally critical component

Because thermal control components are sensitive to temperature, it is necessary to pay special attention to their thermal control design (e.g. battery, propellant plumbing of attitude control system etc.) which are defined as a thermally critical components.

Normally, the following items are considered thermally critical components

・ components with narrow allowable temperature ranges

・ components with wider allowable temperature ranges which diverge significantly from average temperature level of a usual spacecraft

As the thermally critical component is difficult to secure within the temperature margin (ΔT_1+ΔT_2) as a general component, it is required that they satisfy one of the following.

(1) to apply active thermal control
(to keep margin not to temperature, but control ability, refer to 5.2.7)

(2) to design the small prediction error (to adopt a validated/proven control system, to let it thermally separate from others, and to isolate the influence of uncertainty from other portions of the spacecraft,)

5.3 Thermal interface

For the purpose of implementing the design of both a spacecraft system and its components smoothly, the thermal interface requirements are defined below.

5.3.1 Related document with thermal interface

(1) thermal interface condition document

This document defines the thermal interface condition of the system, necessary to implement the thermal design of a spacecraft system and its components (equipment). This document is prepared by the system side.

(2) interface control document

This document is called ICD, and defines the thermal interface condition of the component necessary for the system side to implement the thermal design of a spacecraft system. This document is prepared by the component side based on the instruction defined at section 5.3.1 (4).

(3) interface thermal mathematical model

This is the information which properly expresses the component necessary to implement the thermal analysis by the system side, in the format of a thermal mathematical model. The provision of this information is prepared using electronic data and the document, based on the instructions defined at section (4) below. This information shall be attached to the manual describing the contents of the thermal mathematical model. This thermal mathematical model and manual is prepared by the component side.

(4) instruction for interface thermal mathematical model and ICD

This document describes the instructions to prepare the above interface thermal mathematical model and ICD by the component side, and is prepared by the system side.

5.3.2 Thermal interface demarcation

5.3.2.1 Definition of thermal interface demarcation

The purpose is to define the most suitable border region for the thermal design conditions of the spacecraft system and components, as well as to define the conditions in this region.

This border region is named “Thermal Interface Demarcation”, and is principally the following region. Concretely, it is described as the thermal interface condition document and ICD of each component defined in section 5.3.1.

(1) internal onboard equipment

(a) thermal interface demarcation concerning conductive heat coupling

will be placed at the clamp face on the system side and component side of each component. Furthermore, the internal onboard equipment (which includes insulated components) will be treated the same as exposed onboard equipment.

(b) thermal interface demarcation concerning radiation heat coupling

In cases when the thermal design of a component is not implemented by the information of conductive heat coupling, it shall be implemented as the need arises. In terms of setting, it shall be the outer surface of each component.

(2) exposed onboard equipment

(a) thermal interface demarcation concerning conductive heat coupling

will be placed on the clamp face on the system side and the component side of each component.

(b) thermal interface demarcation concerning radiation heat coupling

will be placed on the clamp face on the component side, on the surface of component and on the outer surface, opposite the spacecraft, along with other components.

Fig.5.3 and Fig. 5.4 show the conceptual diagram of thermal interface demarcation of internal onboard equipment and exposed onboard equipment

