Thermal Design Strategy Utilizing Transformable Structure of Spacecraft

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As an innovative spacecraft system, a transformable spacecraft is proposed, which is composed of many form elements coupled with each other by an internal force actuator and capable of transferring the form to each other. In addition to being able to switch the functions of the spacecraft largely by changing the form on the orbit, the transformable spacecraft is characterized by its ability to perform propellant-free attitude control using internal forces and nonholonomic properties. Moreover, not only functions of spacecraft can be changed by shape change, but also various characteristics caused by shape can be changed on orbit. Thermal characteristics, structural characteristics, etc. can be mentioned, but if these are positively used, it is possible to select the characteristics suitable for the mission on the orbit. In this research, we focus on the thermal characteristics, evaluate the relationship between the panel configuration of the transformable spacecraft and the thermal environment by thermal analysis, and aim to obtain a guideline of the panel configuration suitable for the observation mode. In addition, from the relationship obtained, the conditions such as the number of panels and the layout of the transformable spacecraft to realize the requirements of the thermal environment are clarified, and the system design method of the transformable spacecraft is considered. Furthermore, since the panel configuration also affects the power, attitude, etc., optimization that takes them into consideration is considered.

可変構造を用いた宇宙機の熱設計

可変構造宇宙機(Transformable spacecraft)とは，互いに内力アクチュエータで結合された多数の形態要素から構成され，相互に形態を移行できる多機能を有した宇宙機のことである．可変構造宇宙機は軌道上で形態を変化させることによって宇宙機の機能を大きく切り替えられることに加え，内力と非ホロノミック性を利用した推薬不要の姿勢制御ができることが特徴である．また，可変構造宇宙機は形態変化により機能が変えられるだけでなく，形態に起因する様々な特性を軌道上で変化させることができる．熱特性，構造特性などが挙げられるが，これらを積極的に利用すれば軌道上でミッションに応じて適した特性を選択することができる．本研究では，特に熱特性に着目し，可変構造宇宙機のパネル形態とミッション部の熱環境との関係を熱解析により評価し，観測モードに適したパネル形態の指針を得ることを目的とする．また，得られた関係から，ミッション部の熱環境の要求を実現するための可変構造宇宙機のパネル枚数や配置等の条件を明らかにし，熱特性を考慮した可変構造宇宙機のシステム設計手法を提案する．さらに，パネル構成は電力や姿勢等にも影響するため，それらも考慮した最適化を検討する．

Key Words: *Thermal design, Transformable structure, multi-objective optimization*

1. Introduction

As shown in Fig. 1, a transformable spacecraft is a multi-function spacecraft that is composed of several arbitrary structure that are connected to each other by internal force actuators and that functions can switch between each other. As a result, the function of the spacecraft can be switched greatly by changing the form in orbit, and the attitude control without propellant using internal force, nonholonomic property, and solar radiation pressure (SRP) is possible. The transformable spacecraft not only can change its function by changing its form, but also can change various characteristics due to its form in orbit. Thermal characteristics, structural characteristics, etc. can be mentioned, but if these are used positively, characteristics suitable for the mission can be selected in orbit. In this study, we focus on the thermal characteristics the stability of the attitude due to SRP and consider the change of the characteristics due to the shape change. In general spacecraft, thermal control is performed by changing the attitude or using thermal control equipment such as heater. However, if a transformable structure is used, it is possible to passively cool and heat specific parts of the spacecraft due to changes in the thermal environment such as the construction of a heat shield. On the other hand, attitude control without propellant is possible by adjusting the SRP received using changing its form. Since these are realized by morphological changes, both are coupled. Therefore, it is necessary to consider the trade-off when considering the entire system.

