General

JERG-2-215A

DESIGN STANDARD

SOLAR ARRAY PADDLE

Mar　27,2013　RevA

Japan Aerospace Exploration Agency

This is an English translation of JERG-2-215A. Whenever there is anything ambiguous in this document, the original document (the Japanese version) shall be used to clarify the intent of the requirement.

Published by

Japan Aerospace Exploration Agency

Safety and Mission Assurance Department

2-1-1 Sengen Tsukuba-shi,Ibaraki 305-8505, Japan

-Contents-

1. Scope

This design standard is specified to be applied to the lightweight, rigid-type solar array wing system of the spacecraft developed by the Japan Aerospace Exploration Agency (JAXA). The content of this design standard specifies the key requirements to be applied to the design and development of the lightweight, rigid-type solar array wing system.

2. Applicable documents

JERG-2-141	Space Environment Standard
JERG-2-143	Radiation Resistance Design Standard
JERG-2-144	Space Debris Protective Design Standard
JERG-2-152	Disturbance Control Standard
JERG-2-200	Electric Design Standard
JERG-2-211	Charging/Discharging Design Standard
JERG-2-212	Wire Derating Design Standard
JERG-2-213	Insulation Design Standard
JERG-2-214	Power Supply System Design Standard
JERG-2-310	Thermal Control System Design Standard

3. Terminology, Definition and Abbreviation

Shown in Appendix **I**.

4. General requirements

4.1 General

The spacecraft solar array wing system converts solar light into electrical energy, and most of the functional portions are placed outside the spacecraft's structure. The system must generate electricity required by the spacecraft over the entire mission period while being exposed to "thermal and mechanical environment during launch and ascent" from launch to spacecraft separation and a "hostile space environment" after separation for an extended period of time.

4.2 Composition and components

A lightweight, rigid-type solar array wing is a solar array wing having an aluminum honeycomb sandwich structure covered by CFRP as the substrate and a solar cell panel with a solar array affixed as the main component. Figure 4-1 shows an example of the general composition of a lightweight, rigid-type solar array wing system.

Because the necessity of each component differs depending on the mission requirements and power generation requirements of the spacecraft, the necessary function configuration shall be specified in accordance with the mission requirements of each spacecraft in the course of development. For a large solar array wing system requiring a large amount of power, it is difficult in some cases to verify the entire paddle system due to restrictions of the ground test equipment. In the verification of this development, in addition to performing a series of verifications on each component, it is necessary to create a proper development plan and verification plan according to the solar array wing system configuration and check the validity of the design and the soundness of the flight products.

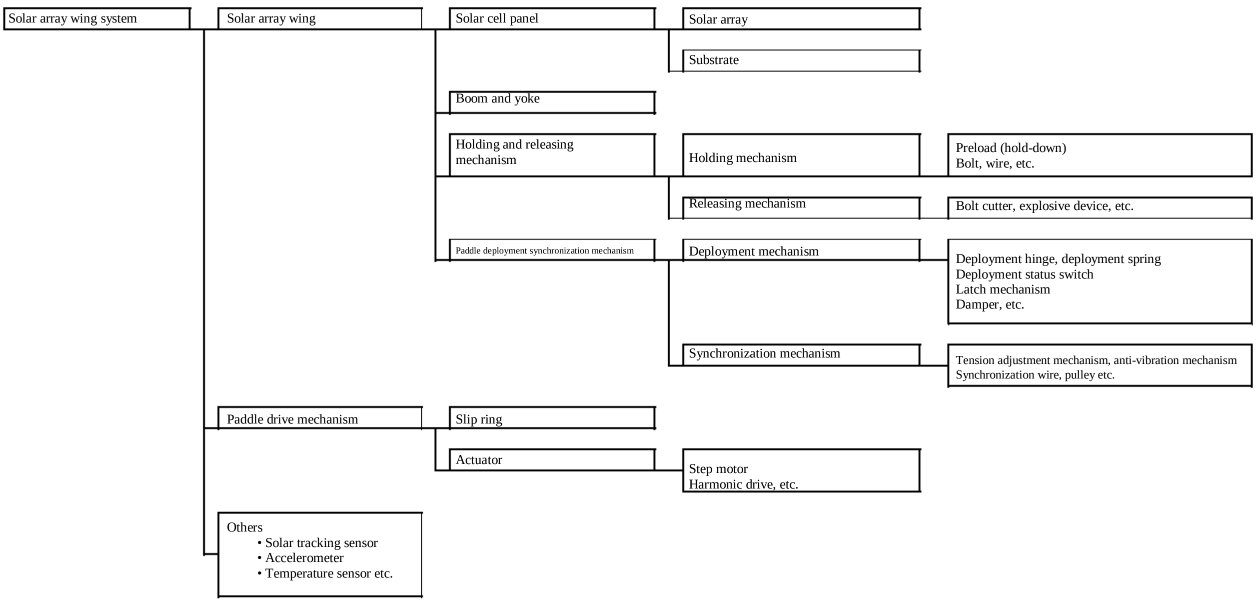


Figure 4-1 Example of lightweight, rigid-type solar array wing system configuration

5. Subsystem design and analysis

5.1 Electric design and analysis

This section describes the design points to note in solar array wing electric design and analysis.

5.1.1 Power generation design

In power generation design, the power required by the system shall be satisfied throughout the mission period with particular attention paid to the following design items and design contents.

(1) Selection of solar cell type and review of array circuit configuration (number of series-parallel circuits)

(2) In reviewing the number of series-parallel circuits, consideration shall be given to the influence of series resistance, temperature characteristics, deterioration due to influence of space environment such as radiation and cover glass, adhesive loss and the like in performing the power generation analysis.

Figure 5-1 shows an abbreviated flow of power generation analysis.

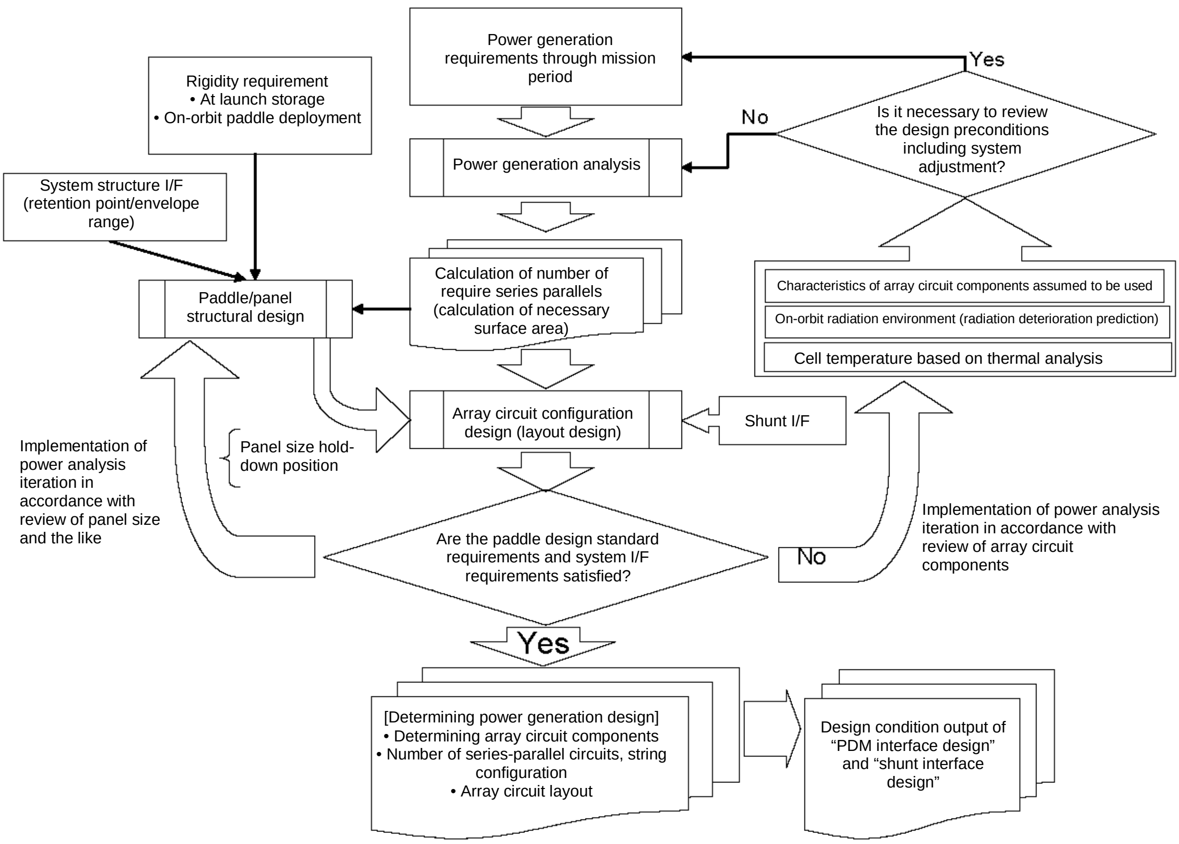


Figure 5-1 Abbreviated flow of power generation design

5.1.1.1 Array circuit component design

The following contents shall be clarified in designing the array circuit components.

(1) Solar cell

The solar cell appearance and following specifications shall be clarified. When a dedicated charge array is provided, the specifications shall be clarified.

(a) Cell type

(b) Output parameter (short circuit current Isc, open voltage Voc, maximum power current Imp, maximum power voltage Vmp)

For the maximum power current Imp, the average minimum load current (minimum average) in the bare cell shall be clarified in consideration of the cell grade.

(c) Power generation efficiency in BOL and EOL

(d) Cell size (length × width × thickness)

(e) Average mass

(2) Cover glass

The cover glass appearance and the following specifications shall be clarified.

(a) Glass type (model name)

(b) Substrate material

(c) Coating

(d) Glass size (length × width × thickness)

(e) Average mass

(f) Solar light transmittance and UV deterioration rate

(3) Interconnector and bus bar

The interconnector, bus bar appearance, shape and the following specifications shall be clarified. In manufacturing the interconnector and bus bar, design instructions shall be given so that no burrs are left in the edge.

(a) Material

(b) Thickness

(c) Average mass

(4) Bypass diode for shadow countermeasures

The appearance of a bypass diode for shadow countermeasures and the following specifications shall be clarified.

(a) Forward voltage Vf
(b) Reverse current Ir
(c) Shape
(d) Average mass

(5) CIC (Cover Integrated solar Cell)

Out of the above-mentioned solar cell circuit components, solar cells with the interconnector welded and the cover glass adhered are generally called CIC. The following specifications shall be clarified with respect to such assemblies.

(a) Interconnector weld strength (45 degree peel strength)
(b) Surface characteristics under CIC state (α, ε)
(c) Output grade classification under CIC state

5.1.1.2 Power generation analysis

The power generation specified point shall be the paddle drive mechanism output terminal. The power generation at the interface voltage with the system shall be analyzed.

Based on the result of thermal analysis and radiation analysis, the required number of series-parallel circuits shall be designed after indicating the V-I curve characteristics of the entire solar cell array circuit satisfying the power required by the system throughout the mission period. In addition, the trend of power generation between BOL and EOL in each season (spring, summer, fall and winter) shall be clarified.

In power generation analysis, the solar cell output parameters shall be considered properly as a precondition for analysis in accordance with the development phase.

- Design phase: Consideration shall be given to the solar cell output distribution indicated in the solar cell application data sheet and the like and the output grade distribution in CIC state accordingly.
- Manufacturing and test phase: Reflection of array circuit output characteristics measured in an irradiation test.

In addition to the above-mentioned analysis preconditions, the following analysis parameters shall be clarified before performing power generation analysis.

(1) Solar light strength

The details of AM0 solar light strength (solar constant) at one astronomical unit (1 AU = average distance between the earth and the sun), which shall be in accordance with JERG-2-141 Space Environment Standard, are given below as reference.

[1] Solar light strength = 1336 W/m^2@1AU

[2] Uncertainty of solar light strength (including variation due to the number of sunspots and measurement errors) = ±10 W/m^2

[3] The approximate values of seasonal variation of solar radiation strength due to changes in the distance between the sun and the earth (excluding uncertainty of the above-mentioned [2]) are shown in Table 5-1. The error (1σ) in the table is 0.6 W/m^2.

Thus, the solar light strength used in power generation analysis shall be roughly expressed in the formula below:

Solar light strength max/min = (1366±10 W/m^2) × (seasonal variation of the sun-earth distance change +3.36%/-3.22%)

Table 5-1 Approximate value of seasonal variation of solar radiation strength

Month	Solar radiation strength W/m^2
January 3 (perihelion)	1412
February	1406
March	1391
April	1368
May	1345
June	1329
July 4 (aphelion)	1322
August	1326
September	1342
October	1363
November	1387
December	1405

(2) Solar light incident angle

In the case of a static earth-orbiting spacecraft, a solar light incident angle of ±23.4 degrees shall be taken into consideration. As for the polar-orbit spacecraft, the cant angle shall be taken into consideration at the cant paddle at the local time of descending node passage.

(3) Paddle tracking angle error

Consideration shall be given to the tracking error estimated as the base of analysis.

(4) Solar cell temperature
Consideration shall be given to the temperature prediction error margin estimated with respect to the orbital nominal prediction temperature obtained from the thermal analysis result.
See the technical data for the estimated temperature prediction error margin track record of earth-orbiting spacecraft.
The temperature prediction error margin corresponds to the uncertainty of design and equal to <maximum prediction temperature range> - <design prediction temperature range>. Normally, a temperature prediction error margin of 10°C or more is used. In the case of use in a similar environment with an orbital track record, a proper temperature margin shall be specified based on the technical judgment of the project

(5) Preservation ratio of solar cell output parameters in accordance with radiation exposure

(6) Solar cell each output parameter temperature coefficient
Consideration shall be given to the temperature coefficient in accordance with radiation exposure.