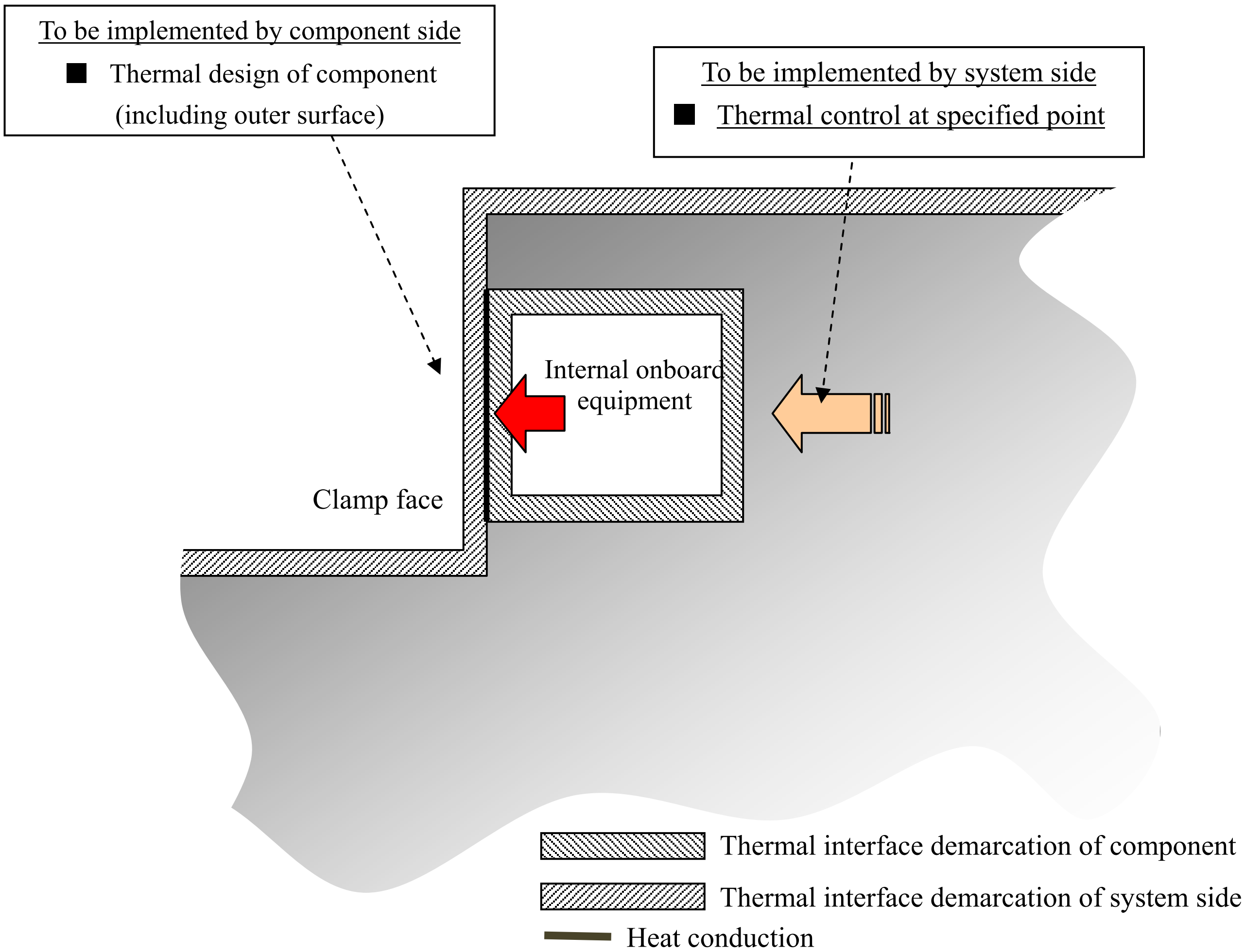


Fig. 5.3 Thermal interface demarcation of internally onboard equipment and spacecraft

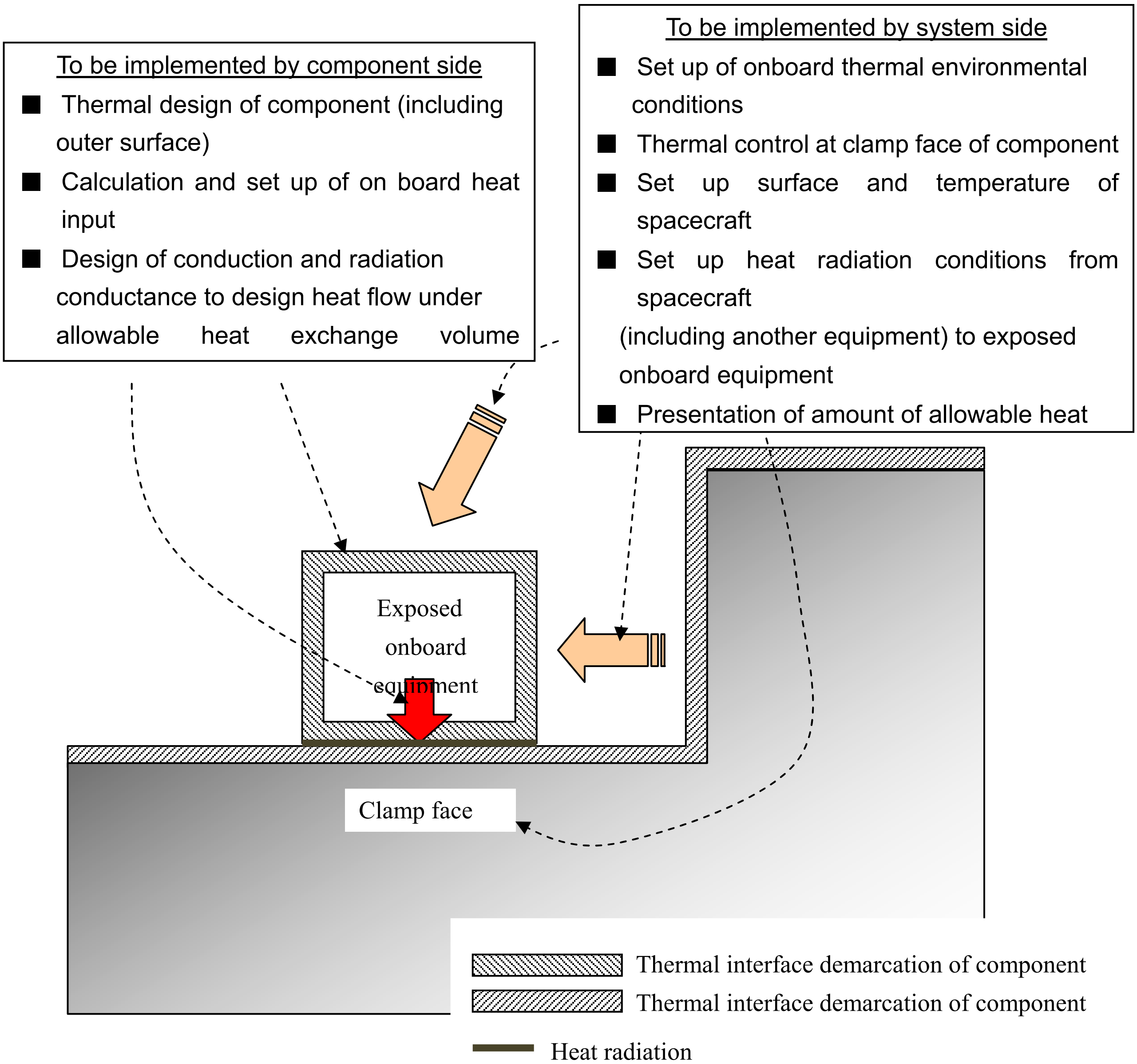


Fig. 5.4 Thermal interface demarcation of exposed onboard equipment and spacecraft

5.3.2.2 Items to be defined as thermal interface conditions

(1) thermal interface between system and internal onboard component

The following items shall be defined as the thermal interface between the spacecraft and the internal on board component. The concrete contents of these items shall be defined in ICD and prepared by the component side.

(a) geometries and contact area

Regarding the clamp face of a component, the diameter of a mounting hole, the position of mounting hole, the shape of the part of clamp screw including thickness, dimensions and contact area shall all be defined. Regarding the outer surface except the attaching portion, the shape and dimensions shall be specified in ICD.

(b) finishing treatment for thermal control

The clamp face of the component, employed material, surface treatment, plane roughness and

deviation from flatness shall be defined according to official standard as JIS and MIL standard. Additionally, with respect to the outer surface except the clamp face, the employed material, external surface treatment and emissivity shall be defined separately.

(c) calorific value

Calorific value of the component shall be defined with consideration to its operating condition and include its tolerance..

(d) heat capacity

The value of heat capacity (mass X specific heat) shall be calculated for each component, and defined to include its tolerance.

