Trans. JSASS Aerospace Tech. Japan
Vol. 14, No. ists31, pp.Pd_1 - Pd_6, 2017

ISTS-2017-d-153／ISSFD-2017-153

Model Error Compensator for Attitude Control of 2-Wheel Spacecraft with On-line FRIT based Tuning

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(Received June 21st, 2017)

The attitude control of the 2-wheel spacecraft using the input-output linearization has been proposed as a failsafe system for the reaction wheel failure. This method is the model based non-linear controller. In actual environment, it is difficult for high precision control via model based controller to be achieved due to a model error. For this problem, this paper proposes applying model error compensator (MEC) using fictitious reference iterative tuning (FRIT). MEC is one of the methods to suppress the influence of the model error. The algorithm to compensate the model error is to modify input by using a difference between output of the controlled object and the one of the nominal model. FRIT is the method to update the parameter of the controller consecutively to become the optimum value. Therefore, the influence of the model error can be suppressed by applying FRIT to MEC even though the model error fluctuates. The validity of this proposed method is verified by numerical simulation.

Key Words: Model error compensator, Two-wheel spacecraft, Attitude control, Failsafe, Model based controller

Nomenclature

$\boldsymbol{\omega}$	:	Angular velocity
Σ	:	Coordinate frame
$\boldsymbol{J}$	:	Inertial matrix
Subscripts		
s	:	spacial
b	:	body
w	:	reaction wheel
n	:	nominal

1. Introduction

A spacecraft is indispensable in the weather forecast and the radio communication in our daily life. When a spacecraft achieves these missions, an attitude control is essential to point an antenna to a target direction, or to point the solar cell paddle to the sun direction. The performance of the weather forecast and the radio communication depends on the one of the attitude control. Thereby, the high accurate attitude control is required. In general, the required pointing accuracy for the spacecraft is approximately 0.05 deg.[1)]

The various methods of the attitude control of the spacecraft exist. Especially, three axises stability methods using reaction wheels are utilized for high accurate attitude control. Three-axis stability method can achieve an arbitrary attitude by attaching three reaction wheels independently. However, the reaction wheel is sometimes broken down by discharge or sudden temperature change in the outer space. But, it is difficult to repair the broken reaction wheel in outer space. In addition, the system becomes underactuated when the number of the available reaction wheels becomes less than three. In this case, the attitude control becomes difficult. When the spacecraft cannot realize the attitude control with enough precision, the mission is suspended. In the worst case, the spacecraft is discarded and the new spacecraft is launched. For example, a reaction wheel is broken down in a mission in Kepler of a planetary probe satellite developed in NASA, and the observation mission is stopped because the high accurate attitude control cannot be achieved.[2)] It is desirable to operate the launched spacecraft as long as possible because the cost of production and the launch of the spacecraft becomes enormous. Therefore, it is necessary to think about the failsafe system so that the spacecraft continues the mission when a reaction wheel is broken down and becomes only two reaction wheels. In the past study, the method of the attitude control using two reaction wheels in consideration of total angular momentum is proposed by Katsuyama and others.[3)] The attitude control is performed by using this method when the number of available reaction wheels become two. Thereby, the spacecraft becomes able to be managed for a long term, and reduction of the launching cost is expected.

In this method, exact model parameters are essential because the non-linear model based controller is used. However, it is difficult to acquire the exact model. For examples, the disturbance torque, motor dynamics or elastic motion of the paddle are difficult to build the exact model, and then the model parameters in the controller might be fluctuate in actual environment. When such a model error exists in the model based control, the control performance deteriorates and cannot achieve the high accurate attitude control. For this problem, we propose the method to modify input so that the actual controlled object follows the movement of the ideal model by applying model error compensator (MEC)[4)5)6)7)8)9)] using fictitious reference iterative tuning (FRIT) which is one of the adjustment method of the controller parameters. The validity of this proposed method is verified by numerical simulation.

2. Controlled object and state equation

2.1. Controlled object

The spacecraft is controlled object in this study. Model parameters is listed in Table 1. Inertial coordinate frame Σ_s, body coordinate frame Σ_b to go along the inertial central organization are established to be shown in Fig. 1. The antenna is attached

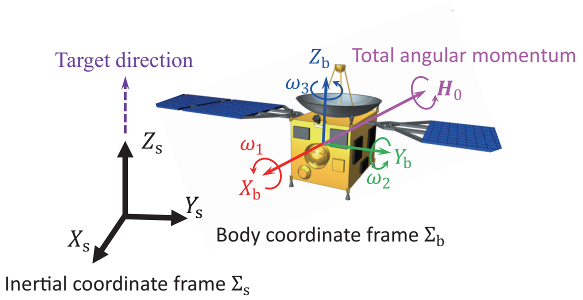

Fig. 1. Model of a spacecraft

to the Z_b axis of body coordinate frame Σ_b and we set inertial coordinate frame Σ_s so that the target direction is the Z_s axis of inertial coordinate frame Σ_s.