(7) Cover loss and ultraviolet radiation deterioration rate

(8) Line drop (harness resistance)
Consideration shall be given to the isolation diode, interconnector resistance and their temperature characteristics.

(9) Measurement error of paddle V-I characteristics
Consideration shall be given to the light source setting error, illumination distribution error and measurement error as error factors of V-I characteristics measurement using the solar simulator.

(10) Yaw steering and positional error
Consideration shall be given to the attitude angle profile in accordance with the yaw steering and the positional error with respect to the profile.

(11) Paddle having a reflecting plate or condensing plate

For a paddle having a reflecting plate or condensing plate, consideration shall be given to the reflection and condensing efficiency.

(12) Shadow influence

When the spacecraft structure or loaded objects cast a shadow on the paddle cell surface, power analysis shall be conducted in consideration of the influence of that shadow.

When adopting a solar cell having an IBF (Integrated Bypass diode Function) function, care shall be taken not to allow reverse bias calorification due to the influence of shadow (especially in the solar cell range where a shadow is cast) to cause a high-temperature state exceeding the allowable range of the solar cell and substrate.

Additionally, consideration shall be given to the following factors as uncertain errors/margins.

(a) Solar cell circuit random failure
About 5% of the power generation or one-array circuit, whichever is larger, is recommended as the random failure margin due to space debris.
Refer to the technical data for the track record of power generation variation in earth-orbiting spacecraft.

(b) Loss due to ESD
The details shall be in accordance with JERG-2-211Charging/Discharging Design Standard and the influence of ESD (Electrical Static Discharge) shall be reflected in the power generation analysis based on the following review.
i) The discharge occurrence frequency during the mission period shall be analyzed.
ii) The power generation deterioration rate shall be grasped by conducting a coupon panel test as needed.
iii) The power generation deterioration rate shall be analyzed from the above-mentioned two conditions.

5.1.1.3 Array circuit configuration (layout) design

After determining the required area as the panel based on the number of series-parallel circuits obtained from the power generation analysis, an array layout shall be designed in consideration of the following points.

(1) The number of parallel circuits in one-array circuit
Array circuit (parallel array circuit family corresponding to one shunt) configuration in the range satisfying I/F with the power supply system shunt element. The one-array circuit current upper limit for avoiding sustaining discharge shall be in accordance with the requirements specified in JERG-2-211 Charging/Discharging Design Standard or the following (2) isolation diode shall be installed for every series circuit.

(2) an isolation diode for every series circuit in the array circuit
In view of failure separation and avoidance of sustaining discharge, consideration shall be given to the installation of an isolation diode for every series circuit.

(3) CIC output grade grouping

After assembling the CIC, obtain the output characteristics for each CIC, perform grouping for each output level and apply to the assembly of the CIC module so that no specific CIC will be the output-limiting cell.

(4) CIC layout position

In view of efficient power generation, in consideration of wiring loss and shunt I/F upper limit current, the CIC module layout shall be optimized in accordance with the output grade.

(5) Considering laydown around holding point

While adjusting with the holding point position during storage required from the structural design, CIC laydown shall be performed around the position in consideration of the following viewpoints.

(a) Avoid CIC laydown to places where shadow is cast steadily due to a bolt catcher and the like.

(b) Ensure a distance for avoiding occurrence of sustaining discharge if the array high potential portion is close to the hold-down section.

(6) Considering potential difference between adjacent cells

In designing the solar cell array layout, a potential difference between different array circuits or between folded adjacent cells shall be laid out to meet the requirements specified in JERG-2-211 Charging/Discharging Design Standard for avoiding sustaining discharge. When evaluating the potential difference between other array circuits, consideration shall be given to the potential difference due to shunt operation.

(7) Controlling magnetic moment

To control magnetic moment which develops on the solar cell panel, take account of a design that can cancel the magnetic moment developed in the solar cell panel in consideration of the following points regarding cell layout and harness routing.

(a) In the layout, the polarity of the solar cell array circuit in the solar cell panel shall be line-symmetric to the greatest extent possible.

(b) In the harness layout from each circuit of the solar cell array to the connector which interfaces with the paddle drive unit, occurrence of the magnetic field associated with the current loop shall be minimized.

(8) Setting the bypass diode for shadow countermeasures

If a shadow of the structure mounted in the spacecraft is cast on the solar cell panel surface when the spacecraft orbits in a general attitude or attitude is lost, the shaded cell will be in a reverse bias state. The worst case is when only one cell in one series circuit is totally shaded. A bypass diode for shadow countermeasures must be installed to prevent a breakdown without exceeding the reverse bias withstand voltage of the cell. Because the general silicon cell has a reverse withstand voltage of about 30 volts, it is desirable to install a bypass diode for shadow countermeasures for every 30 series circuits. As the reverse withstand voltage has temperature dependence, the number of insertion steps shall be set after grasping the temperature characteristics and unevenness of the solar cells to be used in consideration of proper derating (50% or less recommended). Because the III-V compound-based solar cell has a low reverse withstand voltage in general, the bypass diode for shadow countermeasures shall be installed on a cell-by-cell basis. In selecting bypass diode parts for shadow countermeasures, a model which is capable of passing electric current for one series circuit in consideration of derating (50% or less recommended) shall be chosen. Some cells are integrated with a bypass function (IBF; Integrated Bypass Function). When IBF cells are used, it is possible to simplify the panel integration because the independent shadow diode can be omitted.

(9) Charge array layout

For a solar array wing having a charge array, consideration shall be given to the following points with respect to the layout of the charge array.

(a) Consideration to risk of power reduction due to ESD

Based on charging analysis, consideration shall be given to the array layout to the location which minimizes the risk of reduction in power generated by the charge array due to ESD.

(b) Minimizing influence of shadow cast by satellite body

Consideration shall be given to the array layout to the position which minimizes the reduction in power generation due to a shadow of the satellite body cast on the charge array.

5.1.1.4 Design of interface between PDL and paddle drive mechanism

As an electrical interface design of a PDL-to-paddle drive mechanism, consideration shall be given to the following design points with respect to:

1) Paddle drive mechanism slip ring design;
2) Array circuit occurrence voltage and slip ring withstand voltage design;
3) Processing of spare ring; and
4) Design notes for ground line.

(1) Paddle drive mechanism slip ring design

Based on the design of the power generation analysis and telemetry (such as temperature sensor line and angle detection signal line)/command and design of the heater line and ground line, electric design of the slip ring (ring current capacity and the number of rings) required in the paddle drive mechanism shall be performed. In selecting the slip ring current capacity, consideration shall be given to derating (50% or less recommended).

In determining the current capacity of the hot-side ring for electric power, the ring capacity selection condition differs depending on whether the shunt is mounted to the solar cell array side (so-called yoke mounted) from the paddle drive mechanism or to the spacecraft structure side. When the shunt is mounted to the structure side, it shall be necessary to select the hot-side slip ring for electric power having the current capacity corresponding to the one-array circuit connected to the shunt element. When the yoke is mounted, the large-capacity slip ring can be used in the range where both HOT and RTN rings can be selected.

In selecting the slip ring for electric power capacity selection, consideration shall be given to derating with respect to the worst case including abnormal attitude and the like (conditions under which generation current is maximized (BOL, solar light vertical incidence, high temperature condition and array circuit shunt state)).

Capacitive surge current due to charging/discharging of nongrounded insulators such as solar cell array cover glass may pass through the slip ring, harness and connector (including connector pin). Although a surge current due to discharge is about 100 A at the maximum and it continues for about several tens of μ seconds (depending on insulator capacity and charged potential), it is desirable that this capacitive surge be considered in selecting the capacity of the slip ring and the like.

(2) Array circuit occurrence voltage and slip ring withstand voltage

For the withstand voltage between adjacent slip rings and the withstand voltage between brushes, consideration in design shall be given to the creepage distance and spatial distance having a proper discharge tolerance with respect to the maximum voltage (PDM output terminal voltage in electric power source (EPS) system interface) generated by the solar cell array and other signal voltages. In designing the withstand voltage distance, consideration shall be given to influence of abrasion powder due to sliding between the slip ring and the brush. Open mode failures related to the harness and connector shall be always reviewed with respect to the worst-case voltage.

(3) Processing of spare ring

In view of charging and discharging (see JERG-2-211Charging/Discharging Design Standard for details), the spare ring shall be grounded so as not to leave any suspended conductive materials. When performing grounding, it is desirable to insert resistance to the empty ring in which a signal capable of passing a large current is laid out to the adjacent ring.

(4) Design precautions to ground line

When employing a design on the premise that the substrate ground fault of the solar cell array is permitted without mounting the breeder resistance (ground line resistance), the number of slip rings and ground line wire current capacity which are properly within the allowable range with respect to the estimated maximum ground fault current shall be designed.

5.1.1.5 Shunt interface design

In accordance with the power generation analysis, it shall be clarified that the design satisfies the maximum allowable current with respect to the shunt circuit. As the electrical interface design between PDL and shunt, the following design consideration shall be given to the current and voltage.

(1) Current interface

For the current capacity design of the one-array circuit, the allowable current in light of derating shall not be exceeded in the worst case including attitude abnormalities (conditions under which the generation current is maximized (BOL, solar light vertical incidence, high temperature condition and array circuit shunt state)).

(2) Voltage interface

For the design where the shunt is mounted to the solar cell array side (so-called yoke mounted) from the paddle drive mechanism, when brush chattering is likely to occur due to an environment where the equipment undergoes a physical shock, consistency with the shunt element withstand voltage shall be checked with respect to the worst case voltage (BOL, low-temperature condition and array circuit open state) as array circuit occurrence voltage. In case of inconsistency, it shall be clarified that measures be taken with respect to the power supply system shunt design.

5.1.2 Operation mode and power consumption design

The power consumption of the paddle system subsystem in each operation mode shall be specified. Mainly, a paddle drive mechanism driving power and damper heater power shall be included.

5.1.3 Electromagnetic compatibility design

Design requirements for grounding system design, bonding and electromagnetic interference limit shall be entered.

5.1.4 Signal design

5.1.4.1 Suspended capacity of signal wiring

In telemetry design of the solar array wing, PA (passive analog) telemetry may be used for deployment status signal and panel temperature. In the solar array wing, the time constant related to wiring stray capacitance will be large due to the influence of these lengthened signal wiring routes and temperature (low-temperature environment). This is the large solar array wing ground test and it is difficult to check consistency with the spacecraft system in simulation of the low-temperature environment in an End-to-End test. Thus, consideration shall be given in design and analysis to consistency between the data sample timing of the data handling system and the time constant of the above-mentioned PA telemetry signal (reduction in stray capacitance, checking of sample timing interface consistency, design corresponding by active analog or deployment status design to decrease the time constant after deployment).

5.1.4.2 Consideration of influence of noise from the power line associated with shunt operation

The signal lines (such as temperature telemetry) of the solar array wing are susceptible to noise from the power line associated with shunt operation. In large solar array wing ground test, it is difficult in this part to check the consistency with the spacecraft system in an End-to-End test. Thus, consideration in design shall be applied to the signal lines such as adoption of shield lines or separation from the power line in the line routing in the range free from troubles in deployment reliability as necessary. If it is difficult to deal with noise, the signal lines susceptible to noise shall be designed with a function only for a telemetry monitor and not be used for on-board control.

5.2 Structural design and analysis

This section describes the design points to note in solar array wing electric design and analysis.

The design items to be checked by carrying out structural design and analysis shall be as follows. The structural design flow is shown in Figure 5-2.

- In the solar array wing storage state, the structure shall have the necessary strength to withstand the load conditions provided from the spacecraft system and satisfy the rigidity requirements for the solar array wing system. In structural design and analysis, consideration shall be given to acceleration at launch, acceleration during a sine wave vibration test, cumulative fatigue, acoustic environment, shock environment and heat load between the structure and solar array wing. When another subsystem device is mounted to the solar array wing system, a proper interface shall be set to the mechanical environment of the section where the device is mounted. Consideration shall be given to load concentration due to uneven rigidity in the system structure side in the solar array wing mounting section.
- During solar array wing deployment operation, the structure shall have a proper strength against shocks from the deployment latch.
- In the solar array wing deployed state, the structure shall have strength to tolerate external disturbance caused by spacecraft orbit control and attitude control and rigidity to satisfy interface requirements with the attitude control system. Interface requirements with the attitude control system shall be satisfied with respect to the orbital disturbance in which the solar array wings can be a source of vibration such as thermal snap.