(e) average heat generating density

The average heat generating density (W/cm^2) of the clamp face of the component shall be calculated according to the definitional equation indicated at 5.3.4.1 (5), and defined.

(f) temperature specified point

As a general rule, the point of base plate (mounting flange etc.) shall be selected as a temperature specified point of the component, and the details of the position shall be defined in the ICD.

(g) conditions for mounting

Meeting the demand for mounting, thermal filler or heat insulator shall be used as an interface at the attaching portion of the component. The conditions regarding its mounting shall be defined.

(h) active thermal control

Regarding the component's applied active thermal control, control system, calorific value, mounting position of heater etc., the controlled temperature range and operating method shall be defined.

(i) interface thermal mathematical model

When using the interface thermal mathematical model defined at 5.3.1 (3) as the thermal interface condition, the node splitting, the shape and dimensions of the outer surface node, the conduction /radiation combined network between nodes and heat capacity of each node etc. shall be defined.

In creating this mathematical model, instruction for the interface of the thermal mathematical model shall be based on the model defined at section 5.3.1 (4).

(2) thermal interface between the system and the exposed onboard component

The thermal interface of an exposed onboard component is different from the interface of an internal onboard component because of the thermal interface direct exposure with outer space, as well as because the thermal coupling of outer space is generally stronger than the coupling of a spacecraft system. Moreover, large portions of the outer surface of the spacecraft cannot control freely, and exist under an exposed visual field etc. Therefore, the thermal control design is mainly the thermal interface control for outer space, which has the high independence. Moreover, most of the surface temperature of the component is not uniform nor is it regarded as one node. In principle, the interface is defined with the thermal mathematical model which includes the inside of the component.

5.3.3 Thermal interface design prepared by the system side

In implementing the thermal interface design of the component by system side, internal onboard equipment is designed to be able to let calorific value emit from clamp face of component by thermal conduction.

Regarding the exposed component, the thermal balance of exposed in-orbit equipment must be considered at the most suitable temperature, under the trajectory conditions of the spacecraft, the installation condition of exposed onboard equipment with the spacecraft system and the thermal condition of internal heat generation etc. by exposed onboard equipment.

The system side implements thermal analysis for acquiring the information and implementing the interface thermal mathematical model into the system thermal mathematical model described in ICD and indicated by the component side, which conducts the thermal design for the system.

5.3.4 Thermal interface design prepared by component side

The component side implements the thermal design analysis based on the thermal interface condition document defined at 5.3.1 (1) as indicated by the system side, and confirms the suitability of the thermal design of the component. Furthermore, the interface thermal mathematical model (defined at 5.3.1 (3)) prepared by the component side and provided to the system side, shall be validated as to its suitability through tests and analysis implemented by the component side.

5.3.4.1 Requirements for internal onboard equipment

Internal onboard equipment shall be designed to satisfy the following requirements. In case it does not satisfy these requirements, the component side shall coordinate with the system side and specify the discrepancies to ICD.

(1) approach to radiation

In the thermal design of the internal onboard equipment, it shall be the premise that self-heating is released by conductive heat transfer only from the clamp face to the spacecraft.

In cases when another heat abstraction is necessary without the conductive heat transfer from clamp face, the component side shall specify ICD after coordinating with the system side, providing the interface thermal mathematical model defined at 5.3.1 (3) to the system side.

(2) homogenization of surface temperature

In the design of a component, the temperature, not only at the clamp face but also at each section of the component casing including the clamp face, shall be homogenized.

(3) means of attachment

The component is joined to the system's body structure by means of screw fastening in principle. Flanges or tabs are principally used along the component. The pitch of the mounting hole for the component with the generation of heat shall be determined not only considering mechanical environmental conditions, but also self-heating properties, because thermal contact resistance depends on the pitch of mounting hole.

The hole or clearance gap on the clamp face shall be prohibited from infiltrating the thermal filler, because sometimes thermal filler is applied during component installation. The system side will determine if the thermal filler is applied or not.

(4) surface roughness and flatness

The surface roughness and flatness factors shall be specified as necessary to heat transfer at the plane of union of the subject component.

The recommendation value is provided as follows.

<1> surface roughness : better than 1.6 Ra “JIS B 0601” (Ra : arithmetical mean roughness)

<2> flatness factor : under 0.001 mm/1mm

Flatness shall be calculated according to the following formula, and indicated by mark to ICD. But flatness factor shall be defined by rising with 0.05mm pitch.

(flatness) = (flatness factor) X (length of the longest part of contact surface)

(5) average heat generation density

The average heat generation density shall be under 0.06 W/cm^2 for contact surface of clamp face. But the definition of average heat generation density is as follows.

(average heat generation density) =

(maximum heat generation: note 1)/(contact area of clamp face)

Note 1 : except maximum heat generation within 15 minutes

When the component does not satisfy this rule, there shall be coordination with the system side and specified to ICD.

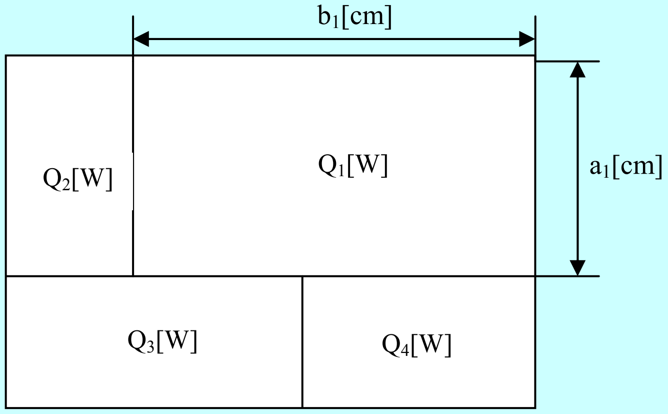
(6) distributed heat source

The deviation of heat generation density of component shall be under ±50% of average heat generation density. When the component does not satisfy this rule, there shall be coordination with the system side and specify to ICD. The definition of deviation of heat generation density is as follows.