In addition, we suppose that the reaction wheel is attached to the X_b, Y_b axis and the reaction wheel of the Z_b axis is broken down. unknown and nonzero angular momentum exists in a whole system. Moreover, the inertial matrix has the model error. Additionally, it is assumed that the following data is available to calculation of input; angular velocities $\boldsymbol{\omega_b}$ and $\boldsymbol{u}$, and the attitude information of the body is $\boldsymbol{q}$. The attitude expression of the body uses quaternion. Quaternion expresses attitude in four parameters using unit vector $\boldsymbol{k} = [x_r,\ y_r,\ z_r]^T$ and angle of rotation θ defined on the inertial coordinate frame Σ_s.

$$\boldsymbol{q} = \begin{bmatrix} q_0 & q_1 & q_2 & q_3 \end{bmatrix}^T = \begin{bmatrix} \cos(\frac{\theta}{2}) & x_r\sin(\frac{\theta}{2}) & y_r\sin(\frac{\theta}{2}) & z_r\sin(\frac{\theta}{2}) \end{bmatrix}^T.$$

In addition, the above equation is restricted in

$$q_0^2 + q_1^2 + q_2^2 + q_3^2 = 1.$$

Table 1. Model parameters

Σ_s	Inertial coordinate frame (X_s, Y_s, Z_s)
Σ_b	Body coordinate frame (X_b, Y_b, Z_b)
$\boldsymbol{J}_b$	Inertial matrix of the body is fixed in Σ_b $\boldsymbol{J}_b$=diag(J_{bx},J_{by},J_{bz})
$\boldsymbol{J}_{wl}$	Inertial matrix of reaction wheel is fixed in Σ_b $(l = 1, 2)$
$\boldsymbol{H}_0$	Total angular momentum is fixed in Σ_s $\boldsymbol{H}_0=[h_{0x}, h_{0y}, h_{0z}]^T$
$\boldsymbol{R}_{bs}$	Rotation matrix (Σ_b w.r.t. Σ_s)
$\boldsymbol{u}$	Angular velocity of reaction wheel $\boldsymbol{u} = [u_1, u_2]^T$
$\boldsymbol{\omega}_b$	Angular velocity of the body $\boldsymbol{\omega_b} = [\omega_{bx}, \omega_{by}, \omega_{bz}]^T$
$\boldsymbol{v}_l$	Direction of reaction wheel is fixed in Σ_b $(l = 1, 2)$ $\boldsymbol{v}_1=[v_{1x}, v_{1y}, v_{1z}]^T$, $\boldsymbol{v}_2=[v_{2x}, v_{2y}, v_{2z}]^T$

2.2. State equation

We derive the state equation of the spacecraft based on law of conservation of angular momentum. We show law of conservation of angular momentum based on the reaction wheel of the X_b,Y_b axis and total angular momentum as follows:

$$\boldsymbol{H}_0 = \boldsymbol{R}_{bs}(\boldsymbol{q})\Big\{\boldsymbol{J}_b\boldsymbol{\omega}_b + \boldsymbol{J}_{w1}(\boldsymbol{\omega}_b + \boldsymbol{v}_1 u_1) + \boldsymbol{J}_{w2}(\boldsymbol{\omega}_b + \boldsymbol{v}_2 u_2)\Big\}. \tag{1}$$

We can express equation (1) as follows:

$$\boldsymbol{\omega}_b = \boldsymbol{J}_t^{-1}(\boldsymbol{R}_{bs}^T(\boldsymbol{q})\boldsymbol{H}_0 - \boldsymbol{J}_{w1}\boldsymbol{v}_1 u_1 - \boldsymbol{J}_{w2}\boldsymbol{v}_2 u_2), \tag{2}$$

where $\boldsymbol{J}_t$ is an inertial parameter of the total:

$$\boldsymbol{J}_t = \boldsymbol{J}_b + \boldsymbol{J}_{w1} + \boldsymbol{J}_{w2}. \tag{3}$$