As an example, the transformer (Fig. 2) under investigation leading JAXA is equipped with an infrared telescope that requires cooling as the mission equipment [1], and we consider applying a thermal design utilizing the transformable structure. If the sunlight is blocked by the panel, the thermal control can be performed without changing the attitude of the mission unit, so the flexibility

of the mission can be improved. In addition, studies on attitude control for transformable structure spacecraft using SRP have been reported [2]. In conventional spacecraft development, thermal design and attitude control like these have been studied separately. However, it can be said that it is inefficient to examine these separately in the design of the transformable spacecraft.
Therefore, the purpose of this study is to evaluate the relationship between the panel form and material of the transformable spacecraft and the thermal environment of the mission section by thermal analysis, and to obtain a guide for the panel form suitable for the observation mode. In addition, from the relationship obtained, the conditions such as the number of panels and materials of the transformable spacecraft to realize the thermal environment requirements of the mission section will be clarified. Furthermore, we propose an optimal system design method for transformable spacecraft using genetic algorithm (GA) that considers not only the thermal environment but also attitude stability and power generation capacity.

2. Analysis method

In this chapter, the analysis method for heat and attitude used in the proposed method and the optimization method GA are explained.

2.1 Thermal analysis method

In this study, the thermal network method was used for thermal analysis. The thermal network method is a method of performing thermal analysis using the electric circuit method based on the similarity between the electric circuit and heat. e similarity between an electric circuit and heat.

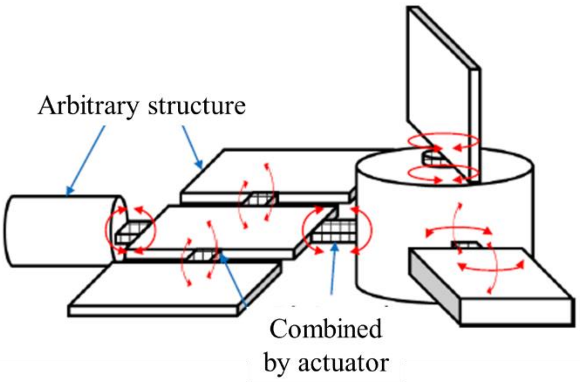

Fig. 1. Example of Transformable Structure of Spacecraft.

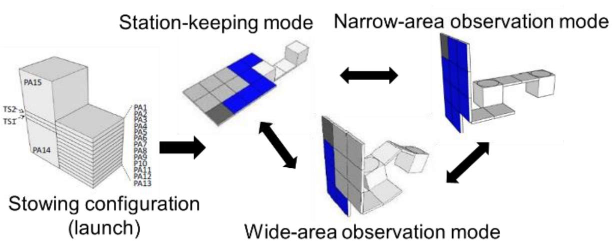

Fig. 2. Figure of Transformer for each mode.

By dividing the object of analysis into several regions (mesh) and thermally connecting the nodes representing the region, thermal analysis with spatial and temporal distribution can be performed [3]. The thermal network method expresses the conservation equation of heat energy considering the heat balance by applying Kirchhoff's law for each node. This is called a nodal equation, and there are as many nodes as there are. By solving these simultaneously, the temperature of each node can be obtained. Thermally connected nodes exchange heat by conduction, convection, or radiation. By considering the thermal resistance obtained by solving the basic equations corresponding to these, each nodal equation can be derived. In particular, in this study, heat transfer due to convection can be ignored because the analysis target that exists in space and does not have an active thermal control device such as a cooler inside.

The amount of heat transferred by heat conduction and radiative heat transfer is given by Eqs. (2) and (5) using heat conduction conductance K_{ij} and radiation conductance R_{ij}. Heat conduction conductance and radiative conductance are given by Eqs. (3) and (6). These are derived by the Fourier equation (Eq. (1)) that explains the phenomenon of heat transfer in the individual and the Stefan-Boltzmann law that expresses the relationship between the energy of electromagnetic waves emitted by thermal radiation and the temperature.

$$dQ = -\lambda \frac{d\Theta}{dx} \tag{1}$$

$$Q_i^K = K_{ij}(T_i - T_j) \tag{2}$$

$$K_{ij} = \lambda A_K / l \tag{3}$$

$$I = \sigma T^4 \tag{4}$$

$$Q_i^R = R_{ij}({T_i}^4 - {T_j}^4) \tag{5}$$

$$R_{ij} = \sigma A_R F_{ij} \tag{6}$$

Where Q is the amount of heat moving between the nodes, T is the temperature at a certain node, λ is the thermal conductivity, A_K is the cross-sectional area of the heat conduction region, l is the distance between the nodes, σ is the Stefan-Boltzmann constant, A_R is the radiation Area, F_{ij} is a view factor, and subscripts *i* and *j* are node numbers.