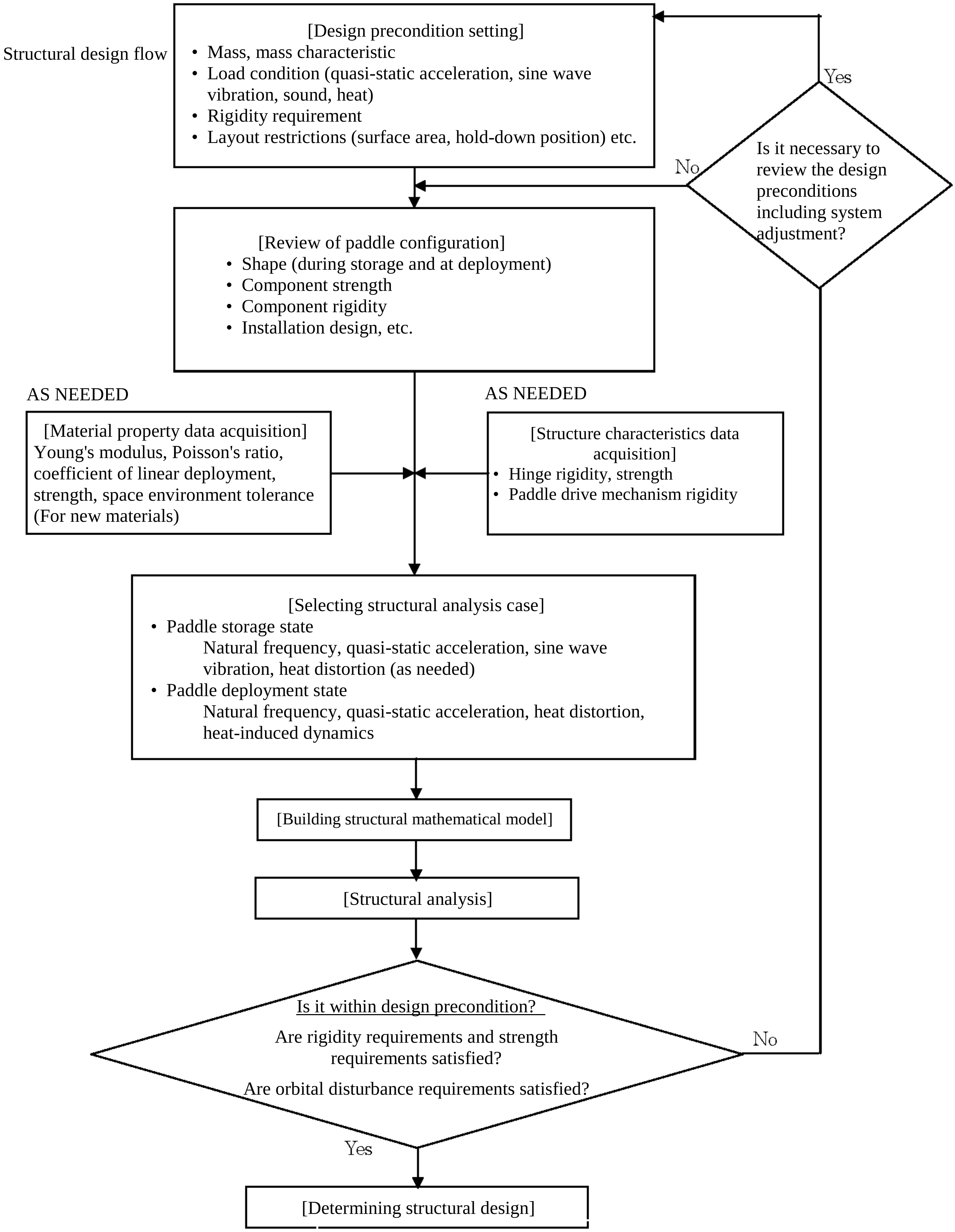


Figure 5-2 Structural design flow

5.2.1 Panel substrate design

The substrate is a panel mounting the solar cell array which is a major part of the solar array wing.

The design items related to a substrate shall be as follows:

- A substrate is a major structural element of the solar array wing and usually has a CFRP face sheet honeycomb panel structure to realize lightweight and high rigidity. Because the CFRP face sheet on the substrate surface is a conductive material, a polyimide film must be attached to the surface mounting the solar cell array to prevent a short circuit. Before starting laydown, the polyimide film shall be checked for insulation resistance.
- The honeycomb core shall be checked to confirm whether perforations are properly open before using. When using t10 (10 mm thick) or less, while noting the influence of absence of perforations.
- To prevent the solar cell surface from reaching a high temperature, emissivity shall be improved by affixing some heat control material or increasing the surface roughness.
- The substrate hold-down shall be surface-treated to prevent the joint surface from adhering in orbit or in use with the surface-treated pad pinched. Care shall be taken to prevent the hold-downs from coming into contact with each other during deployment due to the linear deployment difference between the spacecraft structure and the paddle, thereby hindering deployment.
- Because the substrate is often exposed to a severe temperature environment, when selecting the material to be adhered to CFRP, the difference of linear deployment coefficients shall be as small as possible. In using adhesive, proper thickness shall be controlled. Care shall be given so as not to develop cracking under a thermal environment. Materials having a glass transition point within the operation temperature range shall be evaluated using the property values at the maximum and minimum temperatures.
- All metal materials to be mounted to the substrate shall be electrically coupled to the conduction reference point (substrate ground point) which is electrically coupled to the satellite. By so doing, care shall be taken so that no potential difference is generated. When the honeycomb core is coupled, care shall be taken so that electrical coupling between honeycomb cores is ensured.
- Consideration shall be given so that no damage occurs due to the difference of heat distortion between the substrate and the spacecraft structure when the substrate is mounted to the spacecraft.
- Contact prevention measures shall be taken with portions where displacement is large under a vibration environment such as substrate angle and center sections.

Consideration shall be given so that no joints are adhered in orbit by performing solid lubrication or ceramics film treatment on contact-prevention measures portions so that deployment is not hindered.

5.2.2 Boom and yoke design

The boom and yoke shall have a function to deploy a solar cell panel from the spacecraft body to a proper distance in orbit and maintain the panel. The design of boom and yoke shall be determined in consideration of the range of shadow generated by spacecraft-loaded objects, mass characteristic requirements, spacecraft body structural element layout (hard point), solar array wing synchronization mechanism layout and harness layout.

The design items related to boom and yoke shall be as follows:

- The length of boom and yoke elements shall be determined in accordance with the requirements for a spacecraft system (analysis of shadow formed by objects loaded in the spacecraft, mass characteristic analysis).
- Check the distance between the boom/yoke elements and solar cell panel. Element design shall be done so that no boom/yoke elements will come into contact with the panel even when the elements are vibrated under a vibration environment.
- A harness wiring shall be located in a place out of direct solar light where possible in consideration that harness temperature is likely to increase in portions close to the spacecraft structure due to reflected heat of the structure. If it is inevitable to install wiring under direct solar light, shielding shall be provided by MLI or SLI.
- Consideration in design shall be given to adhesion in the hold-down section as in the case with the substrate.
- Consideration in design shall be given to bonding to CFRP elements as in the case with the substrate.
- The boom and yoke shall be electrically coupled to all the metal materials to be mounted in the boom and yoke.

5.2.3 Design of release section

The release section shall have a function to fix and hold objects such a solar cell panel and boom and yoke with bolts at launch of the spacecraft and release the parts in orbit with the releasing parts including explosive devices incorporated.

The design items related to release section shall be as follows:

- During releasing operation, it is desirable that the release section will come into contact with no other structural elements regardless of the locus of the preload bolt. In case of contact, it shall be fully checked that snagging will not hinder deployment.
- To prevent the preload bolt from moving backward after releasing the holding state, preventive measures shall be taken such as providing an unreturning spring material.
- Consideration in design shall be given to adhesion in the hold-down section as in

the case with the substrate.

・In consideration of conditions of installation to the system structure, the template for locating the holding console shall be prepared as needed.

- When installing an explosive device, the explosive device shall be located easily. There shall be easy access to the explosive device.
- The coupling interface with the system side EED harness shall be clarified. Check the presence/absence of MLI on the EED harness side. Check for interference with the deployment locus of the paddle structural elements.
- When installing the MLI to the holding console, an interface with the system structure side MLI shall be clarified with regards to aspects such as the presence/absence of the tape.
- The release section shall be electrically bonded to the explosive device to be installed using wires and the like.
- For shock release, a proper interface with the system structure shall be specified. When conducting a verification test at the subsystem level, the configuration such as jig composition shall conform to the actual machine.

5.2.4 Structural analysis

5.2.4.1 Analysis of structure during storage

- Analysis of characteristic values during storage shall be conducted to check that the rigidity requirements to the solar array wing are satisfied.
  The boundary conditions with the system structure shall be clarified (completely fixed in general).
- Acceleration load strength analysis at launch and strength analysis during the sine wave vibration test shall be conducted to check that the strength requirements for the solar array wing are satisfied.
  Examples of strength assessment shall be as follows: In the case of evaluating the strength of an actual machine, the test result of a model simulating the actual machine shall be used where possible.
  - Material fracture (Tsai-Wu)
  - Face sheet interlayer shear
  - Skin buckling (intra cell buckling (*1) or wrinkling (*2))
  - Insert joint evaluation (such as opening and slip)

    (*1) Fracture mode in which the honeycomb panel face sheet buckles in the range within the honeycomb core cell.

    (*2) Fracture mode in which the honeycomb panel face sheet buckles across the wall surface of the honeycomb core cell core due to a short wavelength and the panel is damaged as the load is not transferred.

5.2.4.2 Structural design and analysis after paddle deployment

(1) Flexible structure mode after deployment

The subsystem side of the solar array wing system shall analyze the flexible structure of the solar array wing system after completion of normal deployment in orbit by using finite element methods for structural analysis (such as finite element method, differential equation numerical solution and differential equation analysis solution). Modelization and analysis shall be carried out on the entire solar array wing system. When the system has a paddle drive mechanism, a paddle drive mechanism shall be included. In analysis, fix-free boundary conditions in which the spacecraft body installation section is assumed to be a completely rigid interface surface shall be applied. The natural frequencies from the lowest order mode to the waveband specified in coordination with the spacecraft system side in this analysis and the corresponding zero-order and primary coupling coefficient vectors and the mode-damping ratio shall be indicated. These flexible structure parameters and corresponding allowable errors shall be specified and provided to the spacecraft system. It shall be checked that these flexible structure parameters and corresponding allowable errors are within the allowable range of the flexible structure parameters specified by the spacecraft system.

The spacecraft system side shall specify the flexible structure parameters under fix-free boundary conditions provided from the subsystem side to the spacecraft model and obtain the flexible structure parameters under fix-free boundary conditions, thereby performing analysis of spacecraft dynamics, attitude control and attitude stability using the model. Consideration shall be given to the flexibility of the installation section in the spacecraft body side. The flexible structure parameters from the spacecraft system side and corresponding allowable errors shall be specified. It shall be required to specify not only the minimum natural frequency but also the allowable waveband of the low-order natural frequency in accordance with the need for attitude control and stability.

The above-mentioned analysis, regulations and evaluation shall be carried out on major deployment abnormality modes as needed.

Verification of the flexible structure mode after deployment shall be carried out at least by building the entire mode through buildup of rigidity measurement results of individual parts or similar parts. In actual machine verification at the subsystem level, the rigidity in a deployed state shall be statically measured to check the natural frequency of the low-order flexible structure mode in a deployed state.

(2) Orbital heat distortion analysis

The subsystem side shall perform analysis of orbital temperature variations in each portion after reaching a thermally stationary cycle state with respect to the solar array wing system which completed deployment in orbit by using the finite element methods for thermal analysis (such as finite element method, differential equation numerical solution and differential equation analysis solution). Consideration shall be given to thermal influence of the spacecraft body and rotation of the paddle (spacecraft body) in analysis. Any temperature change in both sides of the paddle shall be modeled. The obtained temperature distribution profile shall be entered to the finite element methods for structural analysis (such as finite element method, differential equation numerical solution and differential equation analysis solution) to obtain the orbital deformation profile of the solar array wing system in a non-controlled attitude state (quasi-static heat distortion profile and dynamic response profile) and the installation point torque. The transient responses consisting of the quasi-static heat distortion variations and dynamic responses between equilibrium state under irradiation and equilibrium state in the shade which occur immediately after the start and end of an eclipse shall be called "heat-induced dynamics."

(3) Strength analysis

Strength analysis shall be carried out on each section of the solar array wing system with respect to the following load inputs using the finite element methods for structural analysis (such as finite element method, differential equation numerical solution and differential equation analysis solution) to check that M.S. is positive.

a) Acceleration, angular acceleration input to the solar array wing system installation section at thruster injection and application of large torque: Consideration shall be given to the barycenter offset resulting from deformation due to heat distortion in the solar array wing system as well as inertia.
b) Deformation of the solar array wing system due to thermal factors during orbit (quasi-static heat distortion profile and dynamic response profile) based on analysis in (2).
c) Coupled vibration of paddle drive mechanism and solar array wing and spacecraft body associated with paddle driving.

For b) and c), the fatigue tolerance to the cycle applied during mission period shall be checked. It shall be necessary to check the tolerance of not only the mechanism but also the structural elements such as substrate, electric parts such as solar cells and their composite articles.

(4) Heat-induced dynamics analysis

The subsystem side shall perform heat-induced dynamics analysis under fix-free boundary conditions based on the result of transient response between equilibrium state under irradiation and equilibrium state in the shade in heat distortion analysis in the above-mentioned (2). Analysis shall be carried out in accordance with the finite element method, motion equation numerical simulation and analysis solution. It shall be necessary to provide the spacecraft system side with a paddle deformation profile, installation point reaction torque and barycenter variation obtained from analysis.

The spacecraft system side shall carry out heat-induced dynamics analysis including contribution of the spacecraft body, other flexible structures and attitude control system based on the analysis of the subsystem side, thereby evaluating the influence of disturbance affecting attitude control and attitude stabilization.

5.2.5 Mass characteristic analysis

- Mass evaluation shall be carried out based on the measured values of the solar array wing to check that solar array wing mass requirements are satisfied.
- Mass characteristic analysis in a storage state shall be carried out to check that the solar array wing mass characteristic requirements are satisfied. Tuning of the analysis model shall be carried out based on the measured values of mass characteristic test to calculate the mass characteristic values with respect to all the coordinate axes.
- Mass characteristic analysis in a deployed state in orbit to check that the solar array wing mass characteristic requirements are satisfied. Tuning of the analysis model shall be carried out based on the measured values of a mass characteristic test to calculate the mass characteristic values with respect to all the coordinate axes.