Relations of time differential of quaternion $\dot{\boldsymbol{q}}$ and the angular velocity of the body $\boldsymbol{\omega_b}$ are as follows:

$$\dot{\boldsymbol{q}} = \frac{1}{2}\begin{bmatrix} -q_1 & -q_2 & -q_3 \\ q_0 & -q_3 & q_2 \\ q_3 & q_0 & -q_1 \\ -q_2 & q_1 & q_0 \end{bmatrix}\boldsymbol{\omega}_b = \boldsymbol{Q}(\boldsymbol{q})\boldsymbol{\omega}_b. \tag{4}$$

The following non-linear state equation is provided by substituting equation (2) for equation (4).

$$\dot{\boldsymbol{q}} = \boldsymbol{f}(\boldsymbol{q}) + \boldsymbol{g}(\boldsymbol{q})\boldsymbol{u}, \tag{5}$$

where state variable $\boldsymbol{q}$ is the attitude information of the body, input $\boldsymbol{u}$ is angular velocity of the reaction wheel. In addition, $\boldsymbol{f}(\boldsymbol{q})$, $\boldsymbol{g}(\boldsymbol{q}) = [\boldsymbol{g}_1, \boldsymbol{g}_2]$ are as follows:

$$\boldsymbol{f}(\boldsymbol{q}) = \boldsymbol{Q}(\boldsymbol{q})\boldsymbol{J}_t^{-1}\boldsymbol{R}_{bs}^T(\boldsymbol{q})\boldsymbol{H}_0, \tag{6}$$

$$\boldsymbol{g}_1(\boldsymbol{q}) = -\boldsymbol{Q}(\boldsymbol{q})\boldsymbol{J}_t^{-1}\boldsymbol{J}_{w1}\boldsymbol{v}_1, \tag{7}$$

$$\boldsymbol{g}_2(\boldsymbol{q}) = -\boldsymbol{Q}(\boldsymbol{q})\boldsymbol{J}_t^{-1}\boldsymbol{J}_{w2}\boldsymbol{v}_2. \tag{8}$$

3. Attitude control with two reaction wheels[3)]

3.1. Control objective

We perform the attitude control of the spacecraft when the reaction wheel of the antenna installation axis is broken down in this paper. The control objective is to point the antenna attached to Z_b axis of the body coordinate frame Σ_b to Z_s axis of the inertial coordinate frame Σ_s which is the target direction. Therefore, we can achieve control target because it becomes $[x_r,\ y_r,\ z_r]^T = [0,\ 0,\ 1]^T$ by zeroing the attitude of the body each q_1, q_2.

3.2. Input-output linearization[10)]

Because the state equation of the spacecraft is non-linear, we make the input-output relations of the controlled object linear. We design the output function $\boldsymbol{h}(\boldsymbol{q})$ as follows:

$$\boldsymbol{y} = \boldsymbol{h}(\boldsymbol{q}) = \begin{bmatrix} q_1 & q_2 \end{bmatrix}^T. \tag{9}$$

We explain a derivation method of the input to perform input-output linearization in the following. First, we continue differentiating the output function until input $\boldsymbol{u}$ appears. Next, we demand input $\boldsymbol{u}$ to remove the non-linear clause of a provided equation.

$$\dot{\boldsymbol{h}}(\boldsymbol{q}) = \frac{\partial \boldsymbol{h}(\boldsymbol{q})}{\partial \boldsymbol{q}}(\boldsymbol{f}(\boldsymbol{q}) + \boldsymbol{g}(\boldsymbol{q})\boldsymbol{u}), \tag{10}$$

$$\boldsymbol{u} = -\boldsymbol{\alpha}(\boldsymbol{q}) + \boldsymbol{\beta}(\boldsymbol{q})\boldsymbol{\mu}, \tag{11}$$

$$\boldsymbol{\alpha}(\boldsymbol{q}) = \left(\frac{\partial \boldsymbol{h}(\boldsymbol{q})}{\partial \boldsymbol{q}}\boldsymbol{g}(\boldsymbol{q})\right)^{-1}\frac{\partial \boldsymbol{h}(\boldsymbol{q})}{\partial \boldsymbol{q}}\boldsymbol{f}(\boldsymbol{q}),$$

$$\boldsymbol{\beta}(\boldsymbol{q}) = \left(\frac{\partial \boldsymbol{h}(\boldsymbol{q})}{\partial \boldsymbol{q}}\boldsymbol{g}(\boldsymbol{q})\right)^{-1}.$$

Therefore, a linear system surrounded in a red frame of Fig. 2 is provided.