(deviation of heat generation density) = (partial heat generation density – average heat generation density)/(average heat generation density)

The definition of partial heat generation density is as follows. (refer to Fig. 5.5)

(partial heat generation density) = (heat generation density corresponding to each node dividing clamp face of component in case of generation of thermal mathematical model of component)



$Q = Q_1+Q_2+Q_3+Q_4$

(Q is calorific value getting away from each node to satellite panel at each clamp face)

$q_i[W/cm^2]=Q_i/(a_i X b_i)$

Fig. 5.5 Definition of partial heat generation density

(7) shape of clamp face

The clamp face of component shall not be prepared for counterbore. However, if the component is under 0.06 W/cm^2 of average heat generation density and under ±50% of deviation of heat generation density, this rule shall not be applied.

(8) surface treatment of clamp face

The clamp face of each component shall not be part of the surface treatment bringing heat insulation such as painting. The surface treatment of the clamp face shall be determined by evaluating level of generated corrosion (electrolytic corrosion) etc. caused by contact with dissimilar metals.

(9) surface treatment of outer surface except clamp face

Emissivity of outer surface shall be over 0.8 of hemispherical emissivity in principle. In terms of area being restricted for surface treatment, the surface treatment of the range and and the hemispherical emissivity shall be described to ICD beforehand.

(10) other cases

In case there are any concerns or doubts regarding the requirements for internal onboard equipment, the system side and the component side shall coordinate with each other.

5.3.4.2 Requirement for exposed onboard equipment

The special feature of exposed onboard equipment is its requirement to function and perform outside of the spacecraft exposed to outer space,, usually for observation rather than to perform a thermal control function. The degree of freedom for its mounting position is relatively limited. Occasionally a suitable arrangement for thermal control purposes becomes an unsuitable position for observation purposes. Most importantly, however, it is directly exposed to the hard space thermal environment.

The thermal control of exposed onboard equipment must perform taking into account a balance of the space thermal environment and self heating. Of primary importance is the design of the external form and the suitableness of the position of primary thermal control face. For this reason, the design of the internal structure for all exposed onboard equipment shall be started at first.

The thermal control for the exposed onboard equipment often becomes untenable because of the rules affecting the clamp face and the easy adjustment of thermo-optical properties such as the internal onboard equipment. Sometimes there is a complicated outer surface shape of exposed onboard equipment, or because of the mounting position of exposed onboard equipment and system, the multiple reflection of external heat input exists between the external surfaces of the component or between the component and the system.

Therefore, it is necessary that the exposed onboard equipment side shall be designed to fit in and around the thermal environment condition in the thermal design of the exposed onboard equipment.

In principal, the thermal design shall be implemented according to the following standards based on thermal interface condition document defined at 5.3.1 (1) and prepared by the system side.

(1) volume of heat exchange

The exposed onboard equipment shall be insulated against radiation and and conduction within system's structure. However, the volume of heat exchanged shall be defined using a suitable value by the system side.

(2) surface reflection properties

The surface shall diffuse solar light, albedo and earth and moon/planet infrared radiation shine to the utmost. Unless there is a process for radiation treatment with OSR etc., this shall be excluded.

(3) heater for temperature control

In principle, when there is thermal controlling of exposed onboard equipment with a heater, s the exposed onboard equipment shall have its own heater with a control circuit. This will especially be the case when the temperature directly effects the efficiency of the component (for example, when the temperature dependence of efficiency is large, and there is interfere with the unit's efficiency without a control circuit), the control circuit shall nevertheless be included.

However, when the thermal control or operation of the component is not considered an obstacle, it may be possible to request the heater control function from the system side. The system side may sometimes require mounting a survivable heater (including thermostat) or replacement heater (i.e. a heater that generates heat equally to the component) for the component side if necessary.

5.3.5 Interface with propulsion system

It is necessary to address the propulsion system for the thermal interface, with the interface requirements defined below.

5.3.5.1 Attitude control system

Mainly the attitude control system is composed by following component.

- Attitude control system
  - Thruster module
  - Propellant feed system
    - Propellant tank
    - Plumbing
    - Filter
    - Fill and drain valve
    - Isolation valve
  - Measurement equipment

The following shall be considered as thermal interface items between spacecraft system and attitude control system

・ thermal characteristic value of component
・ allowable temperature range of component
・ caloric value of component
・ electric power for thermal control of component
・ thermal environmental conditions of component

(1) thruster module

Because the thruster module generates so much heat and is affects the heat of a spacecraft both outside and inside it, the interface information shall be provided to spacecraft system side with the thermal mathematical model if necessary. The following shall be considered as thermal interface items for the thruster module.

<1> thermal characteristic value : defined using the thermal mathematical model
<2> allowable temperature range : to be defined for component classified without injection (at inert and at stand by), at injection and at soak back heating

<3> caloric value : to be defined using heat input (at injection, at no injection and at soak back heating) to spacecraft system
<4> electric power value for thermal control
<5> thermal environmental conditions : heat input from external environment

(2) propellant feed system

Generally the components of the propellant feed system do not generate heat and is thermally controlled by a heater to keep it within allowable temperature range of propellants. The following shall be considered as thermal interface items of the propellant feed system.

<1> thermal characteristic value : to be defined by the area of the base plate, materials, flatness, surface roughness, geometries, infrared emissivity of outer surface and heat capacity to each component

<2> allowable temperature range : to be defined by the propellant and each component

<3> electric power value for thermal control

<4> thermal environmental conditions : to be defined by the inner temperature of the spacecraft system surrounding the component with the propellant feed system

Every definition shall be considered under the baseline of worst high temperature case and worst low temperature case for the propellant system.

5.3.5.2 Large-sized liquid thruster

The following items shall be considered as the thermal interface of the large-sized liquid thruster (to be called liquid engine hereafter) such as a thruster used for orbital transformation and the apogee engine.

<1> radiation heat input from liquid engine to spacecraft

<2> conduction heat input from liquid engine to spacecraft

<3> convection heat input from liquid engine to spacecraft

<4> allowable temperature range

<5> electric power for thermal control

<6> contamination

The outline of each item is explained as follows.