5.3 Mechanism design and analysis

Solar array wing mechanism parts (deployment mechanism, synchronization mechanism) shall have a function to develop the solar cell panel in orbit and maintain the deployment state over the entire mission period. The parts shall have the strength to withstand external disturbance due to spacecraft orbit control and attitude control and rigidity to satisfy interface requirements with the attitude control system.

This section describes the design points to note in solar array wing electric design and analysis.

Figure 5-3 shows the mechanism design flow.

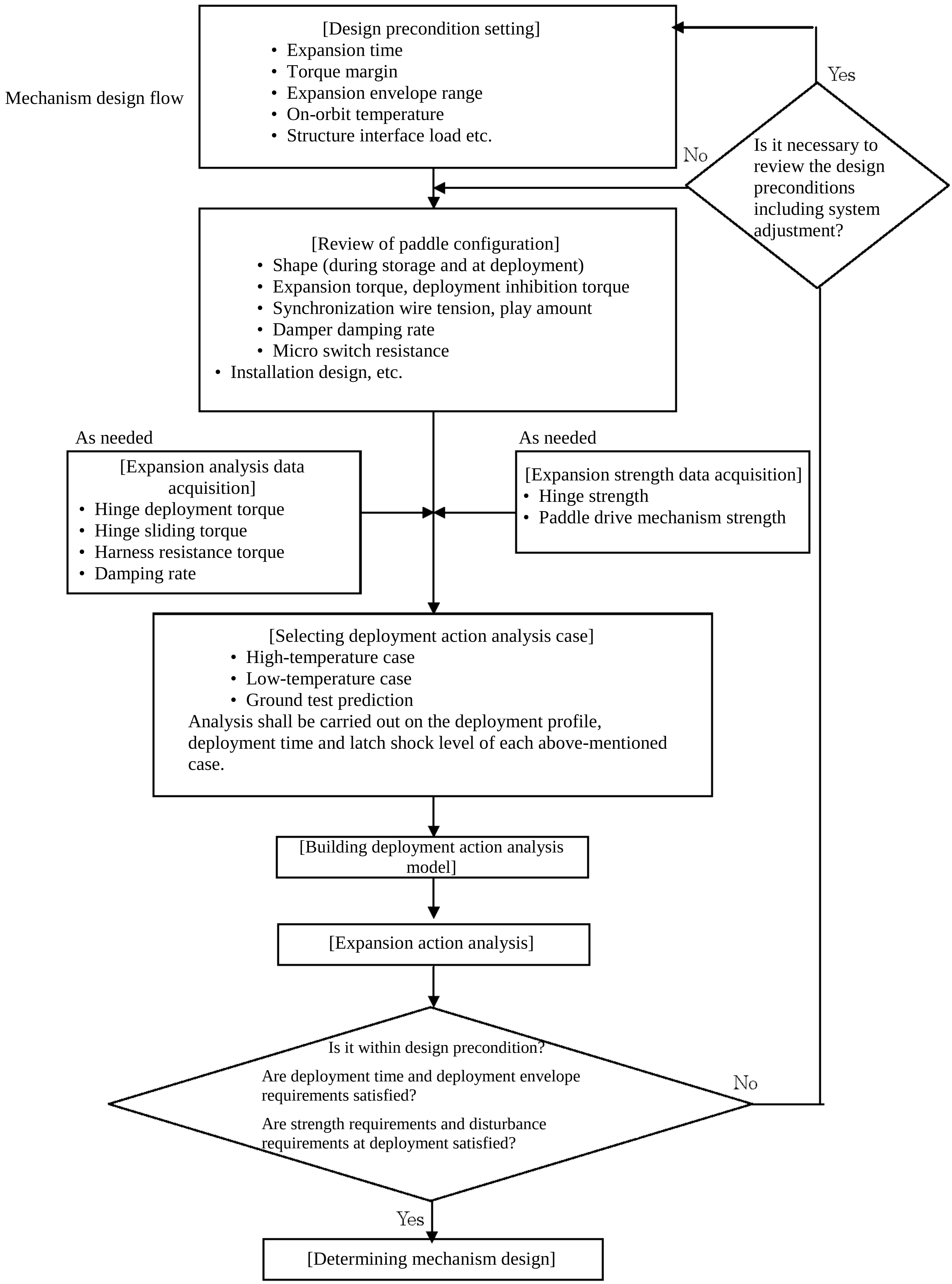


Figure 5-3 Mechanism design flow

5.3.1 Expansion mechanism design

The deployment mechanism shall deploy the boom, yoke and solar cell panel released in orbit at a constant speed and maintain the deployment state.

The design items related to deployment mechanism shall be as follows:

- The sliding section at deployment of each hinge shall be surface-treated such as solid lubrication to prevent adhesion.
- The bearing section shall be designed so that the clearance of the outer ring/inner ring is ensured under a low-temperature environment in consideration of the temperature range to be loaded.
- The hinge of one side inside the hinge line shall be provided with free play in the thrust direction and designed to absorb displacement difference due to linear deployment difference between the panels sandwiching the hinge line.
- Calculation result of deployment torque margin derived from the following formula shall be 1 or more. However, if the requirements cannot be satisfied, judgment shall be made based on the project.

  deployment torque margin = deployment torque/resistance torque -1

  (resistance torque = harness resistance torque + hinge sliding torque)
- Harness resistance torque shall be designed based on the measured value using a harness equivalent to that of the actual machine where possible.
- Hinge sliding torque shall be designed based on the measured value using the hinge equivalent to that of the actual machine where possible.
- The telemetry to confirm the completion of deployment of the hinge shall be detected and output by the micro switch provided in the hinge. When the micro switch is installed, connect a different resistance for every hinge and design the mechanism so that you can check which hinge completed deployment by monitoring the resistance value.
- The mechanism shall be designed so that the result MS of deployment latch shock evaluation will be positive. The damper shall be installed to relieve shock at the time of latch deployment where appropriate.

5.3.2 Synchronization mechanism design

When multiple components such as a boom, yoke and solar cell panels in the solar array wing are coupled by hinges and it is likely that a collision will occur between components and with the spacecraft body at deployment action, it shall be necessary to install a synchronization mechanism to synchronize the solar array wing with the deployment angle of the hinges.

- The deployment synchronization mechanism shall be designed to provide the synchronization wires with a proper tension in consideration of on-orbit temperature variations at paddle deployment action.
- The synchronization wire section including caulking parts shall be checked in advance for soundness by applying proof load.

- The mechanism shall have a structure not to develop slack due to non-synchronization at deployment action to the extent possible. The deployment action envelope of the synchronization wires shall be free from protrusion and height difference so that no snagging occurs in case of contact.
- The synchronization wires installed in the boom and yoke are likely to cause changes in amplitude during a vibration test due to the free play of the wires. Steady rests shall be installed at intervals so that no contact with the panels occurs.

5.3.3 Deployment design and analysis

The subsystem side shall analyze the deployment behavior of the solar array wings in orbit by multi-body dynamics numerical simulations (such as multi-body dynamics numerical simulation software, differential equation numerical solution, differential equation analysis solution) to obtain the deployment configuration profile, deployment angle profile, deployment signal profile, deployment time, latch shock torque and installation point reaction torque profile. Analysis shall be carried out at the moment of releasing in the case of passive deployment by the spring and the like and immediately before motor drive by the motor and the like as for active deployment. For the boundary conditions of paddle fixed end, be sure to carry out analysis under not only fix-free conditions but also free-free conditions including the satellite body capable of rotating the space freely. (Free-free conditions shall be the minimum required conditions.) The satellite body for analysis under free-free conditions may have a rigid body when other flexible structures are not deployed. The subsystem side at least shall carry out analysis under fix-free conditions. The subsystem side or the satellite system side or both sides shall carry out analysis under free-free conditions. However, the satellite system side shall bear the ultimate responsibility. The subsystem side shall interface the conditions required in analysis under free-free conditions to the satellite system side. For fix-free or free-free conditions, analysis shall be carried out at least on two cases including assumed highest and lowest temperature cases. Among analysis parameters shall be hinge torque, hinge deployment torque, harness resistance torque, damper resistance torque, deployment synchronization mechanism tension, temperature, hinge offset and the like. The values verified based on the results of element prototype tests shall be used including characteristic changes due to temperature. It shall be required to carry out analysis of worst-case combinations of parameters which envelop the variation range of analysis parameters to check the deployment reliability. Based on the analysis results, check whether the deployment time, latch shock torque, M.S. at latch shock and deployment configuration envelope range satisfy the requirements. In addition, the satellite system side shall check whether the attitude rate at deployment and attitude angle at deployment satisfy the requirements.

The subsystem side shall verify the deployment of the solar array wings in a ground deployment test and model the deployment behavior of the solar array wings in the ground deployment test in consideration of the contribution of aerodynamic force and frictional force as well as varied boundary conditions (under fix-free conditions in a ground deployment test in general), thereby performing analysis by multi-body dynamics numerical simulation as in the case with on-orbit behavior. Tune the analysis parameters of ground deployment to conform to the ground test results and reflect the tuning results in the on-orbit deployment analysis.

As needed, set the conditions in simulation of fluctuations in harness resistance torque out of the assumed range, failures of damper and deployment synchronization mechanism and conduct on-orbit deployment analysis to check for robustness during deployment.

5.4 Thermal design and analysis

An example of thermal design flow in solar array wings is Figure 5-4.

5.4.1 Thermal design

In accordance with the equipment allowable temperature of parts and components of the solar array wing system, it shall be necessary to carry out thermal design of the system and clarify temperature control requirements with respect to the spacecraft system. In accordance with JERG-2-310 Thermal Control System Design Standard and the thermal design standard specified in the project, design policies regarding design uncertainty (uncertainty of errors and tolerances) and design margin (allowance for unexpected occurrence) shall be specified.

[1] Precondition

- Internal calorific value (consider the exothermic density as for equipment with high calorific value)
- External heat environment (during test, transportation, storage, launch, on-orbit operation mode and abnormal attitude)
- Thermal interface conditions provided from the system side
- Equipment allowable temperature (characteristic values of parts and components) and rate of allowable temperature change
- Requirements of operating life and reliability
- Power generation requirements (the conversion efficiency of the solar cell is decreased as the

temperature increases)

[2] Thermal design

- Basic structure of the paddle shall be determined in consideration of the following points:
  - Shape
  - Surface area
  - Thermo-optical characteristics of cover glass and panel back side
  - Installation method
  - Surface treatment
  - Temperature regulation point
  - Thermophysical property

- Selecting thermal control materials (parts and materials)
  Selection shall be performed in consideration of thermophysical properties (such as thermo-optical characteristic, heat conduction, heat capacity), reliability, environmental resistance (such as radiation, atomic oxygen, charging/discharging), process (quality, reproducibility), outgas, cleanness, performance deterioration due to contamination, flight track record, availability, useful operating life and workability.
- Conductivity of thermal control materials and grounding
- Optimization of heat radiation area and heating capacity (power)
- Calculation of high and low worst-case predicted temperatures
- Temperature measurement points and temperature sensors
- Adhesive strength, outgas, deterioration (aging, temperature)
- Vibration at launch and strength and separation under a reduced pressure
- Influence of adhesion of objects separated from the spacecraft structure and their shadow

[3] Test temperature

- Qualification test: Temperature range higher than the maximum predicted temperature range by 5°C or more and not exceeding each equipment allowable temperature (parts and components)
- Acceptance test: Temperature range outside the maximum predicted temperature range and not exceeding the acceptance test temperature range of each equipment allowable temperature (parts and component)
- Proto-flight test: Same as qualification test

5.4.2 Thermal analysis

- In accordance with the thermal interface conditions, a detailed thermal math model of solar array wings shall be created. While improving accuracy by verification and co-relation, design review of solar array wings, on-orbit temperature prediction, thermal equilibrium test evaluation, design validation and on-orbit evaluation shall be carried out.

- As thermal analysis cases, the worst case (including solar light strength, thermal interface conditions and thermo-optical characteristics) shall be selected for both the high-temperature and low-temperature cases of solar array wings. As for the high-temperature case, consideration shall be given to full shunt. The high-temperature worst cases of harness bundles and connectors shall be considered in consideration of self-heating.

- As needed, an interface thermal math model for system thermal analysis shall be created. The validity shall be checked through testing and analysis.

[Reference: JERG-2-310 Thermal Control System Design Standard]

- Predicted design temperature range: Nominal design temperature in consideration of seasonal variations and aging deterioration

- Maximum predicted temperature range: Temperature in consideration of uncertainty such as difference between actual machine and thermal math model, analysis tools, design parameters, manufacturing tolerance and measurement errors.