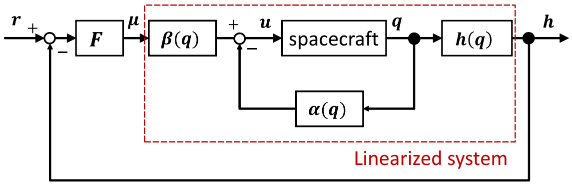

Fig. 2. Input-output linearization

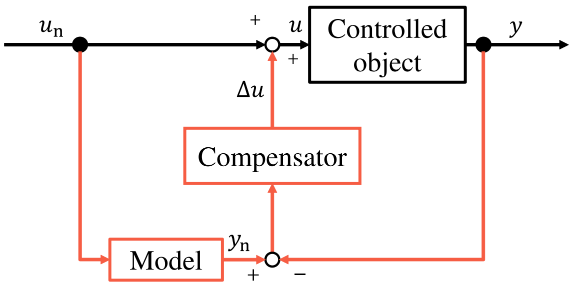

Fig. 3. Structure of MEC

3.3. Output zeroing control

We build the feedback controller to make q_1, q_2 a zero to achieve the control objective. We design virtual input $\boldsymbol{\mu}$ as follows:

$$\boldsymbol{\mu} = \frac{\mathrm{d}\boldsymbol{y}}{\mathrm{d}t} = -\boldsymbol{F}\boldsymbol{y}, \tag{12}$$

where feedback gain $\boldsymbol{F}$ is as follows:

$$\boldsymbol{F} = \mathrm{diag}(F_1, F_2), \quad F_1, F_2 > 0.$$

4. Model error compensator

4.1. Compensator structure

MEC is one of the method to suppress the influence of the model error.[5] MEC modifies input by using a difference between the output of the controlled object and the one of the nominal model in every sampling period. The structure of MEC is depicted in Fig. 3. The compensation gain $\boldsymbol{K}$ is set in Compensator in Fig. 3. The calculations of the compensation input $\Delta\boldsymbol{u}$ using this compensation gain are as follows:

$$\Delta\boldsymbol{u} = \boldsymbol{K}(\boldsymbol{y}_{\mathrm{n}} - \boldsymbol{y}), \tag{13}$$

where $\boldsymbol{y}_{\mathrm{n}}$ is the nominal output which is obtained from the model. The calculations of the actual input $\boldsymbol{u}$ is as follows:

$$\boldsymbol{u} = \boldsymbol{u}_n + \Delta\boldsymbol{u}. \tag{14}$$

As explained above, MEC suppresses the influence of the model error by compensation of the input.

4.2. Design of Compensation Gain

We explain the adjustment method of the compensation gain $\boldsymbol{K}$ in this section.

FRIT is the parameter adjustment method for PID gain. FRIT is usually carried out off-line. However, FRIT can be carried out on-line by using Recursive Least-Squares (RLS).[11)12)] We can ajust the compensation gain on-line by applying this method to MEC. The structure of the method that applied FRIT to MEC in Fig. 4. $\boldsymbol{L}_{\mathrm{a}}$ represents the actual system with input transformation, $\boldsymbol{L}_{\mathrm{n}}$ represents the ideal linear model without the model error, $\boldsymbol{K}$ represents the compensation gain, $\boldsymbol{r}(i)$ represents the reference signal, $\boldsymbol{h}(i)$ represents the actual output, $\boldsymbol{h}_{\mathrm{n}}(i)$ represents

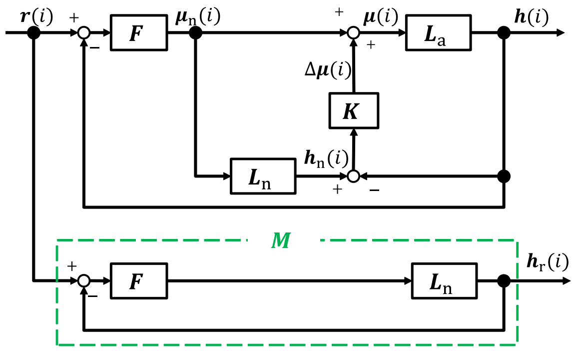

Fig. 4. Structure of method that applied FRIT for MEC

the nominal output and $\boldsymbol{h}_{\mathrm{r}}(i)$ represents the ideal output which is provided when the ideal system without the model error is controlled. In this section, we set discrete steps $i = 1, \ldots, N$ to think about the discrete system. N expresses the current time. In addition, we can express the virtual input $\boldsymbol{\mu}$ as follows:

$$\boldsymbol{\mu} = \boldsymbol{F}(\boldsymbol{r}(i) - \boldsymbol{h}(i)) + \boldsymbol{K}(\boldsymbol{h}_{\mathrm{n}}(i) - \boldsymbol{h}(i)). \tag{15}$$