(a) radiation heat input from liquid engine to spacecraft

The heat flux flowing into the spacecraft by the combustion of the liquid engine shall be defined by thermal mathematical model, if necessary.

(b) conduction heat input from liquid engine to spacecraft

The heat flux flowing into the spacecraft through an attached portion of the spacecraft during the liquid engine firing and after engine firing stops, shall be defined by the thermal conductivity and temperature change at the attached portion of the spacecraft. using the thermal mathematical model, if necessary.

(c) convection heat input from liquid engine to spacecraft

The convection heating from the plume (backflow) from the combustion of the liquid engine, shall be defined, if necessary. Plume shield shall be adjusted, if necessary.

(d) allowable temperature range

The allowable temperature range of the component containing the liquid engine shall be required to have a thermal control design.

(e) electric power for thermal control

The electric power shall be set up based on the worst low temperature prior to the firing of the liquid engine. The elements are components for thermal control by heater.

- propellant valve
- injector
- others

(f) contamination

Contamination from the exhaust plume of a liquid engine shall be defined, if necessary.

5.3.5.3 Ion engine system

Ion engine system is one of the electrical propulsion systems, and the composition is described below.

- Ion engine system
  - Ion engine thruster
  - Power supply device
  - Propellant storage and feed system
  - Valve driving electronics
  - Thruster controller

It is necessary to consider the following thermal interface items between spacecraft system and ion engine system.

<1> thermal characteristic value or thermal mathematical model of the component
<2> allowable temperature range of the component
<3> external heat input to the component
<4> calorific value and heat soak-back of the component
<5> electric power for the thermal control of the component

The thermal interface of the component for the ion engine system is described below.

(1) ion engine thruster

The ion engine thruster (called “thruster” hereafter) is a component which generates high amount of heat that is exposed to outer space. Because it is basically isolated from the spacecraft, it has its own independent temperature control. The radiation on the external surface of spacecraft, the presence of heat from incidental solar light, and heat soak-back etc. to the spacecraft through the clamp face are important parameters of thermal control design for the spacecraft and the thruster. The interface information is provided to the spacecraft system side with a thermal mathematical model, if necessary. The following shall be necessary to consider as the thermal interface items for the thruster.

<1> thermal characteristic value
<2> node tearing diagram
<3> node heat capacity and calorific value
<4> conduction resistance and radiation resistance between nodes
<5> external heat input : amount of external heat input for each node
<6> thermal control heater : lower/upper limit temperature of heater calorific value
<7> allowable temperature : allowable temperature range at inactive/turn-on/active
<8> amount of heat soak back : amount of conduction/radiation soak-back

(2) power supply device, thruster controller and valve driving electronics

The main feature of this power supply component is its generation of high amounts of heat and heat density, and the large change of calorific value when it is active versus when it is inactive. In spite of above features, because the power supply device is located inside of spacecraft, it is necessary to consider an equal thermal interface of these items to the other equipment on board the spacecraft. The thruster controller and valve driving electronics have relatively small calorific value and it is possible to handle it in the same manner as the other equipment on board the spacecraft. Therefore, the thermal interface items of these types of components can be treated the same way as the items described in section 5.3.2.2.

(3) propellant storage and feed system

The propellant storage and feed system stores the propellant and supplies the thruster. The structure of the system, therefore, is different according to the kind of propellant used. In the interface to the spacecraft, usually heat insulation to spacecraft is processed and thermal control by the heater is performed, especially for the thermal control at low temperature. The following shall be considered as the thermal interface.

<1> thermal characteristic value : heat capacity of parts, area of attached portion, materials, flatness, surface roughness, shape of parts, level, dimensions, infrared emissivity of the surface and contact heat transfer coefficient.

<2> heat generation distribution : heat generation distribution of parts, heat generation profile

<3> active thermal control : controllable lower/upper limit and heat generation distribution of thermal control heater

<4> allowable temperature : allowable temperature range when it is active and when it is inactive (definition of temperature specified point is necessary)

5.3.6 Thermal interface between system and launch vehicle

The following items shall be considered regarding the thermal interface between the spacecraft and the launch vehicle.

<1> air conditioned temperature inside of the fairing at launch site
<2> radiation heat from fairing
<3> free molecular flow heating (after fairing separation)
<4> heat conduction and radiation heat at separation section of the spacecraft
<5> decompression profile in fairing

(1) air conditioned temperature inside of fairing at launch site
Thermal environment in the fairing shall be controlled by an air conditioner while the spacecraft is encapsulated into the fairing.

(2) free molecular flow heating
It shall be set up using the free molecular flow heating rate after the fairing separation.

(3) heat conduction and radiation heat at separation section of the spacecraft
It shall be set up with temperature history, heat transfer coefficient and emissivity of each section of the spacecraft separation part from launch to spacecraft separation.

(4) decompression profile in the fairing

It is necessary to define the decompression profile in the fairing, considering to prevent the expansion and peeling of MLI.

5.4 Thermal control device/material technology

5.4.1 Thermal control technology

It is necessary to select the most suitable method of thermal control technology for the spacecraft. The technology to maintain the spacecraft at a suitable temperature is called thermal control technology.

The requirements for the thermal control system is mentioned in section 5.4.1.1below, and of the various types of thermal control methods are located in 5.4.1.2.

5.4.1.1 The requirements for the thermal control system

The function of thermal control system of the spacecraft is to maintain the component of spacecraft within a specified temperature range and temperature rate change, from launch and throughout its entire mission. The general requirements from the spacecraft system side regarding mass, electrical power and reliability for the thermal control system are described below.

(1) to be light weight and have a simple composition
(2) to be small utilizing the necessary electric power to the utmost
(3) to have stable performance and high reliability under the harsh environment of space, accounting such things as radial rays, ultra violet rays, atomic oxygen, high vacuum, and high and low temperatures.

Fig. 5.4 shows the general classification of the thermal control device and material to satisfy above requirement. The thermal control device/material is classified roughly into passive type thermal control material, such as coating and multi layer insulation (MLI) etc. and active type thermal control device, such as a heater, a heat pipe or a thermal louver etc.