View factor is a dimensionless quantity that geometrically represents the proportion of the amount of heat received by another surface. This view factor is determined only from the geometric relationship of the two faces and is given by equation (7).

$$F_{ij} = \frac{1}{A_i} \int_{A_i} \int_{A_j} \frac{\cos \Phi_i \cos \Phi_j}{\pi r^2} dA_i \, dA_j \tag{7}$$

Where Φ_i and Φ_j are the angles between the normal of the micro-surfaces dA_i or dA_j and the lines connecting the micro-surfaces, and *r* is the length of the line connecting the micro-surfaces.

2.2 Attitude stability analysis method

This section describes the attitude stability analysis method used in this study. In 2016, Ono et al. formulated the attitude motion of spacecraft under SRP by introducing an SRP model that considers absorption, specular reflection, and diffuse reflection [4]. However, in this formulation, it is assumed that the spacecraft does not deform or is a very small deformation, and as it is, it cannot be adapted to the attitude motion of the transformable spacecraft. Therefore, Chujo proposed a model in which the preceding formulation was extended to have large deformations [5].

As shown in Fig. 3, the inertial coordinate system, the sun-oriented coordinate system, and the spacecraft fixed coordinate system are defined. Also, the point where the plane perpendicular to the z_S axis of the sun-oriented coordinate system intersects with the z_B axis of the spacecraft fixed coordinate system is expressed by the out-of-plane sun angle φ and the in-plane sun angle θ. The rotation angle around the z_B axis is ψ. The SRP force considering absorption, specular reflection, and diffuse reflection (Lambert reflection) is expressed as Eq. (8).

$$\begin{aligned} \mathrm{d}\boldsymbol{f} = -\frac{S_0}{c}\left(\frac{1}{AU}\right)\big[&|\boldsymbol{s}\cdot\boldsymbol{n}|(C_a + C_d)\boldsymbol{s} \\ &+ (\boldsymbol{s}\cdot\boldsymbol{n})\{B_f C_d + C_a\kappa \\ &+ 2C_s|\boldsymbol{s}\cdot\boldsymbol{n}|\}\boldsymbol{n}\big]\mathrm{d}A \end{aligned} \tag{8}$$

Where $\mathrm{d}\boldsymbol{f}$ is the SRP force acting on the micro surface, S_0 is the solar constant, c is the speed of light, AU is the solar distance, C_a is the absorptance of micro surface, C_d is the diffuse reflectance of micro surface, and C_s is the specular reflectivity of micro surface, B_f is the Lambertian constant, $\mathrm{d}A$ is the area of the micro surface, $\boldsymbol{s}$ is the sun direction vector, and $\boldsymbol{n}$ is the normal vector of the micro surface.

In the model proposed by Chujo, when the solar angle (φ, θ) is not large, the SRP torque acting on the entire spacecraft is expressed by parameters determined by the shape and optical characteristics of the spacecraft. By using this SRP torque, the equation of motion of the spacecraft can be expressed by parameters determined by the shape and optical characteristics of the spacecraft. The system design method proposed in this study also determines the eigenvalue of this equation of motion to determine stability.

2.3 Genetic algorithm (GA)

GA is a model that simulates the mechanism of evolution, inspired by the inheritance and natural selection of biological systems in nature, and was proposed by Holland et al. [6, 7]. In GA, a virtual group of organisms (individual population) with genes (parameters) is expressed in a computer. GA performs generation change simulation so that the probability that an individual (candidate solution of the target problem) adapted to a predetermined environment (evaluation function, etc.) will leave offspring is high. In other words, the optimal solution of the target problem is searched by evolving individual genes and populations. Specifically, first, solution candidates for the target problem are expressed as gene groups, and the fitness of each gene to the target problem is evaluated using an evaluation function. Depending on this fitness, the number of offspring left in the next generation is increased or decreased. After that, two individuals in the population are randomly combined, and each gene sequence is partially exchanged with a certain probability. In addition, for each individual, each gene is exchanged with an opposite gene with a certain probability (mutation). In this way, GA is an engineered model of natural selection (genes with higher fitness for the environment have a higher survival rate) and genetic phenomena (generation of new individuals based on parental information). The conventional optimization algorithm updates one search point, but GA treats multiple solution candidates as an individual group and updates the entire group together with the generation. It is considered that the optimal solution can be included in the population by configuring the initial population with search points distributed globally.