By rearranging equation (15) , the following equation obtained.

$$\tilde{\boldsymbol{r}}(\boldsymbol{K}, i) = \boldsymbol{F}^{-1}\{\boldsymbol{\mu}(i) - \boldsymbol{K}(\boldsymbol{h}_{\mathrm{n}}(i) - \boldsymbol{h}(i))\} + \boldsymbol{h}(i). \tag{16}$$

In FRIT, it is important that we express this fictitious reference signal $\tilde{\boldsymbol{r}}$ for a function of compensation gain $\boldsymbol{K}$. Based on this fictitious reference signal, compensation gain $\boldsymbol{K}$ is adjusted so that following criterion function $J(\boldsymbol{K})$ is minimized.

$$J(\boldsymbol{K}) = \sum_{i=1}^{N} (\boldsymbol{h}(i) - \boldsymbol{h}_{\mathrm{r}}(\boldsymbol{K}, i))^2, \tag{17}$$

$$\boldsymbol{h}_{\mathrm{r}}(\boldsymbol{K}, i) = \boldsymbol{M}\,\tilde{\boldsymbol{r}}(\boldsymbol{K}, i), \tag{18}$$

where $\boldsymbol{M}$ is the ideal closed loop model that does not have the model error. In addition, this criterion function is designed so that when the influence of the model error becomes small, the evaluation value level becomes small. However, this optimization calculation is not generally solved effectively. To calculate this optimization problem, we think about an ideal case $J = 0$.

$$\boldsymbol{h}(i) - \boldsymbol{M}\,\tilde{\boldsymbol{r}}(\boldsymbol{K}, i) = 0. \tag{19}$$

By substituting equation (16) in equation (19), and multiplying feedback gain $\boldsymbol{F}$ from the left, the equation becomes as follows:

$$\begin{aligned} &\boldsymbol{K}\boldsymbol{M}(\boldsymbol{h}_{\mathrm{n}}(i) - \boldsymbol{h}(i)) \\ &\quad - \{\boldsymbol{M}\boldsymbol{\mu}(i) - \boldsymbol{F}(\boldsymbol{h}(i) - \boldsymbol{M}\boldsymbol{h}(i))\} = 0. \end{aligned} \tag{20}$$

Hereby, we replace problem to minimize equation (17) with a problem to minimize the following criterion function.

$$\widehat{J}(\boldsymbol{K}) = \sum_{i=1}^{N} \boldsymbol{\varepsilon}^2(\boldsymbol{K}, i), \tag{21}$$

$$\begin{aligned} \boldsymbol{\varepsilon}(\boldsymbol{K}, i) = &\boldsymbol{K}\boldsymbol{M}(\boldsymbol{h}_{\mathrm{n}}(i) - \boldsymbol{h}(i)) \\ &- \{\boldsymbol{M}\boldsymbol{\mu}(i) - \boldsymbol{F}(\boldsymbol{h}(i) - \boldsymbol{M}\boldsymbol{h}(i))\}. \end{aligned} \tag{22}$$

The virtual error $\boldsymbol{\varepsilon}(\boldsymbol{K}, i)$ is rewritten as follows based on equation (22):

$$\boldsymbol{\varepsilon}(\boldsymbol{K}, i) = \boldsymbol{K}\boldsymbol{\xi}(i) - \boldsymbol{\eta}(i), \tag{23}$$

$$\boldsymbol{\xi}(i) = \boldsymbol{M}(\boldsymbol{h}_{\mathrm{n}}(i) - \boldsymbol{h}(i)), \tag{24}$$

$$\boldsymbol{\eta}(i) = \boldsymbol{M}\boldsymbol{\mu}(i) + \boldsymbol{F}(\boldsymbol{h}(i) - \boldsymbol{M}\boldsymbol{h}(i)). \tag{25}$$

In this case, the minimization problem of $\widehat{J}(\boldsymbol{K})$ is regarded as a least-squares problem because $\boldsymbol{\varepsilon}(\boldsymbol{K}, i)$ is linear with respect to $\boldsymbol{K}$. Thereby, we apply RLS for equation (21). The algorithm of RLS demands $\widehat{\boldsymbol{K}}(i)$ recursively using optimal estimate $\widehat{\boldsymbol{K}}(i-1)$ of the compensation gain at the previous time $i-1$.