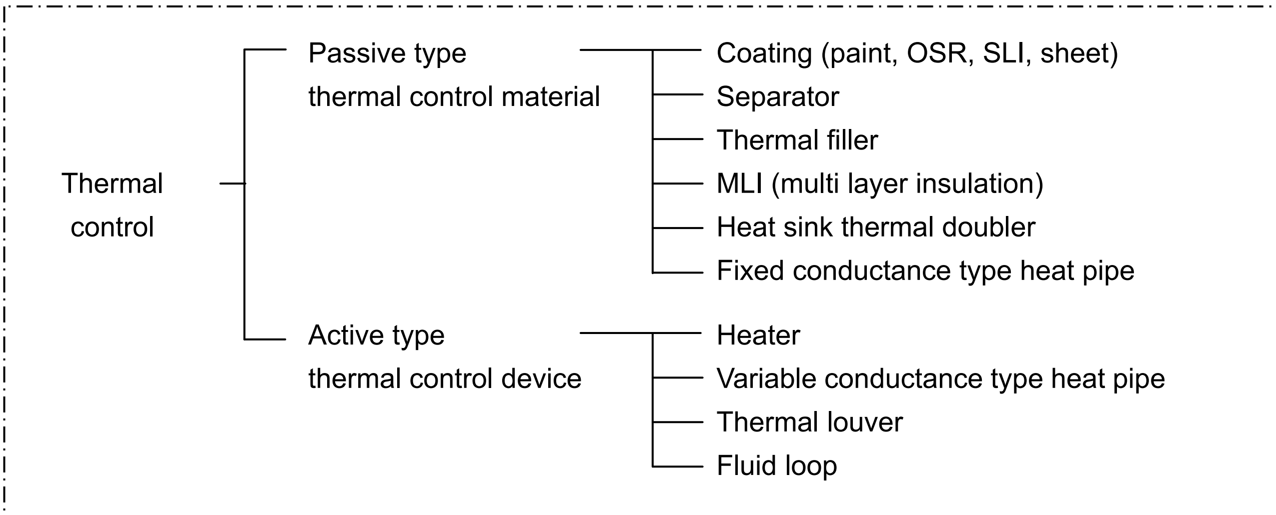


Fig. 5.4 Classification of thermal control device/material

5.4.1.2 Selection of thermal control method

The thermal control methods for spacecraft are classified into two types - the passive type and the active type based on the thermal control device or material, which comprise the thermal control system. The passive type thermal control method is the system designed to control the temperature of the component by coordinating the route of radiation and conduction with the physical characteristics (such as the shape) that are inherent in the material. Specifically, despite the change to its physical property, it has high reliability. Additionally, the heat input on orbit and the change of calorific value directly affects the temperature change of the component. The physical property hardly changes the temperature as compared to heat input on orbit or the change of calorific value. The active type thermal control method is a system to control temperature of the component using a technology which has mechanical and/or moving parts or by using fluid, and which utilizes the change of heat with electric energy. The typical thermal control device used for the active type thermal control method is a heater, thermal louver, variable conductance type heat pipe, or a fluid loop.

In spite of the attitude control method (spin or tri-axial control) of a spacecraft, the passive type thermal control material such as coating or multi layer insulation is considered basic, while the active type thermal control device is complex and used for a component with extreme requirements of to maintain the temperature at a suitable thermal control range. The following describe the various kinds of passive type thermal control materials and active type thermal control devices.

[passive type thermal control material]

(1) coating (paint, OSR, SLI and sheet)

- Used on the outside of spacecraft structure, with white paint for heat insulation control
- Used on the inside of spacecraft structure, with black paint for homogenization of temperature in the interior of the structure
- Used on the radiating surface, with glass system rigid OSR, macro-molecular system flexible OSR and Teflon with silver deposits etc.
- Used as thermal control tape with adhesives on film or sheet etc.

(2) separator

- Used as heat insulation control between spacecraft structure and component etc.
- Used with materials with low thermal conductivity

(3) thermal filler

- Used for heat transfer control between spacecraft structure and component etc.
- Used for materials with high thermal conductivity

(4) MLI

- Used for heat insulation control of spacecraft structure
- polyimide and Beta-cloth® etc. used as an external layer

(5) heat-sink thermal doubler

- Used for thermal diffusion of high heat element

(6) fixed conductance type heat pipe

・Used for transportation of heat exhaust from onboard equipment with high calorific value or heat generation density, such as relay equipment for telecommunication etc.

・ Used for homogenization of temperature of the spacecraft structure or panel for onboard equipment

・ Used for development of heat radiation capability by heat pipe and radiator

[active type thermal control device]

(1) heater

・ Used for thermal control of thermally critical component such as attitude control system and battery

・ heater generating heat similar to component when the component is OFF

(2) variable conductance type heat pipe

・ thermal control of high heat onboard equipment used to maintain within a fixed and narrow temperature range

(3) thermal louver

・ Used for thermal control of spacecraft, external heat input changes for a relatively short period such as a satellite in low earth orbit

・ Used to control heat discharge from spacecraft, calorific value of internal onboard equipment changes for a relatively short period

・ thermal control of component used to maintain temperature in narrow temperature range

(4) fluid loop

Thermal control when long-distance thermal transportation is required, when the ON/OFF of thermal transfer function is required, when gravity must be controlled during the ground test or when flexibility of exhaust heat route is required.

5.4.2 Thermal control device/material

The criteria for selection and points to note regarding thermal control devices and/or materials and its processes are described below.

5.4.2.1 Criteria for selection of thermal control devices and/or material

The thermal control device and/or material must satisfy the requirements for performance, physical properties, reliability and environmental resistance for both the component and the spacecraft and are generally selected according to following sequence.

(1) recommended parts/entry item in list of material by public institution (such as JAXA,NASA and ESA etc.).

(2) material produced domestically with an established evaluation for space use for JAXA or with results, an/or with the possibility of getting technical documents confirming their capabilities.