<design uncertainty> = <maximum predicted temperature range> - <predicted design temperature range> Example: 10°C or more

<design margin> = <equipment allowable temperature> - <maximum predicted temperature range> Example: 5°C or more

Thermal design flow

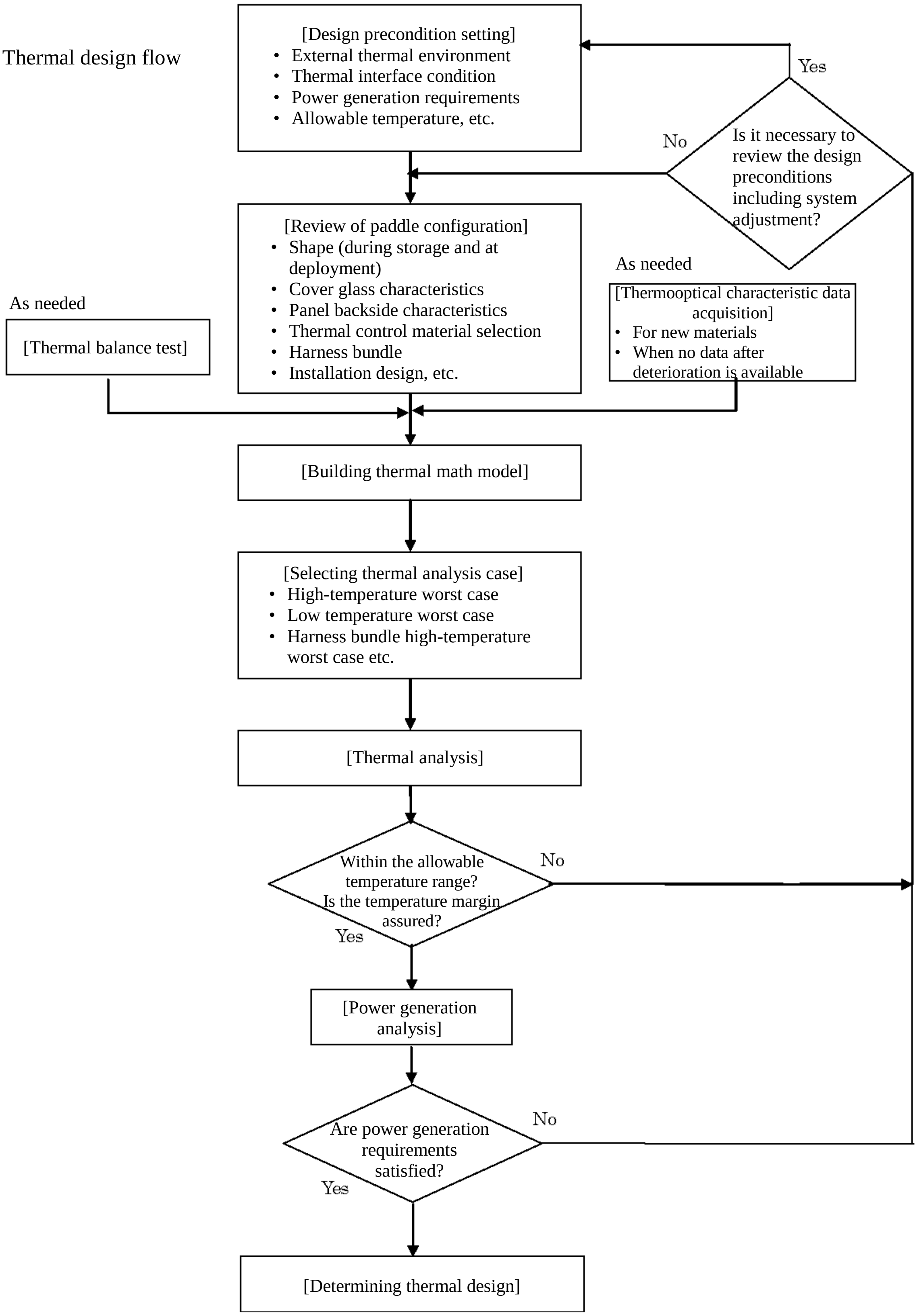


Figure 5-4 Thermal design flow

5.5 Paddle drive mechanism design

The functions of paddle drive mechanism shall be as follows:

(1) Rotates the solar array wings around the 1 axis to direct the solar array wings to the sun.
(2) Transfers generated power of the solar array wings to the spacecraft body.
(3) Transfers electric signals between the solar array wing and spacecraft body.
(4) Outputs drive shaft rotation position signals by the angle detector.
(5) Outputs drive shaft reference position signals by the reference point sensor (in accordance with system requirements).

5.5.1 Electric design

(1) Actuator

The actuator shall be clarified after performing I/F adjustment with the attitude control system/drive circuit with respect to the specifications of the following items.

(a) Drive characteristics

- Excitation circuit method
- Excitation sequence
- Winding impedance/inductance
- Drive voltage/current
- Drive phase counter electromotive voltage
- Electric power consumption, calorific value
- Pulse rate
- Step angle

(2) Slip ring assembly

The slip ring assembly shall be clarified after performing I/F adjustment with the power supply system and attitude control system with respect to the specifications of the following items.

(a) Ring configuration: Rated current value and number of power ring (RTN), power ring (HOT) and signal ring

(b) Maximum working current value: 1/2 or less of rated current value. It is desired to consider the capacitive surge.

(c) Maximum working voltage: Maximum applied voltage to each slip ring

(d) Connector: Type, number, pin assignment

(e) Empty ring: Empty rings (reserve rings) which are not to be used and not connected to anything shall be grounded in the system.

(f) Static resistance value Static resistance shall not increase at the stop. (However, temporary increase under a ground environment (when the value returns to the normal value due to rotation) shall be allowable.)

(g) Dynamic resistance The variable width of resistance value during rotation shall be regulated.

(h) Short circuit prevention The barrier section shall have a proper height and width to prevent a short circuit between adjacent rings caused by the brush exceeding the barrier due to vibration, shock and the like. Interior of the slip ring case shall be short-circuit proof.

(i) Countermeasures for discharge The distance between adjacent rings and distance between brushes shall be designed properly to prevent discharge between slip rings.

(3) Angle detector

The angle detector shall be clarified after performing I/F adjustment with the attitude control system/drive circuit with respect to the specifications of the following items.

- Machine angle: (Definition of relationship between mechanical origin and angle detector origin in particular)
- Angle/output characteristic: (Including angle detector unit accuracy)
- Input/output impedance
- Input voltage
- Number of working slip rings
- Connector pin assignment

(4) Selecting maximum working current value of harnesses, connectors and connector pins

- As in the selection of slip ring maximum working current value, the maximum current value for the connector and connector pin used in PDM shall be 1/2 or less of the rated current value.

However, the total conduction current of the connector shall be designed within the allowable temperature range in accordance with MIL-W-5088-based derating guidelines indicated in the connector application data sheet and, ultimately, thermal analysis.

- The current value of harnesses shall be in accordance with JERG-2-212 Wire Derating Design Standard while the current value of externally exposed harnesses (outside of spacecraft structure, shaft section harnesses and the like) shall be specified properly in accordance with the thermal analysis requirements in 5.4.2.

5.5.2 Machine design

(1) Review of structure characteristic

After clarifying the I/F conditions with the solar array wings and spacecraft structure, characteristic value analysis and strength analysis shall be carried out on the paddle drive mechanism to check whether the rigidity requirements and strength requirements are satisfied. Major evaluation items shall be as follows:

- Allowable load (translational force, bend, torsion)
- Allowable temperature (cabinet/output shaft temperature difference)
- Equivalent rigidity (bend, torsion)

(2) Reviewing drive characteristics

For the paddle drive mechanism, the following items shall be reviewed and clarified with respect to drive characteristics

- Rotation direction
- Step angle
- Backlash
- Rotation angle accuracy (maximum accumulated errors)
- Output shaft generating torque and torque margin (definition shall be clarified)
- Operating life analysis/evaluation

(3) The holding torque of the paddle drive mechanism in the stand-by mode and hold mode shall satisfy the requirements for paddle drive mechanism.

(4) Torque margin

The torque margin M shall be defined as in the following formula and the requirement value shall be 1 or more.

$M=T0/(T_I+T_D)-1$

T_O: Output torque

T_I: Torque to accelerate the solar array wings

T_D: Paddle drive mechanism internal drag torque

Output torque T_O can be obtained by step motor output torque x deceleration ratio.

Internal drag torque can be obtained with:

- Harmonic drive ® starting torque
- Harmonic drive ® efficiency,
- Friction torque of drive module bearing, spur gear efficiency
- Friction torque of slip ring assembly, and
- Friction torque of drive shaft bearing.

Because these values and step motor output torque vary depending on the temperature, a verification test shall be conducted on newly adopted articles in advance.

5.5.3 Thermal design

For the paddle drive mechanism, the following items shall be reviewed and clarified with respect to thermal design.

- Equipment calorific value
- Paddle drive mechanism installation (contacting surface area)
- Installation surface exothermic density
- Surface treatment of installation surface and other than installation surface
- Component temperature analysis and heat resistance evaluation
  (For the power wire harnesses in particular, it is desirable to carry out review and analysis in consideration of the thermal input from an external heat environment as well as self-heating due to energization.)
- Thermal design attribution as equipment
- Paddle drive mechanism I/F temperature (installation surface, harness connector I/F point)

5.5.4 Reviewing environmental resistance

Check for conformity to the Space Environment Standard JERG-2-141. (It is desirable in particular to evaluate the externally exposed sections of the paddle drive mechanism (including harnesses) for tolerance to a complex environment such as UV, AO, thermal cycle and radiation.)

5.5.5 Clarifying test plan and test items

For the paddle drive mechanism, the test plan including test items shall be clarified.

5.6 Paddle drive control design and analysis

(1) Paddle solar tracking control

Paddle solar tracking control shall be generally carried out by the spacecraft system (attitude control system in many cases) with the use of the angle detection sensors (such as paddle solar tracking angle sensor and paddle drive mechanism rotation angle sensor) and the drive actuator mounted in the solar array wing system. The subsystem side of the solar array wing system shall output the angle detection sensor measurement information with a specified accuracy and rotate the drive actuator with an accuracy specified according to the input signals. The sensor measurement information and actuator drive model shall be provided to the spacecraft system side. Alignment information shall be included in these sensor models and actuator models.

The spacecraft system side (attitude control system in many cases) shall design a paddle solar tracking control system based on the sensor models and actuator models provided from the solar array wing system to perform analysis of tracking control function and performance (such as control sequence, control accuracy, and disturbance).

(2) Drive mode design

Generally, the subsystem side of the solar array wing system shall design and specify paddle drive mechanism modes including standby mode, transition mode, hold mode, through mode, and clock mode as drive modes of the solar array wing system. On the other hand, the spacecraft system side (in many cases, attitude control system performing paddle solar tracking control) shall design and specify paddle sun acquisition standby mode, paddle sun search mode, solar tracking mode, array trim mode, automatic array trim mode, constant clock rate mode, and manual array trim mode (with duplication) as drive modes for macro action. The subsystem side of the solar array wing system and the spacecraft system side shall clarify the correspondence and combination of drive modes and interface each other. Information on the driving rules, action and sequence related to the drive modes shall be reflected in the tracking control function/performance analysis and paddle drive disturbance analysis of the modes.

(3) Failure detection identification reconfiguration (FDIR)

The spacecraft system side (in many cases, attitude control system performing paddle solar tracking control) shall design a failure detection identification reconfiguration (FDIR) function of paddle solar tracking control. The judgment conditions of abnormality detection, action sequence after detection, and restoration procedure in operation shall be clarified and provided to the subsystem side of the solar array wing system. The subsystem side of the solar array wing system shall check these for inconsistency with the design of the subsystem side of the solar array wing system.

(4) Paddle drive disturbance

The subsystem side of the solar array wing system having the paddle drive mechanism shall create a coupled dynamics model consisting of the step motor, harmonic drive, gear, solar array wing flexible structure mode, and spacecraft body and analyze the drive disturbance, vibration profile, reaction torque, and barycenter variations when driven in accordance with the tracking control drive signals provided from the spacecraft system by numerical simulation and provide the analysis results as well as the dynamics model to the spacecraft system. The subsystem side shall check that the analysis results are within the range specified by the spacecraft system.

The spacecraft system side shall incorporate the paddle drive disturbance model provided from the subsystem side into the spacecraft dynamics model including other flexible structures and conduct attitude/orientation disturbance analysis by numerical simulation. Consideration shall be given to the flexibility of the spacecraft installation section. Check for conformity to the attitude/orientation accuracy requirements in accordance with the results of attitude/orientation disturbance analysis. Specify the allowable disturbance amount according to the waveband with respect to the subsystem side of the solar array wing system.

See the disturbance control standard (JERG-2-152) and disturbance control manual (JERG-2-152-HB001) for the standard and method of disturbance control.

5.7 Space environment design

5.7.1 Radiation design and analysis

Details of radiation environment, radiation dose and radiation deterioration are described in Space Environment Standard JERG-2-141 and Radiation-proof Design Standard JERG-2-143. An overview is as follows:

- The radiation environment and radiation dose which the spacecraft encounters shall be predicted on a mission-by-mission basis using the radiation environment model specified by Space Environment Standard JERG-2-141. Because the confidence level which must be applied to each mission period is specified with respect to the prediction of solar proton, it shall be necessary to observe Space Environment Standard JERG-2-141.
- As the prediction of radiation dose involves substantial uncertainty, a proper margin shall be provided when reflecting the result in the design.
- It is difficult to determine the margin quantitatively but important items shall be reviewed and evaluated properly for an assured and efficient radiation-proof design.
- It shall be verified whether radiation-induced deterioration can be allowed with respect to the mission period by testing if possible.
- Omission of testing may be reviewed when using materials that have a track record of use in the orbit and period to be operated.

In particular, the influence of radiation exposure in orbit of the solar array wings shall be reflected in the design in the following two main points.

(1) Cell electric performance deterioration due to the solar cells exposed to radiation
(2) Physical deterioration of employed materials due to radiation exposure

For the above-mentioned (1), the deterioration degree of solar cells shall be quantified in accordance with the predicted radiation environment conditions and reflected in the electric design (determine the initial generated power in anticipation of deterioration). Among solar cell deterioration prediction methods are relative damage coefficient (RDC) method and displacement damage dose (Dd) method. Because the details of specific deterioration prediction procedures based on RDC method and Dd method are described in Radiation-proof Design Standard Handbook JERG-2-143-HB001, an overview and flow chart of the two methods are described here.