Standard RLS uses all data from the initial time to the present time. When the characteristic fluctuates the controlled object by changing the model error, standard RLS cannot cope a change appropriately by considering the data before the change. We use RLS with the forgetting factor $\lambda(0 < \lambda < 1)$ to cope with this.[11] When we use the forgetting factor, the criterion function is as follows:

$$\widehat{J}(\boldsymbol{K}) = \sum_{i=1}^{N} \lambda^{N-i} \boldsymbol{\varepsilon}^2(\boldsymbol{K}, i). \tag{26}$$

Therefore, small weight is multiplied the old error data. In addition, standard RLS algorithm can be expressed by setting $\lambda = 1$.

The calculation of the RLS algorithm with the forgetting factor is settled as follows:

$$\boldsymbol{P}(i) = \frac{1}{\lambda}\left\{\boldsymbol{P}(i-1) - \frac{\boldsymbol{P}(i-1)\boldsymbol{\xi}(i)\boldsymbol{\xi}(i)^T\boldsymbol{P}(i-1)}{\lambda + \boldsymbol{\xi}(i)^T\boldsymbol{P}(i-1)\boldsymbol{\xi}(i)}\right\}, \tag{27}$$

$$\widehat{\boldsymbol{K}}(i) = \widehat{\boldsymbol{K}}(i-1) + \frac{\boldsymbol{P}(i-1)\boldsymbol{\xi}(i)(\boldsymbol{\eta}(i) - \widehat{\boldsymbol{K}}(i-1)\boldsymbol{\xi}(i))}{\lambda + \boldsymbol{\xi}(i)^T\boldsymbol{P}(i-1)\boldsymbol{\xi}(i)}, \tag{28}$$

where $\boldsymbol{P}$ is the correlation matrix. To initialize the RLS algorithm, we need to specify the initial compensation gain $\widehat{\boldsymbol{K}}(0)$ and the initial correlation matrix $\boldsymbol{P}(0)$. Usually we set the matrix

$$\boldsymbol{P}(0) = \gamma \boldsymbol{I}, \tag{29}$$

where $\boldsymbol{I}$ is identity matrix, and $\gamma > 0$ is set for the fixed number of the big value when signal-to-noise ratio is high. Based on RLS algorithm stated above, the compensation gain $\widehat{\boldsymbol{K}}(i)$ is updated at each time. This variation of the compensation gain may be large at the beginning of the algorithm. Due to this, the control performance is deteriorated. Thus, to reduce the variation of $\widehat{\boldsymbol{K}}(i)$ may be of the compensation gain, we use the following update rule of the implemented compensation gain $\boldsymbol{K}(i)$:[11]

$$\boldsymbol{K}(i) = (1-\kappa)\boldsymbol{K}(i-1) + \kappa\widehat{\boldsymbol{K}}(i), \tag{30}$$

where κ is an enough small integer. Equation (30) is discrete low-pass filter. Then, the cut-off frequency becomes big when the coefficient of the low-path filter κ closes to 1, and the cut-off frequency becomes small when the coefficient of the low-path filter κ closes to 0. $\boldsymbol{K}$ in this way changes gently.

In addition, the destabilization of the compensation gain is suppressed by breaking off update of the compensation gain $\boldsymbol{K}$ when $\boldsymbol{\xi}$ close to 0. When output error $\boldsymbol{h}_\mathrm{n} - \boldsymbol{h}$ becomes smaller than threshold ψ, this algorithm breaks off update. Thereby, this method suppresses destabilization of the compensation gain $\boldsymbol{K}$ caused by bringing $\boldsymbol{\xi}$ close to 0.