(3) parts/material guaranteed or recommended by foreign spacecraft manufacturers or companies for which JAXA has confirmed its quality.
(4) parts/material with a newly prepared specification (or technical documents) and guaranteed quality and reliability have been conducted through evaluation and tests etc.

5.4.2.2 Criterion for selection of process

The process for the selection of guaranteed thermal control parts and materials shall be repeated as many times as is necessary to satisfy the performance, physical property, reliability and environmental resistance requirements, and will be generally selected according to sequence below.

(1) Selection according to standardized specification (or procedure manual) from a public institution
(2) Selection according to the already established specifications for space use
(3) Selection according to a newly prepared specification and guaranteed repeatability by evaluation and tests etc.

5.4.2.3 Points to note for applying the thermal control device/material process

The special points to note for the thermal control device/material process for a component and spacecraft are described below.

(1) traceability

The parts/material and process applied to spacecraft (PM, PFM and FM) shall include a traceability of parts, material and its process by lot number etc. defined by parts/material specification and process specification or procedure manual.

(2) outgas

When outgas is expelled from the thermal control device/material, it clings to the surface of a sensor and radiating face, causes characteristic degradation by pollution (contamination). Therefore, the expulsion of outgas from the thermal control device/material must occur as little as possible in space. Generally, the following value is applied as the outgas requirement of parts/material. (refer to JMR-010 Standard for Contamination Control)

(a) total mass loss (TML)・・・・・・・・・・under 1%
(b) collected volatile condensable materials (CVCM)・・・・・・・・・・under 0.1%

(3)cleanliness

The thermal control device/material for spacecraft use shall be controlled for cleanliness in all aspects of manufacturing/integration, testing, transportation, storage and pre-launch operations. The target of cleanliness control includes the ambient environment of where the spacecraft is placed as well as the surface of thermal control parts/material.

(a) cleanliness in clean room

The requirement for cleanliness during manufacturing/integration shall be generally over ISO class 8 (equivalent to FED-STD-209E class 100,000).

(b) surface cleanliness of thermal control device/material

The surface cleanliness shall be defined by the size and number of particulate residual material and mass (mg) of non-volatile residue, removed from area of the thermal control device/material per 1 square feet by solvent. (ISO14952-2)

Requirement for above (a) and (b) shall be defined for each spacecraft according to thermal control parts/material and work-in-progress.

(4) cleaning

When the surface of the thermal control device/material becomes contaminated, cleaning is performed as a part of cleanliness control. At this time, consideration for cleanliness of the cleaning tool and solvent, as well as the compatibility of the solvent for surface of thermal control parts/material etc., shall be considered.

(5) refurbishment

Thermal control device/material of the spacecraft, damaged or contaminated by handling after testing or before launch, shall be refurbished. However, the refurbishment shall not damage the effectiveness of the design and tests that have been performed until that time.

(6) radiation-resistant characteristic

The surface characteristics (thermo-optical properties and light transmittance) and mechanical strength of the thermal control device/material deteriorates because of age and because of radial rays (mainly electrons and protons), ultraviolet rays and the heat cycle. The surface characteristics of the thermal control parts/material are required to have minimal deterioration due to age and be of stable quality. For thermal control devices/materials of the spacecraft projected to orbit at a low altitude (200-700 km) above earth, extra care shall be taken because of expected degradation from atomic oxygen.

(7) environmental characteristics for heat resistance

The surface temperature of antenna and solar panel exposed in outer space, fluctuates between about -160 ~ 100℃. Therefore, the mechanical properties of the thermal control materials used for the antenna and panel, such as paint and adhesive agents are not desirable because they cannot significantly affect this wide temperature range. Though paint and adhesive agents are classified roughly into categories such as epoxy system, acrylic, urethane system, silicon system and inorganic system, they shall be used according to a usable environment (operating temperature range) and according to the strength of their adhesive bonding.

(8) grounding

When the integration of thermal control device/material to the spacecraft occurs, the thermal control parts/material shall be grounded to the spacecraft structure to prevent electrification or an electric discharge, as well as to avoid electromagnetic interference. The steps to be taken for grounding is shown below.

(a) conductive coating or conductive paint

(b) conductive adhesive

(c) ground strap

There will be a requirement for grounding resistance which shall be defined individually for each spacecraft.

(9) service life

The service life of paint, adhesive and tape etc. is limited, and therefore must meet or exceed the service life of the spacecraft. A longer service life is desirable. .

(10) corrosion protection

Beryllium, silver, aluminum and magnesium are necessary and suitable treatments for corrosion protection.

(11) metals with significant outgas

As cadmium and zinc hold relatively high levels of outgas, these metals shall not be used for space parts/materials.

(12) germanium

As germanium degrades easily from moisture in the air, the temperature in the clean room shall be carefully controlled and special measures for storage shall be taken

(13) ITO

When ITO (indium-tin oxide) coating is applied, it shall be carefully handled so as not to damage the ITO coating, which is used for securing conductivity (or resistance to atomic oxygen) on the surface of the thermal control material. Because ITO is transparent and any damage is difficult to detect, careful attention is required, to prevent abrasion or bending. Also, because it has a low adhesion quality with materials containing fluorine such as Teflon with silver deposits it can easily be peeled off.

5.4.2.4 Production of thermal control parts/material

The system side and on board component side shall conduct coordination and manufacturing of hardware for the thermal control parts/materials as necessary for thermal control.

5.4.2.5 Handling of MLI

(1) designed to prevent MLI from peeling off

For the purpose of preventing MLI from peeling off, the design of MLI shall be conducted according to section 2.1 (1) “JERG-2-311 Design Standard to prevent peeling of MLI”.

(2) precautions regarding the design of MLI

The system and onboard equipment shall be designed with consideration for both experiencing a loss of interface with MLI. When the MLI must be installed over an interface demarcation point, intervention by the system side and by the on board component side shall be started as quickly as possible, because it may be a complicated procedure and may take significant amount of time. The result of intervention shall be described to ICD. In principle, ground handling of MLI shall be performed based on section 2.1 (2) “JERG-2-211 Design Standard for electrical charge/discharge”.