On the other hand, GA does not have a general method for solving the target problem, and has a drawback that there are many parameters to be selected by the program designer. However, in this study, the relationship between the material and shape of the target transformable spacecraft and the thermal environment is clarified, and by using this, parameters are set to adjust the sensitivity of gene mutation.

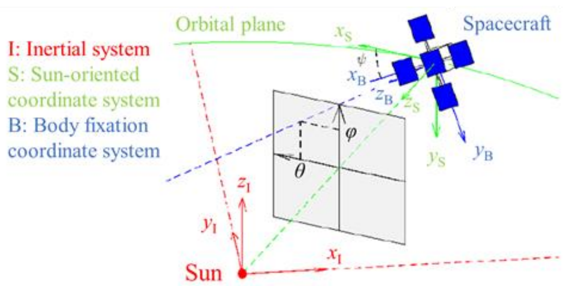

Fig. 3. Definition of coordinate system

3. Optimal system design using the proposed method

In this chapter, we show the material and shape of the transformable spacecraft on orbit that close to requirement of the temperature, power generation, and attitude stability that are obtained using the proposed method. We also discuss the validity and usefulness of the proposed method.

3.1 Analysis target and conditions

As shown in Fig. 5, we considered a transformable spacecraft with 10 panels connected by hinges. We searched for panel surface materials and shape on orbit that satisfy the requirements for attitude stability, spacecraft temperature, and power generation during observation as in the case of the transformer mentioned above as an example. Table 1 shows the panel size, specific heat, density, and thermal conductivity in the thickness direction inside the panel. The materials listed in Table 2 [8] are used as candidates for the panel surface. However, thin-film solar cells are mounted only on the sun direction surfaces of main body and panels A to F and are not mounted on other panels. Here, changes in thermo-optical properties due to radiation in outer space are not considered. The shape on the orbit is searched by changing the relative angle between adjacent panels. In this case, as shown in Fig. 5, the angle parallel to the adjacent panels is 0 deg. Set a node at the center of the both surface of each panel, and define the sun side as 1 and the opposite side as 2. The fixed body coordinate system is set at the center of the main body 1, and the normal of the main body 1 is the z axis. In this case, the sun side is the positive direction and the opposite side is the negative direction, and the relative angle of each panel follows this. As a constraint, panels A to D connected to the main body have a movable range of ± 60 deg, and panels F to I have a movable range that causes a bellows fold. The spacecraft takes a small circular HALO orbit and is always exposed to sunlight.

We describe the target temperature of each panel. The observation equipment (infrared telescope) is mounted on panel I and the required temperature during observation of 50 K is the required temperature of Panel I. The temperature of panel with thin-film solar cells is set to 330 K in consideration of power generation performance, and the temperature of main body and panel A to D opposite the sun are set to 300 K as the required temperature, assuming that bus equipment is mounted on these panels. In order to achieve this requirement, the thermal conductivity between panels was empirically set as shown in Table 1. In power generation capacity, the highest evaluation was made based on the maximum amount of power generated by the installed solar cell in order to supply power to observation equipment. The attitude stability was given higher evaluation as the out-of-plane sun angle φ and in-plane sun angle θ decreased. This is due to the assumption that the sun angle (φ, θ) is small in the process of stability analysis. As regards attitude stability, high evaluation was given to those whose real part of eigenvalues used for judgment approached zero or became negative.

3.2 Optimization using a single evaluation function

Using the proposed method, optimization was performed when the spacecraft temperature, power generation, and attitude stability were individually evaluated. The analysis results are shown below.