[Relative damage coefficient (Relative Damage Coefficient: RDC) method]

[1] Radiation test shall be conducted on electrons and protons of the target solar cells. Or already acquired data sheets shall be prepared.

[2] The energy spectrum distributions of electrons and protons fluence during mission period shall be obtained in accordance with the radiation environment prediction code.

[3] The deterioration curve of solar cells with respect to a 1 MeV electron beam shall be prepared from experience and data sheets.

[4] RDC with respect to electrons and protons of target solar cells (without shielding materials, radiation vertical incidence) shall be calculated. In addition, a 10 MeV proton/1 MeV electron beam conversion coefficient shall be derived.

[5] RDC with respect to electrons and protons of target solar cells shall be converted under the conditions of radiation omnidirectional incidence with the shielding materials.

[6] With the use of the RDC after conversion and the particle energy distribution of electrons and protons, the equivalent fluence of a 1 MeV electron beam shall be calculated and derived for electrons while the equivalent fluence of 10 MeV protons for protons. With the use of 10 MeV proton/1 MeV electron beam conversion coefficients, the equivalent 10 MeV proton fluence shall be converted to the 1 MeV electron beam fluence and added to the equivalent 1 MeV electron beam fluence of the former electron beam. Thus, the deterioration of cells due to electrons and protons flying with various energy/fluence can be converted to one indicator of deterioration with respect to the electron beam of 1 MeV.

[7] In the deterioration curve with respect to 1 MeV electron beam, the cell output at the end of operating life can be estimated by calculating the deterioration amount of solar cell output (preservation ratio) from the obtained equivalent 1 MeV electron beam fluence.

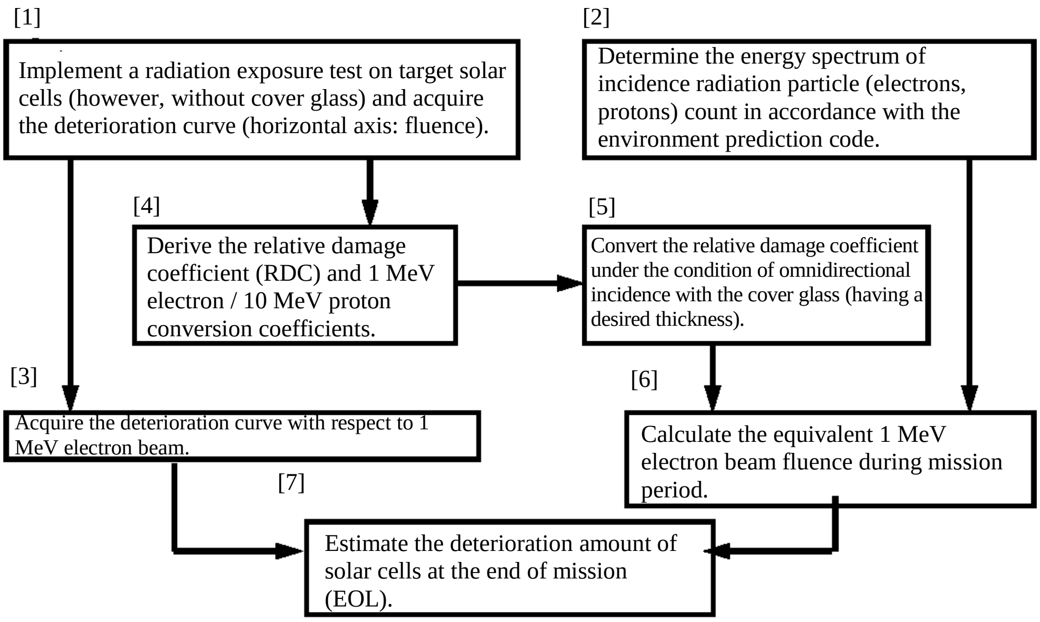


Figure 5.7-1 Procedure for predicting the solar cell deterioration amount by relative damage coefficient (RDC) method

[Displacement damage dose (Displacement Damage Dose: Dd) method]

[1] Derive the particle (proton/electron beam) energy dependence of non-ionizing energy (NIEL) value of solar cell materials (such as Si, GaAs).

[2] Indicate the horizontal axis (electron beam or proton beam fluence) of deterioration characteristic data of solar cells by using an indicator called displacement damage dose (Dd value = NIEL value × fluence).

[3] Calculate the radiation dose during mission period by using the radiation environment model.

[4] Calculate the radiation spectrum after transmission of shielding materials (such as cover glass and CFRP skin).

[5] Calculate all the D_d values during mission period from radiation dose proceeding into the solar cells and NIEL values.

[6] Estimate the cell output at the end of operating life in accordance with the cell deterioration data and all Dd values during mission period.

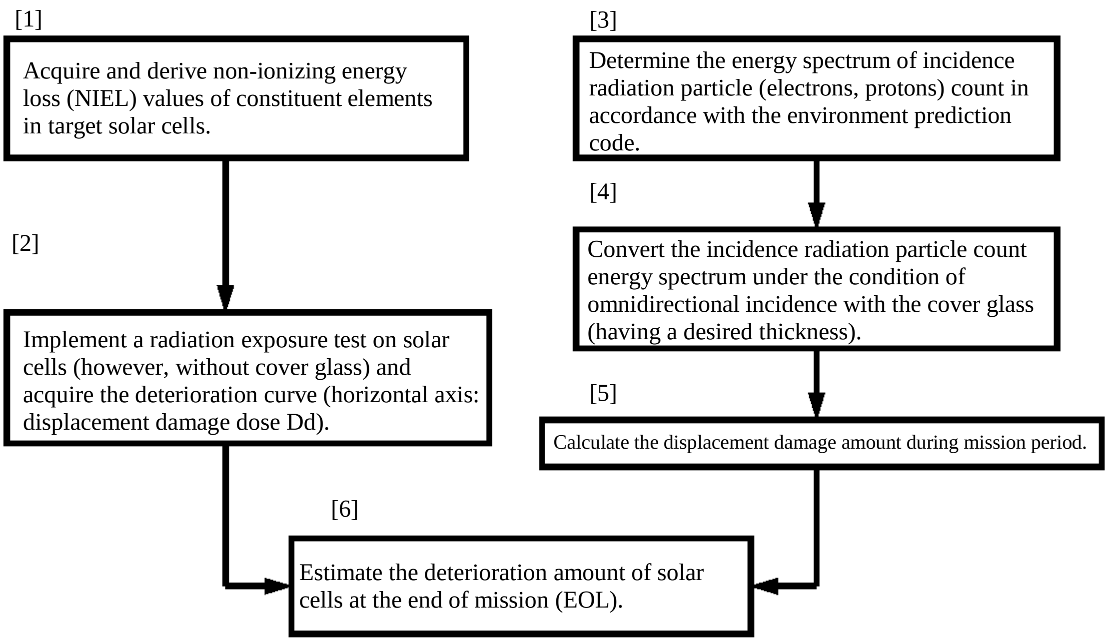


Figure 5.7-2 Procedure for predicting solar cell deterioration amount by displacement damage dose (Dd) method

For the above-mentioned (2), calculate the total ionizing dose and estimate the radiation deterioration in characteristics of applicable parts and materials based on the radiation resistance data of parts and materials. See Radiation-proof Design Standard JERG-2-143 and Radiation-proof Design Standard Handbook JERG-2-143-HB001 for details of the total ionizing dose.

- Consideration shall be given to the radiation deterioration evaluation of thermo-optical and mechanical characteristics and radiation resistance of the materials employed in solar array wings (such as harnesses, adhesive, cover glass, thermal control materials).

- It is desirable basically to select materials excellent in radiation resistance or materials having a flight track record.
- For example, wires covered with radiation-crosslinked ETFE excellent in radiation resistance including exposed sections are often used as cables for transmitting generated power, which is a basic function of solar array wings. The wires have a considerable track record in domestic spacecraft and overseas commercial spacecraft.
- When the radiation-crosslinked ETFE must be used under a high radiation environment and cracks may be developed in the covering, measures shall be taken to prevent a discharge and short circuit with peripheral elements even if the internal electric wires are exposed.
- When characteristic deterioration in parts and materials is within an allowable range in view of usage, the application is judged to be proper.
- If characteristic deterioration exceeds the allowable range, it shall be necessary to take measures such as shielding.
- Addition of shielding materials causes a considerable impact to spacecraft design in terms of weight and the like. Because there are still many uncertain elements including space radiation environment in radiation analysis, a certain degree of margin must be incorporated in design.
- If radiation resistance characteristics of parts and material are not known, deterioration data must be acquired by conducting a radiation test.
- Because the influence of radiation on materials may vary depending on the radiation type, energy and dose rate, attention shall be paid to this point when evaluating the data.

5.7.2 Ultraviolet (UV) countermeasure design and analysis

Among influences of ultraviolet on materials employed in solar array wings are ultraviolet deterioration in CIC generated power (See 5.1.1.1 (2) and 5.1.1.2 (7)), characteristic changes in organic material surfaces such as harness films. (In particular, the solar light absorbing ratio α increases due to browning of white harness films and the increase tendency has a temperature dependence.) In temperature analysis of on-orbit harnesses (in particular harness bundles), care shall be taken to prevent deviation from the harness' allowable working temperature in consideration of these deterioration factors. It shall be necessary, as needed, to take measures such as harness routing to surfaces out of direct exposure of ultraviolet rays and installation of light shielding plates or protection by SLI and the like.
Refer to the Collection of Wire Derating Test Data (JERG-2-212-TM001) for details of ultraviolet-induced characteristic changes in surfaces of organic materials such as harnesses.

5.7.3 Atomic oxygen (AO) countermeasure design and analysis

Atomic oxygen exists in low orbit depending on the orbit height where earth-orbiting spacecraft aviate. This causes erosion in the materials employed in the solar array wings, resulting in significantly high erosion yields of metals such as silver and osmium and organic materials such as polyimide. As materials employed in the solar array wings, silver is used in solar cell electrodes, interconnectors and bus bars of solar cell array circuits while polyimide in panel substrate surface and SLI/MLI. The paddle shall be designed so that functional maintenance of the above-mentioned materials will not be affected by atomic oxygen erosion depending on the spacecraft orbit altitude and mission operation period. In erosion yield analysis, consideration shall be given to the incidence direction of AO. Among countermeasures are adoption of a proper material thickness for functional maintenance or coating by materials having a low reaction rate (such as gold plating for silver and ITO, SiO_2 or germanium coating for polyimide).
See the Radiation-proof Design Handbook JERG-2-143-HB001 for details of atomic oxygen distribution density and reaction rate of each material with respect to orbit altitude.

5.7.4 Debris/micro meteoroids countermeasure design and analysis

Because the details of possibility of collision of debris/micro meteoroids, collision influence and countermeasure design are described in the Space Debris Protective Design Standard (JERG-2-144), countermeasure design is described here in view of improving the reduction in generated power.
The solar array wing is a piece of externally installed equipment which is directly exposed to outer space. It is difficult to take protective measures against debris/micro meteoroids for the solar cell array circuit, in particular, due to its functional action. Thus, it is desirable to consider the following countermeasures in view of improving the reduction in generated power.

(1) Reduction of influence by means of separation of power harness HOT/RTN lines and increase in the number of power harness routes in the system EMC allowable range.
(2) Installation of the power harnesses closely contacted with the structure elements to the extent possible for improving the shielding effect. (However, proper stress relief must be maintained.) When using both harness installation and thermal control materials installation, proper consideration shall be given to thermal design and thermal control materials grounding.
(3) Array circuit ground fault prevention by adoption of bleeder resistance

5.8 Anti-charging/discharging countermeasure design and analysis

The details of anti-charging/discharging countermeasure design and analysis are specified in Charging/Discharging Design Standard (JERG-2-211) and shall be in conformity with the applicable design requirements (JERG-2-211 5.4 “Solar Cell Panel”), analysis requirements (JERG-2-211 6.2.2.4 “Computer Aided Model and Analysis Results”) and test requirements (JERG-2-211 6.4 “Solar Cell Panel”). Except for panel surface side (array circuit side) at present, there are no standard design indicators equivalent to JERG-2-211, Section 9 Appendix-II “Test Results of Inter-solar Cell Distance and Discharge” with respect to the panel back side, paddle drive mechanism and other spots. Thus, it shall be necessary to identify critical spots at which self-lasting secondary discharge attributable to primary discharge may occur and judge the necessity of testing in line with the regulations specified in JERG-2-211 6.4.1.3 “In the case no testing is required.”

5.9 Installation design

Cautions in solar array wing installation design are described here.

(1) Precautions for installation for improving reliability and survivability

Table 5-1 shows the precautions for installation for improving reliability and survivability. Installation is carried out for improving reliability and survivability. Failure to pay attention during installation to the points listed in the table may result in failure or contradiction to design intent.