5. Simulation

5.1. Simulation Conditions

We perform simulation in the case of the nominal inertial matrix of the totals $\boldsymbol{J}_\mathrm{tn}$ to use for decision of the input is different from the actual inertial matrix of the totals $\boldsymbol{J}_\mathrm{t}$, and the unknown total angular momentum exists. The relation of the nominal inertial matrix of the totals $\boldsymbol{J}_\mathrm{tn}$ and the actual inertial matrix of the totals $\boldsymbol{J}_\mathrm{t}$ are as follows:

$$\boldsymbol{J}_\mathrm{t} = \boldsymbol{J}_\mathrm{tn} + \begin{bmatrix} \Delta J_\mathrm{xx} & 0.0 & 0.0 \\ 0.0 & \Delta J_\mathrm{yy} & 0.0 \\ 0.0 & 0.0 & \Delta J_\mathrm{zz} \end{bmatrix}. \tag{31}$$

We assumed the required pointing accuracy of the antenna within 0.05 deg this simulation. We suppose that the spacecraft moved over a geostationary orbit. We assume that angular momentum change to low frequency by the disturbance torque such as gravity-gradient torque and the solar radiation pressure. In addition, we assume that the error of the inertial matrix of the body is about 0.5% of the actual inertial matrix of the totals $\boldsymbol{J}_\mathrm{t}$ based on precision of the measuring equipment.[14] Each parameter of this simulation is listed in Table 2. In a similar way, each parameter of the control system is listed in Table 3. Sampling period is 100 ms.

5.2. Result

The simulation result of the output h_1, h_2 and the point error of the antenna are depicted in Fig. 5, 6, 7. The simulation result of the compensation gain of the proposed method and input u_1, u_2 are depicted in Fig. 8, 9, 10. Then, blue line expresses the ideal case of without the model error, green line expresses the case of without applying MEC when the model error exists, red line expresses the case of applying the proposed method when the model error exists.

5.3. Discussion

From Fig. 5, 6, both h_1 and h_2 becomes vibrational and cannot become a zero when w/o MEC. Whereas, we confirm that the actual output close to the ideal output by suppressing the influence of the model error when we apply the proposed method. From Fig. 7, we cannot achieve control objective when w/o MEC. However, the proposed method can point the antenna to

Table 2. Simulation parameters

$$[x_\mathrm{r0},\ y_\mathrm{r0},\ z_\mathrm{r0}] = [1,\ 2,\ 3]$$
$$\theta_0 = 30\,\mathrm{deg}$$
$$\boldsymbol{J}_\mathrm{bn} = \mathrm{diag}(4000,\ 5000,\ 6000)\,\mathrm{kgm}^2$$
$$\boldsymbol{J}_\mathrm{w1} = \mathrm{diag}(0.45,\ 0.225,\ 0.225)\,\mathrm{kgm}^2$$
$$\boldsymbol{J}_\mathrm{w2} = \mathrm{diag}(0.225,\ 0.45,\ 0.225)\,\mathrm{kgm}^2$$
$$h_\mathrm{0x}(t) = 0.05\sin(2\pi \times 10^{-4}t)\,\mathrm{Nms}$$
$$h_\mathrm{0y}(t) = 0.05\cos(2\pi \times 10^{-4}t)\,\mathrm{Nms}$$
$$h_\mathrm{0z}(t) = 0.05\sin(2\pi \times 10^{-4}t)\,\mathrm{Nms}$$
$$\boldsymbol{H}_\mathrm{0n}(0) = [0.0,\ 0.0,\ 0.0]^\mathrm{T}\,\mathrm{Nms}$$
$$\Delta J_\mathrm{xx} = 20.003\,\mathrm{kgm}^2,\ \Delta J_\mathrm{yy} = 25.003\,\mathrm{kgm}^2,\ \Delta J_\mathrm{zz} = 30.002\,\mathrm{kgm}^2$$
$$\boldsymbol{v}_1 = [1.0,\ 0.0,\ 0.0]^\mathrm{T},\ \boldsymbol{v}_2 = [0.0,\ 1.0,\ 0.0]^\mathrm{T}$$

Table 3. Parameter of control system

Feedback gain F_1	0.0009
Feedback gain F_2	0.0009
Oblivion coefficient λ	0.995
Initial correlation matrix $P_1(0)$	diag(100.0, 100.0)
Initial correlation matrix $P_2(0)$	diag(100.0, 100.0)
Coefficient of the low-path filter κ	0.45
Initial compensation gain $K_1(0)$	[0.0 0.0]
Initial compensation gain $K_2(0)$	[0.0 0.0]
Threshold ψ_1	0.00012
Threshold ψ_2	0.00012

the target direction by suppressing the influence of the model error. The reason why the proposed method is able to suppress the influence of the model error is that the compensation gain is updated consecutively in Fig. 8. As a result, input is modified in Fig. 9, 10. As explained above, we can point the antenna to the target direction because the influence of the model error is controlled by applying the proposed method for the attitude control of the 2-wheel spacecraft when a model error exists.