5.5 Thermal design verification

The thermal design shall be verified as conforming to the requirement of specified performance, as described below.

(1) Thermal analysis or test shall verify that the results of the design adapts to the required function and performance.

(2) Though verification by test is desirable, there must be an explanation if another method is used for verification.

(3) Verification shall be performed to the component and the spacecraft system.

5.5.1 Verification by thermal analysis

In the verification by thermal analysis, conformity to the design requirement shall be verified using suitable analysis modeling and performance prediction.

In the verification by thermal analysis, every condition of test environment and test configuration restricted by constraint of test facility etc. shall be included.

5.5.2 Verification by test

Verification by test means verification of conformity to the design requirement, placing the spacecraft system and components under the conditions which re-create the actual environment that is expected. The demonstration of requirement for thermal control design is called “thermal balance test”, and the demonstration of requirement for behavior of spacecraft is called “thermal vacuum test”. Classification of these tests is shown in section 5.5.2.1.

During testing, the test condition shall be similar to the acceptance level or qualification level, depending upon its required verification level.

5.5.2.1 Classification of tests

Thermal vacuum tests of spacecraft are classified into the thermal balance test which is performed simulating the thermal environment on orbit for the purpose of confirming the appropriateness of thermal control design of the spacecraft system and components, and the thermal vacuum test, which is performed for the narrower purpose of proving that the spacecraft system and components can function and perform under the thermal vacuum condition.

- ■ thermal vacuum test
  - ■ thermal balance test
  - ■ thermal vacuum test
    (thermal vacuum test in narrow sense)

The standard regarding the thermal vacuum test of spacecraft shall follow 2.1 (3) “JERG-2-002 Standard of General Tests for Spacecrafts”, to establish the thermal control design standard, while the thermal balance test is performed for the purpose of validating the thermal control system.

5.5.2.2 Approach regarding the selection of thermal balance test conditions

Although the thermal balance test is conducted to confirm the validity of the thermal control design and the performance of thermal control hardware, also it is also used to confirm the validity of thermal mathematical model applied during the design phase. The following shall be generally be considered for the selection of test conditions.

(a) Using a test condition so that the thermal control design that can be easily evaluated

・ including spacecraft configuration and thermal environmental conditions during its operation phase

・ including worst conditions concerning both the external and internal thermal conditions

・ including thermally critical conditions which are peculiar to the specific spacecraft

・ including suitability during steady state/non-steady state test conditions

(b) Using a test condition that thermal control hardware that can be easily evaluated

・ including being able to evaluate the performance of insulation, coating, heat sink, thermal louver, heat pipe and heater etc.

(c) Using a test condition that thermal mathematical model that can be easily evaluated

・ including being able to evaluate uncertain parameters of the thermal mathematical model

(d) Using a test condition to confirm external heat input during a thermal vacuum test

・ including being able to validate external heat input conditions by setting up at thermal vacuum test in a narrow sense

5.5.2.3 Procedure and selection of thermal balance test

The thermal vacuum test is conducted by placing the spacecraft system and components into a space chamber and simulating the high vacuum low temperature environment of outer space using the gas exhaust in the space chamber and cooling of the shroud. The environmental condition in space chamber during the thermal balance test, shall be set according to the value shown in Table 5.5.2 in principal.

Table 5.5.2 Environmental condition in space chamber at thermal vacuum test

Item	Value
Pressure	Under 1.3 X 10^{-3} Pa *1 (1X10^{-5} Torr)
Shroud Temperature	Under 100 K (-173 ℃)

Note : *1 for the purpose to simulate heat transfer appearance in outer space

A thermal sensor shall be installed at each section of the spacecraft system and component to monitor the temperature and provide thermal data. The onboard heat input simulation during the thermal balance test shall be performed according to the heating method described below.

(1) solar method
(2) IR method (IR lamp method, IR panel method and IR shroud method)
(3) skin heater method
(4) A combination of the above three methods

The irradiation intensity or heat input by these heating methods shall be monitored using a radiation intensity meter or calorie meter or wattmeter etc. The heating method to test the spacecraft during the thermal balance test shall be set up as the most suitable method according to purpose of test, and considering the following.

(1) simulation accuracy is necessary (especially for thermal balance test)
(2) the influence of gravity (for testing with a heat pipe etc., the properties of which change according to the direction of gravity)
(3) Consideration for the influence on the spacecraft etc. according to the establishment of test configuration (e.g.in cases when a skin heater is installed on a spacecraft directly etc.)
(4) facility capability
(5) cost for test

5.5.2.4 Evaluation items

For the purpose of an evaluation to be performed after thermal balance test of the spacecraft system etc., following items and contents shall at least be included.

Regarding the sequence of an evaluation for the thermal balance test, there is first an evaluation performed for the setting up of test conditions, while there is a continuous performance evaluation of the thermal control hardware and an evaluation of thermal mathematical model as follows.

(a) evaluation of setting up of test condition

・ evaluation of environment of the space chamber (degree of vacuum and temperature of shroud)
・ evaluation of set up of external heat input (including heat input from jig)
・ evaluation of set up of internal heat input
・ evaluation of set up of boundary temperature
・ evaluation of measurement system (temperature, degree of vacuum and external heat input)

(b) performance evaluation of the thermal control hardware

・ performance evaluation of the thermal control hardware such as insulation, coating, heat sink, thermal louver, heat pipe and heater etc.

(c) evaluation of the thermal mathematical model

・ evaluation of the thermal mathematical model by comparing the estimated to the actually measured temperature
・ evaluation of uncertain parameters of thermal mathematical model

(d) evaluation of the thermal design

・ direct thermal design evaluation from the measured temperature when test condition is equal

・ thermal design evaluation from flight prediction temperature by evaluated thermal mathematical model, when the test condition is different from flight condition

(e) evaluation of thermal vacuum test (narrowly defined)

・ validation of external heat input condition

(f) evaluation of contamination

・ evaluation of the existence of a contamination occurrence

・ evaluation of the contamination source

・ evaluation of the influence of contamination on the spacecraft