Table 5-1 Precautions for installation design

Target parts or installation \ Major role	Sustaining discharge control	Ground fault current control	Secondary failure avoidance	Notes for installation design
(1) Grouting between cell gaps	○	N/A	N/A	[1] When performing grouting between solar cells to control secondary discharge, proper deforming shall be performed during mixing to leave no voids. [2] Baking shall be performed properly to prevent grouting materials from becoming a contamination source in orbit.
(2) Bleeder resistance	N/A	○	N/A	[1] To avoid contact with discharge plasma, the resistance shall be molded by RTV or covered with insulating (heat shrinkable) tube without exposing the electrodes. If it is difficult to perform installation due to stress relief, consideration shall be given to prevent discharge by providing a sufficient distance and the like. [2] Resistance values shall be selected properly in accordance with the amount of on-orbit influent electrons. Selection shall be done in consideration of normal rated power of resistance in case of ground fault. [3] In case of multiple circuit ground faults concurrently with shunt ON action, the multiple currents exceeding the shunt element rating may flow in the same shunt element. Thus, when installing a bleeder resistance, resistance shall be installed between every panel, yoke and boom to the extent possible. [4] In the case of the paddle mounting the shunt yoke, a ground fault of power harness (HOT) with respect to the boom/yoke is single failure mode of bus short circuit. Thus, it is desirable to install a bleeder resistance to the boom/yoke. [5] When installing a bleeder resistance, deployment synchronization mechanism (synchronization wire) may become an electrically conducting path between panels. To function the original design intention for the above-mentioned [3], insulation between the panel hinge and the panel shall be assured. Care shall be given to prevent the synchronization wire from coming into contact with the substrate or steady strain, which may cause in contact conduction between the panel hinge and panel.
(3) Blocking or isolation diode	○	N/A	○	[1] To avoid contact with discharge plasma, the resistance shall be molded by RTV or covered with insulating (heat shrinkable) tube without exposing the electrodes. If it is difficult to perform installation due to stress relief, consideration shall be given to prevent discharge by providing a sufficient distance and the like. [2] Because a poor soldering in component electrodes and power harnesses may directly cause power drop, it is desirable to check for proper thermal cycle tolerance and install a blocking diode serving as a redundant route.
(4) Bypass diode (panel back side installation is target)	N/A	N/A	N/A	[1] To avoid contact with discharge plasma, the resistance shall be molded by RTV or covered with insulating (heat shrinkable) tube without exposing the electrodes. If it is difficult to perform installation due to stress relief, consideration shall be given to prevent discharge by providing a sufficient distance and the like.
(5) MTC/D-sub connector	N/A	N/A	N/A	[1] In accordance with the installation example of Insulation Design Standard (JERG-2-213), the connector back face (harness connection side) shall be molded by RTV or covered with insulating (heat shrinkable) tube without exposing the electrodes to avoid contact with discharge plasma.
(6) 8W8P connector	N/A	N/A	N/A	[1] In accordance with the installation example of Insulation Design Standard (JERG-2-213), the connector back face (harness connection side) shall be molded by RTV without exposing the electrodes to avoid contact with discharge plasma.
(7) Panel ground line harness capacity and ground line slipping capacity	N/A	N/A	○	[1] When an array circuit ground fault is permitted in the design without installing a bleeder resistance, current capacity shall be selected in consideration of avoidance of secondary failures caused by current concentration to the left line due to multiple ground faults.

(2) Precautions for installing wire harnesses

When wire harnesses are routed and fixed on the panel surface and the like, consideration shall be given to the following points.

- In consideration of thermal shrinkage of wires with respect to the environment temperature range, wire stress release between fixed points shall be provided.
- Consideration shall be given to the minimum bend polarization rate in bending wires such as

when passing through panel through-hole. Care shall be given to susceptibility of film cracks at bend sections under a complex space environment (radiation and thermal cycle).

- When routing is performed near sharp edges such as substrate edges or when harness films may be worn under a vibration environment or near movable sections such as deployment action, covering of sharp edges and protection covering of harness shall be carried out.
- When fixing a harness bundle to the boom/yoke section using a tie wrap, consideration shall be given to a proper restraint force and harness film covering in harness bundle restraining sections to avoid cold flow of harness films.
- If noise causes a critical influence on the functions of signal lines, the provision of routes separated from power lines or the use of shield lines.
- Harness routing in boom/yoke sections shall be performed so that development of a magnetic field associated with a current loop will be minimized in principle. However, in view of survivability requirements, robustness requirements or noise immunity requirements, when a current loop is developed due to multiple routes, adjustment shall be made with the system with respect to the allowance and specified in ICD and the like.

5.10 Definition of interfaces

5.10.1 Definition of external interfaces

Interfaces (electrical, thermal and mechanical) between a paddle subsystem and other subsystems shall be clearly illustrated.

5.10.2 Interface demarcation

Electrical, mechanical and thermal interface demarcations shall be clearly illustrated.

5.10.3 Interface conditions

In the interface conditions defined in 5.10.1, conditions at least for the items listed below shall be organized and clarified.

5.10.3.1 Electrical interface conditions

As electrical interfaces, conditions at least for the items listed below shall be organized and clarified for the power supply system, telemetry/command system, attitude control system and other interfaces defined in 5.1.

5.10.3.1.1 Power supply system interfaces

It shall be necessary to organize and clarify the interface conditions related to the power supply-related interfaces between a paddle drive mechanism and power supply system which transfer the power generated by the paddle subsystem, voltage/current interfaces with the shunt circuit, primary power source interfaces required by the paddle system such as damper heater, and ODC pass interfaces related to explosive device ignition power. For shunt action procedures of the array circuit, if there are no special system requirements, make correspondence between the shunt level in the power supply system and the shunt action array circuit understandable and clarify the measures in ICD.

Other interfaces shall be in conformity to the regulations specified in Power Supply System Design Standard (JERG-2-214).

5.10.3.1.2 Telemetry/command interfaces

All general requirements for telemetry/command shall be in accordance with Electric Design Standard (JERG-2-200). Details of TeleCommand item setting standard and electrical signal interfaces shall be in accordance with the regulations specified in Examples of Signal Interface (JERG-2-200-TM001).

Recommended telemetry items required for a solar array wing system are shown in Table 5-2.

Table 5-2 Recommended telemetry items

Telemetry item	Target component	Remarks
Solar cell panel temperature	Solar cell panel	• It is desirable to directly measure the solar cell temperature but it is difficult to install a temperature sensor on the panel surface side. Thus, install a temperature sensor on the panel backside and grasp the solar cell temperature based on the thermal math models and thermal vacuum test result. • It is ideally desirable to measure each solar cell panel. However, when there are restrictions on the telemetry count and the number of slip rings, it is desirable at least that the farthest and nearest panels of the spacecraft structure be measured.
Boom/yoke temperature	Boom, yoke	• It is desirable to perform measurement in the proximity of the harness bundle installed.
Harness temperature	Boom/yoke section fitting harness bundle	• It is desirable to perform measurement at the center of the harnesses bundled in the boom/yoke section.
Paddle deployment status	Solar array wing	• It is desirable to monitor the deployment status of each hinge by using the micro switch and resistance.
Damper temperature	Damper	
Paddle drive unit temperature • Shaft temperature • Flange temperature	Paddle drive unit	• In the paddle drive unit, it is desirable to measure the temperature difference between the inner ring and outer ring of the bearing.
Paddle drive mechanism angle signal • Potentiometer • Segment switch • Reference point sensor	Paddle drive unit	• The reference point sensor is not a mere piece of telemetry but a signal used for the control system. Thus, it shall be designed in consideration of the interfaces with the attitude control system.
Solar tracking error angle signal	Solar tracking sensor	• The solar tracking sensor is not a mere piece of telemetry but a signal used for the control system. Thus, it shall be designed in consideration of the interfaces with the attitude control system.
Shunt temperature	Shunt	Applied to the spacecraft mounting the shunt yoke.
Shunt drive signal	Shunt	

5.10.3.1.3 Attitude control system interfaces

As solar tracking interfaces, the interfaces related to paddle drive mechanism drive and the interface conditions related to paddle drive mechanism angle detection shall be organized and clarified.

5.10.3.1.4 Other electrical interfaces

When the solar tracking sensor, deployment monitor (DM) and accelerometer are installed to the solar array wing, the electrical interface conditions with the applicable equipment shall be clarified.

5.10.3.2 Mechanical interface condition

The panel size and envelope area in solar array wing storage state shall be specified.
The mechanical interface conditions related to holding and installation between solar array wing and paddle drive mechanism, between paddle drive mechanism and structure, and between PDL and structure shall be clarified. When the solar tracking sensor, deployment monitor (DM) and accelerometer are installed to the solar array wing, the mechanical interface conditions related to applicable holding and installation shall be clarified.

5.10.3.3 Thermal interface condition

The thermal interfaces (including calculation formulas of conductive heat exchange and radiant heat exchange amount) conditions related to holding and installation sections between solar array wing and paddle drive mechanism, between paddle drive mechanism and structure, and between PDL and structure shall be clarified. When the solar tracking sensor, deployment monitor (DM) and accelerometer are installed to the solar array wing, the thermal interface conditions related to applicable holding and installation sections shall be clarified.
For thermal analysis input conditions, it shall be necessary to clarify the interface conditions (combination of the solar light incidence conditions from launch to each on-orbit operation mode and the orbit thermal input constants (solar light strength, albedo constant and earth infrared intensity)) including attitude conditions at the time of attitude abnormality and specify the conditions in application documents such as thermal design standard.
For the above-mentioned solar light incidence conditions, care shall be given to the influence of reflected light from the system structure thermal control materials (such as OSR and Ag-Teflon).

5.10.3.4 Dynamics interface conditions

For design, analysis and verification in 5.2.4.2, 5.3.4, and 5.6, the following dynamics interface conditions shall be specified and interfaced with the spacecraft system. As the solar array wing system is coupled mutually with the spacecraft body and other subsystems, the dynamics will change the behavior. Thus, a torque interface with the paddle system end fixed rigid is not proper because it is impossible to express the motion of the spacecraft and paddle system by the torque interface. Therefore, the spacecraft system shall perform interfacing using a dynamics model which allows for dynamics analysis of all machines.

(1) Paddle mass characteristic

- Storage state, and mass, barycenter position, inertia moment, product of inertia, tolerance and allowable range and regulations coordinate system under a deployment state (after normal deployment and, as needed, at deployment abnormality)

(2) Paddle external disturbance characteristic

- Surface characteristics and surface area for solar radiation pressure calculation
- Residual magnetic moment

(3) Paddle flexible structure characteristics

- Natural frequency, zero-order coupling coefficient vector, primary coupling coefficient vector, mode damping ratio, tolerance and allowable range, regulations coordinate system (after normal deployment, and, as needed, at deployment abnormality)

(4) Paddle heat distortion deformation

- Deformation models, deformation profile, installation point torque profile, quasi-static deformation (warpage) attributable to heat during orbit

(5) Paddle heat-induced dynamics

- Heat-induced dynamics models, deformation profile, installation point torque profile, barycenter variations profile

(6) Paddle deployment dynamics

- Deployment configuration profile, deployment angle profile, deployment signal profile, deployment time, latch shock torque, installation point torque profile, attitude angle profile, attitude rate profile (at least two temperature conditions including high-temperature and low-temperature conditions)

(7) Paddle solar tracking control
- Angle detection sensor models
- Drive actuator models

(8) Paddle drive disturbance
- Step motor, harmonic drive, gear, paddle flexible structure mode, drive profile or driving rules parameter
- Drive disturbance, vibration profile, installation point torque profile, barycenter variations

5.10.3.5 Visual field interface

Install a solar tracking sensor. When there are requirements of visual field interference, visual field interfaces shall be specified in ICD and the like.

5.10.3.6 Movable section envelope interfaces

The action envelope (range in which no mechanical interference occurs with spacecraft system) during deployment action of the solar array wings and on-orbit solar tracking rotation shall be specified in ICD and the like.

5.10.3.7 Interfaces with ground support equipment

It shall be necessary to clarify a set of ground support equipment for manufacturing, assembly, test, maintenance and inspection of the solar array wing system and define electrical and mechanical interfaces.

6. Verification

It shall be verified that a solar array wing system is designed properly and the requirements related to generated power and the like are satisfied. Verification shall be carried out in the PFM/FM component test, intrasystem combined test, and system test in the design analysis and manufacturing/test stages based on the results of manufacturing and testing of EM and the like up to the design stage.

6.1 Deployment/verification plan

In devising a deployment plan, perform technology analysis of the requirement specifications based on similarity and analysis whether element test and deployment test on the components shown in Figure 4-1 are necessary or not. Device a deployment/verification plan in which manufacturing and testing of the necessary element test models and deployment test models are clarified. Organize the plan as a requirement confirmation matrix and an environment test matrix for design and analysis requirements specified in 5. Table 6-1 shows an example of engineering models with consideration given up to system test.

Table 6-1 Example of definition of engineering models

Item	Model name	Remarks (objective)
Solar array wing	Charging/discharging coupon panel	Used for discharge tolerance evaluation
	Coupon panel	Used for checking for thermal cycle life performance
	(Other element models)	
	Deployment hinge section model	Acquisition of deployment element parameters (such as torque and rigidity)
	Holding/releasing mechanism section model	Shock level confirmation
	etc.	
	Structure dummy	Used for system structure model
	Thermal dummy	Used for system thermal model
	EM	Used for total design verification
	PFM/FM	Used for flight quality verification
PDM	Operating life test model	Used for checking movable sections such as slip rings and actuators for operating life
	Structure dummy	Used for system structure model
	Thermal dummy	Used for system thermal model
	EM	Used for paddle combination test and system electric model
	PFM/FM	Used for flight quality verification

For the models shown in Table 6-1, precautions for manufacturing models shall be as follows.

(1) Coupon panel

There have been fractures of solar cells and interconnector welds under a thermal vacuum environment in a past project. When changing substrates, solar cells or interconnectors, it shall be necessary to perform verification in a coupon panel test and the like.

Thus, coupon panels shall be manufactured with materials and design and manufacturing processes similar to those of flight articles in consideration of the following points.