6. Conclusion

This paper proposed applying a model error compensator based on FRIT to underactuated 2-wheel spacecraft in order to suppress the influence of the model error. In addition, this paper verified the effectiveness of the proposed method in simulation. As a result, proposed method was able to point the antenna to the target direction when the model error exists by suppressing the influence of the model error. This is because, the input was modified by MEC structure. Moreover, the compensation gain was automatically tuned by FRIT consecutively.

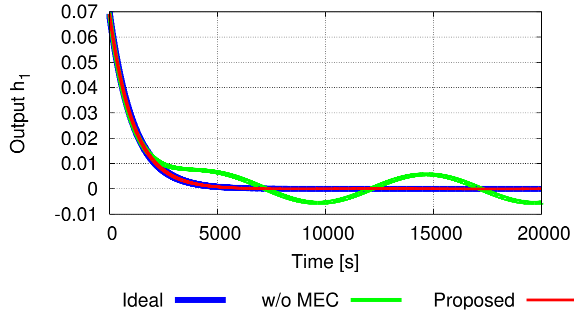

Fig. 5. Output h_1

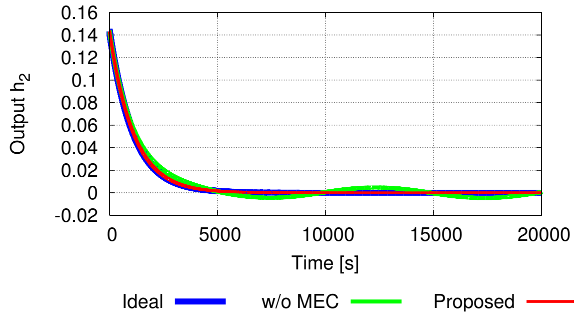

Fig. 6. Output h_2

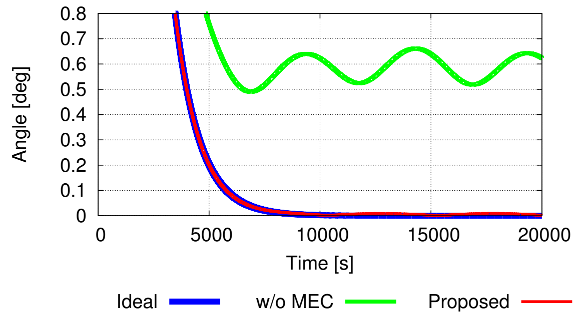

Fig. 7. Pointing error of antenna

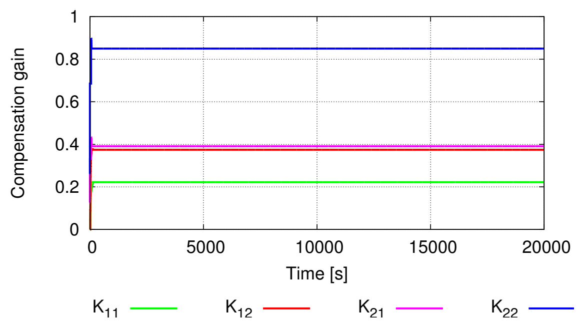

Fig. 8. Compensation Gain K

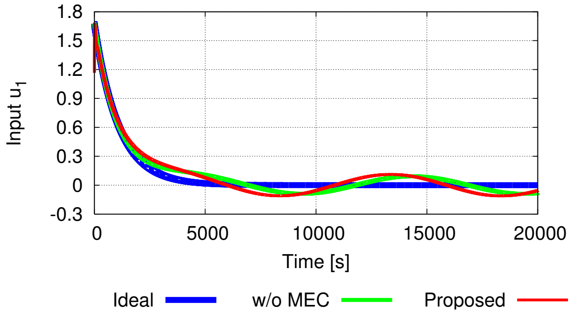

Fig. 9. Input u_1

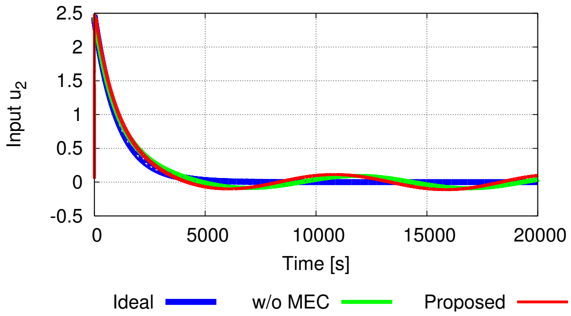

Fig. 10. Input u_2

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