3.2.1 Optimization for spacecraft temperature

Table 3 shows the panel materials optimized for the spacecraft temperature. Fig. 6 shows the spacecraft configuration with these parameters, and Table 4 shows the temperature of each node. In this configuration, the power generation relative to the maximum power generation was 62.5%. The amount of heat input from the outside is important when considering the spacecraft temperature. The overall temperature decreases by reducing the amount of heat input from the outside, it is necessary to reduce the area where sunlight enters or to reduce the solar absorption rate of the panel surface. The analysis results are based on this, and the angle of the panel around the main body is made as small as possible to reduce the area where sunlight enters. However, focusing on panel C and panels E and F connected to C, there is a small area to see the sun, and it receives heat input from the sun. Panel G is also at an angle that allows sunlight to enter, but since a mirror is selected as the material for this panel and the solar absorptance is 0, direct external heat input is 0. From this result, it can be said that some parameters were not optimal values, and the number of generations or probability of causing mutation was not suitable.

3.2.2 Optimization for power generation

Fig. 7 shows the configuration of the spacecraft optimized for power generation. As a constraint, solar cells are mounted only on the main body and panels A to F. When these panels are directly facing the sun, power can be generated with maximum efficiency. From Fig. 7, panels A to F are facing the sun and the power generation efficiency

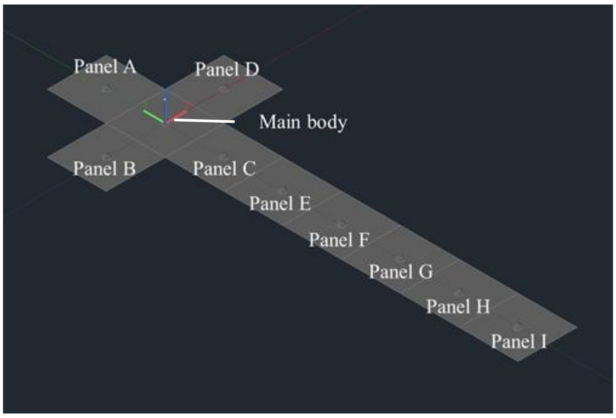

Fig. 5. Analysis target (10 panels)

Table 1. Various analysis factor

Panel	
Size [m]	1×1×0.1
Specific heat [J/kg/K]	8800
Density [kg/m^3]	2700
Thickness direction heat conductivity [W/m/K]	3.766
heat conductivity between panels [W/m/K]	Main body-Panel A, B, C, D: 100 Panel C-E: 50 Panel E-F: 25 Panel F-G: 10 Panel G-H: 5 Panel H-I: 1
Analysis	
Number of nodes	21
Initial temperature of panels [K]	293.15
Temperature of space [K]	2.73

Table 2. Heat and optical characteristics for analysis

	Solar absorptance α	Infrared emissivity ε	Specular reflectivity s	Diffuse reflectance d
Solar cell	0.854	0.6	0.086	0.06
Polyimide	0.37	0.68	0.375	0.255
Mirror	0	0.79	1	0
White paint	0.1	0.83	0.45	0.45
Black paint	0.9	0.83	0.05	0.05

was 99.95% as a percentage of the maximum efficiency. Panels C, E, and F are slightly angled, but such a small gap from the optimum parameter varies in location and value depending on the generation that ends the search. This is because some parameters change from the optimal solution due to mutation just before the search is completed. Such deviation from the optimal parameter can be reduced by adjusting the probability of causing mutation, but there is no general method for obtaining this, and it must be determined empirically.

3.2.3 Optimization for attitude stability

Table 5 shows the optimized parameters for attitude stability. Fig. 8 shows the spacecraft configuration at this condition, and Fig. 9 shows the transition of the smallest fitness (the highest evaluation) in the individual population of each generation. In the shape shown in Fig. 8, the real part of the eigenvalue becomes negative as shown in Fig. 9 and the attitude converges. Fig. 9 shows that the solution converges at the end of the search. It can be said that such a shape is difficult to search by human power, and it is possible to search efficiently by using this method focusing on attitude stability. In addition, the time required for optimization is approximately 1.2 seconds, and it can be said that this method is efficient from the time to obtain the solution.

Comparing the results of each optimization for temperature, power generation, and attitude stability, each advantageous shape is all different. In addition, because the evaluation is performed using a single evaluation function, the results for items that have not been evaluated are worse than when they are evaluated. In this way, temperature, power generation, and attitude stability are in a trade-off relationship, and it is indispensable to study them all at the same time for designing and operating a transformable spacecraft.