- In addition to solar cells, parts installed to flight articles (including resistance, diodes, connectors, thermal control materials and, as needed, brackets such as hold down pads) shall be simulated.
- It is desirable to include a solar cell replacement simulation.
- Surface film joints (flight articles having joints) shall be simulated.
- CFRP face sheet layer number (including doubler boundary section) shall be simulated.

See Solar Cell Panel Quality Assurance Handbook (JERG-2-216-HB001) for requirements for test and evaluation. When adopting an array circuit isolation diode, it is desirable to perform a cycle test with coupon panels with energization state in separate diode section assessed.

(2) Charging/discharging coupon panel

- For intercell gap, the minimum gap shall be simulated within the manufacturing tolerance standard range.

For items other than those above, simulation shall be reflected as in the case with coupon panels. See Charging/Discharging Design Standard (JERG-2-211) for requirements for test and evaluation.

(3) Structure dummy

After being assembled in the spacecraft system structure model, the structure dummy shall be supplied to verify the structural design of the system and check the structure interface design of the spacecraft system and solar array wing system for validity.

The structure dummy shall be manufactured in principle according to the following requirements. Unless otherwise specified, the following requirements shall be applied to both the solar array wings and paddle drive mechanism.

- The exterior shape and dimensions shall be the same with PFM in principle.
- The installation hole position, installation hole diameter (installation fastener size), and plate thickness of installation section shall be same with PFM including tolerance.
- The mass and moments of inertia shall be same with the PFM nominal value. (Allowable errors shall be adjusted with the system.)
- The mass center position shall be same with the PFM nominal value. (Allowable errors shall be adjusted with the system.)
- Shall be same with the main mode. Shall have the same strength with PFM. Normally, notching other than the acceleration equivalent to mass center shall not be allowable.
- The natural frequency of the main mode shall be same with PFM. (Allowable errors shall be adjusted with the system.)

- PFM occurrence shock shall be simulated as needed.
  (Applied only to the solar array wing structure dummy.)

(4) Thermal dummy

- Normally, the main objective of system thermal vacuum test is to check the thermal control state under operation state after solar array wing deployment. It is ideal that solar array wings are supplied in deployment state to system thermal vacuum test. However, it is extremely difficult to install solar array wings in deployment state due to equipment restrictions. Thus, a solar array wing thermal dummy shall be manufactured so that the dimension and shape (give consideration so that the form coefficient conforms as much as possible), heat capacity and thermo-optical characteristic are equal to those of the solar array wing flight articles in line with the evaluation purpose and evaluation items of system test where possible.
  However, solar array wings are exempted from evaluation in a system thermal vacuum test because the solar array wings are insulated from the spacecraft/PDM and independently thermal-controlled. For solar array wings having a small thermal bonding with the spacecraft structure due to the use of yoke and the like, it is allowed in some cases to perform testing with only the solar array wing installation interface hold-down installed to the structure without using the solar array wing thermal dummy. This is because of circumstances related to test configuration such as the power line connections from the solar array simulation power source to the PDM installed in the spacecraft structure.
- If it is difficult to use the flight article paddle drive mechanism in a system thermal vacuum test, it shall be necessary to manufacture a thermal dummy of the paddle drive mechanism having the dimension and shape, heat capacity, heat conduction (installation surface thermal conductance), thermo-optical characteristic, and calorific value equivalent to those of flight articles to the extent possible without interfering with the evaluation purpose and evaluation items of the system test and then to supply the dummy to the system thermal vacuum test.

6.2 Manufacturing and testing

Figure 6-1 shows an example of manufacturing/test flow of proto-flight in solar array wing system. Precautions (reflection of nonconformance cases) for each process of manufacturing/test flow are listed in the notes in the figure and 6.2 (1). Review the test implementation items and implementation procedure properly before verifying the design and analysis requirements in Paragraph 5. The verification items for requirements confirmation matrix and environment test matrix shall be satisfied.

Recommended check items in in-process process are shown in Table 6-2. When checking the solar array wings for flight quality soundness, important items which cannot be checked in a complete state shall be checked for soundness during in-process operation by setting proper criteria.

Table 6-2 In process recommended check items

Target category		Target unit	Check item
Solar cell panel	Substrate	Actual machine	Polyimide surface insulation resistance inspection Dimple developmental rate, dimple depth
		Parallel sample	Flatwise strength test 3-point or 4-point bend test
	CIC peel strength	Condition setting sample	Checking interconnector for welding strength
Deployment synchronization mechanism	Hinge	Actual machine	Sliding torque/deployment torque at deployment action
			Rigidity at deployment
			Synchronization wire (tightening portion) proof load tolerance
	Rotary damper	Actual machine	Damping rate
Holding and releasing mechanism	Preload bolt	Actual machine	Proof load tolerance

(1) Precautions in manufacturing/test flow (reflection of nonconformance cases)

<Solar array wing>

- In-process operation
  - Defective appearance of substrate/CIC due to handling (such as flaws, hollows on substrate and cracks in cover glass)
  - Re-melting of solder in MTC connector heat shrinkable tube operation
- Proto-flight test
  - Defective appearance of substrate/CIC due to handling (such as flaws, hollows on substrate and cracks in cover glass)
  - Incomplete deployment due to poor maintenance of deployment supporting device
  - Damage in mechanism components due to poor notch level setting
  - Deviation from standard due to shock developed by explosive device
  - Temperature control standard value deviation in thermal vacuum test

<Paddle drive mechanism>

- In-process operation
  - Increased resistance value of slip rings and noise generation due to insufficient run-in
  - Peeling in incompletely plated slip ring section and sliver developed by intrusion of foreign matter into brush
- Proto-flight test
  - Standard deviation due to temperature change at reference point sensor trigger point
  - Defective appearance caused by handling (flaws on cabinet and connectors)
  - Temperature control standard value deviation in thermal vacuum test

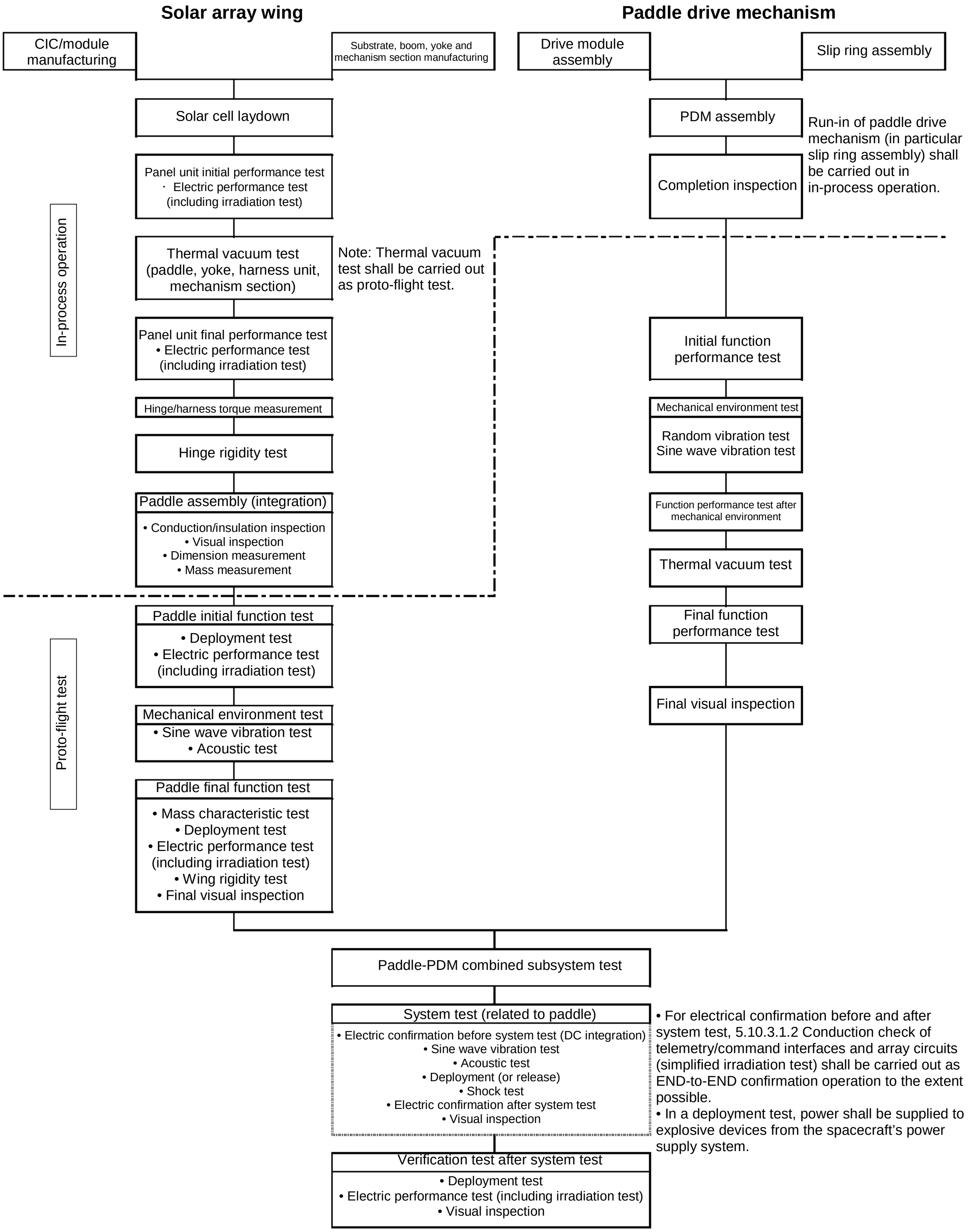
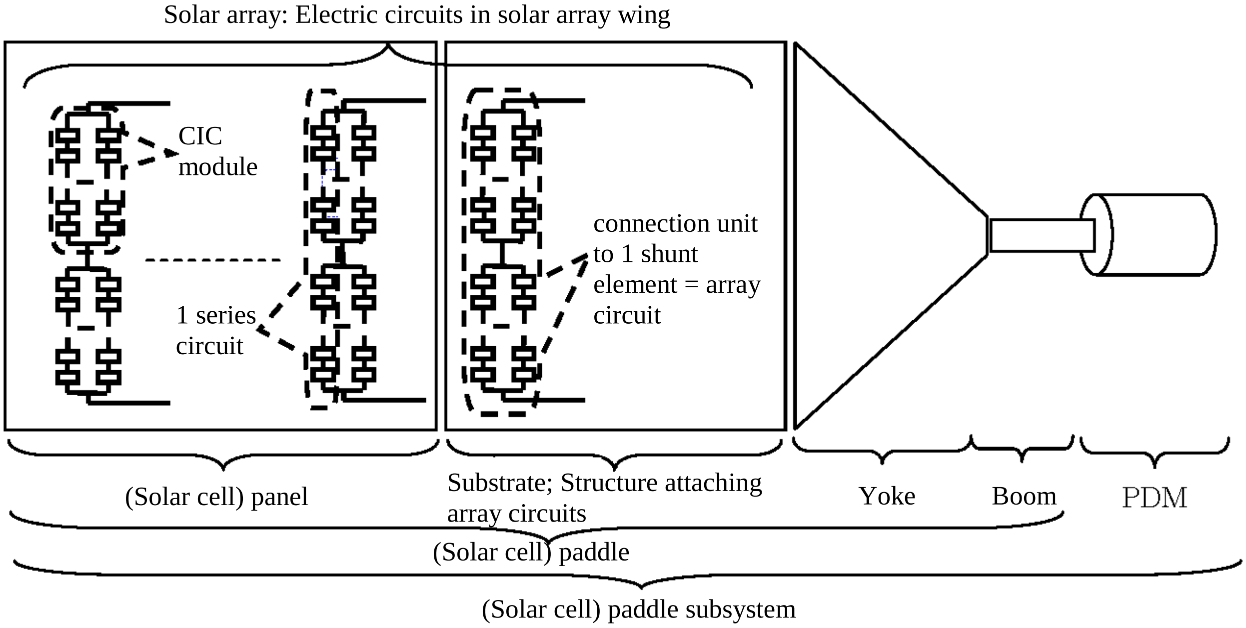


Figure 6-1 Manufacturing/test flow

Appendix I Definition of name

- Definition of name

The name of components in the solar array wing system used in the design standard document shall be defined as follows:



[1] Solar cell: Solar cell element itself. Simply called cell or bear cell.

[2] CIC (Cover-glass Integrated Cell or Connector Integrated Solar Cell): Solar cell installed with interconnector and cover glass.

[3] CIC module: Unit where CIC is connected to arbitrary series-parallel circuit unit with interconnector

[4] Array circuit: Parallel-connected unit corresponding to 1 shunt element

[5] One series circuit: One series circuit in array circuit.

[6] Substrate: Structure attaching array circuits

[7] Solar cell panel: Unit where array circuits are assembled to one substrate

[8] Boom: Beam element which is a part of structure coupling the paddle drive mechanism edge and solar cell panel edge.

The solar array wing system has no boom in some cases.

[9] Yoke: Y-bend element which is a part of structure coupling the paddle drive mechanism edge and solar cell panel edge and attached to the solar cell panel side. The solar array wing system has no yoke in some cases.

[10] Hinge: Mechanism which rotates two structures relatively around itself

[11] Solar array wing: Unit in which solar cell panels, boom and yoke are assembled.

Also called as PAD or SAP.

[12] Solar array: Generic name for power generation circuits of solar array wing

[13] PDM (Paddle Drive Mechanism): Paddle drive mechanism. Also called SADM (Solar Array Drive Mechanism).

[14] Solar array wing system: Entire subsystem including solar array wing and paddle drive mechanism. Also called SPS or PDL.