3.3 Optimization using a multiple evaluation function

Using the proposed method, we optimized the case where multiple items were evaluated. Table 6 shows the panel materials optimized for spacecraft temperature, power generation, and attitude stability. In addition, Fig. 10 shows the configuration of the spacecraft in this condition, and Table 7 shows the temperature of each node. The power generation efficiency in this configuration was 74.3%, and the attitude stability was evaluated to be approximately 1.7 × 10^{-7}. For each evaluation item, fitness decreases for all items when compared to the case of single evaluation.

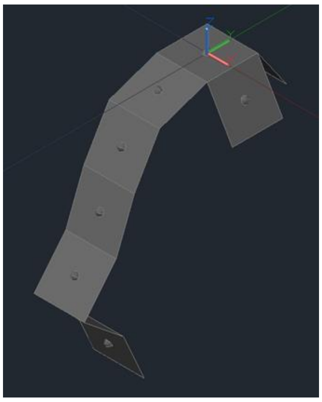

Fig. 6. The shape of spacecraft in case of optimum temperature.

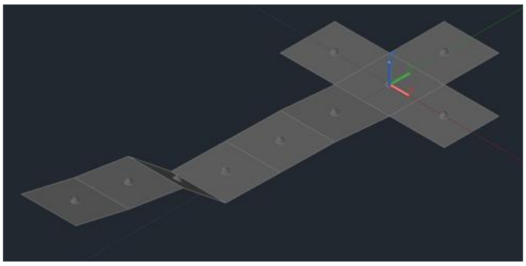

Fig. 7. The shape of spacecraft in case of max. power generation.

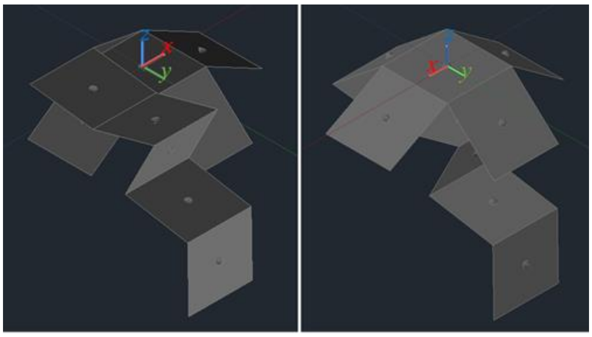

Fig. 8. The shape of spacecraft in case of best attitude stability

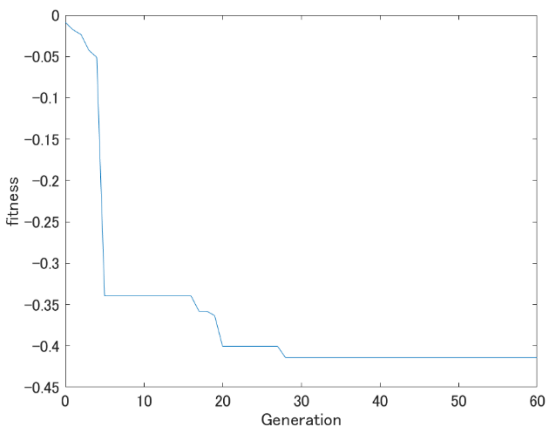

Fig. 9. Transition of smallest fitness in the population.

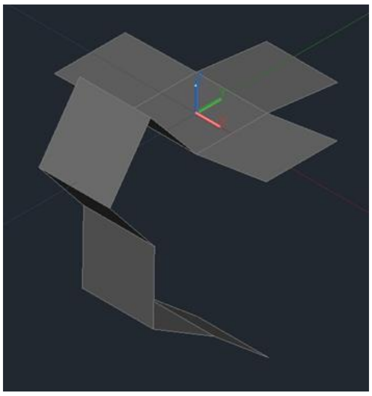

Fig. 10. The shape of spacecraft in case where temperature, power generation and attitude stability were evaluated.

However, overall fitness increased compared to the case of evaluation of single factor.

Thus, by comprehensively evaluating multiple items, it is possible to search for panel materials and forms with high overall fitness. The solution by the proposed method can be changed by adjusting the weighting to the evaluation items. For this reason, the proposed method can respond to various requirements such as what is important for design and operation, for example, the amount of power generation may be small, but the temperature requirement should be strictly achieved. It can be said that this is a useful technique.

4. Conclusions

For transformable spacecraft, which has been studied in recent years, we proposed a method for optimizing the form from the viewpoints of the temperature, power generation capacity, and attitude stability. The analysis method of thermal and attitude stability used in this method was explained, and the optimization method GA of the proposed method was explained. In addition, the algorithm of the proposed method was introduced and the search method of the optimal shape of the transformable spacecraft was shown. The proposed method was used to evaluate and optimize the temperature, power generation capacity, and attitude stability. These results show that the proposed method functions correctly as an optimization method. In addition, the shape when the optimization was performed with a single evaluation item was shown, and the optimal solutions under each condition were different shapes, indicating that there is a trade-off relationship with each other. In addition, multi-objective optimization that evaluates multiple items was performed using the proposed method. By evaluating items that are in a trade-off relationship with each other at the same time, it was possible to obtain shapes that took advantage of each shape.

On the other hand, as shown in the result of evaluating only the temperature, it can be confirmed that the fitness for temperature was higher than the optimal solution for other evaluation targets, but the solution was not optimal. The probability of mutation and the number of generations affect the fitness of the solution, but at present, these parameters are determined empirically. In the future, it will be necessary to study guidelines for determining these parameters. In addition, we have discussed the optimization of the shape of a transformable spacecraft whose number and position of panels have been determined so far, but the proposed method in this paper has dealt only with limited problems. In the future, algorithms that deal with more general problems, such as panel layout problems, are needed.

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Table 3. The result of optimized parameter regarding temperature

Panel	Main 1	A1	B1	C1	D1	E1	F1	G1	H1	I1
Material	Solar cell	Solar cell	Solar cell	Solar cell	Solar cell	Solar cell	Solar cell	Mirror	White	White
Panel	Main 2	A2	B2	C2	D2	E2	F2	G2	H2	I2
Material	Polyi-mide	Polyi-mide	Polyi-mide	Black	Mirror	Polyi-mide	Polyi-mide	Mirror	White	Black

Table 4. Temperature of each node in case of optimization of temperature

Panel	Main 1	A1	B1	C1	D1	E1	F1	G1	H1	I1
Temp. [K]	366.90	351.78	345.53	372.69	344.77	332.98	293.34	174.00	123.16	81.73
Panel	Main 2	A2	B2	C2	D2	E2	F2	G2	H2	I2
Temp. [K]	295.99	295.22	293.54	292.83	290.63	278.44	253.21	171.87	123.17	81.72

Table 5. The result of optimized parameter regarding attitude stability

Panel	Main 1	A1	B1	C1	D1	E1	F1	G1	H1	I1
Material	Solar cell	Solar cell	Solar cell	Solar cell	Solar cell	Solar cell	Solar cell	Mirror	Mirror	Black
Panel	Main 2	A2	B2	C2	D2	E2	F2	G2	H2	I2
Material	Polyi-mide	Polyi-mide	Polyi-mide	Mirror	Polyi-mide	Polyi-mide	Mirror	White	White	White

Table 6. The result of optimized parameter regarding temperature, power generation and attitude stability

Panel	Main 1	A1	B1	C1	D1	E1	F1	G1	H1	I1
Material	Solar cell	Solar cell	Solar cell	Solar cell	Solar cell	Solar cell	Solar cell	Mirror	White	White
Panel	Main 2	A2	B2	C2	D2	E2	F2	G2	H2	I2
Material	Black	Black	Polyi-mide	Black	Mirror	Black	White	Mirror	Mirror	Polyi-mide

Table 7. Temperature of each node in case of optimization of temperature, power generation and attitude stability

Panel	Main 1	A1	B1	C1	D1	E1	F1	G1	H1	I1
Temp. [K]	370.19	372.64	372.65	346.68	372.08	315.08	213.54	151.32	113.66	79.52
Panel	Main 2	A2	B2	C2	D2	E2	F2	G2	H2	I2
Temp. [K]	294.81	297.78	294.96	289.73	291.28	268.10	213.13	150.95	113.69	79.56