Energy Optimal Path Planning of Satellite Attitude Maneuver Using Control Moment Gyros

Yohichiro KUSUDA and Masaki TAKAHASHI
Keio University, 3-14-1 Hiyoshi, Kohoku-ku, Yokohama-shi, Kanagawa, 223-8522, Japan
E-mail: kusuda@yoshida.sd.keio.ac.jp

ABSTRACT
A Control Moment Gyros (CMG) is expected to be applied to an attitude control actuator of an agile satellite that requires rapid rotational maneuverability because it can generate high torque effectively. However, it is pointed out that CMG itself has a singularity problem, which interferes with not only rapid maneuverability but also energy consumption and safe use of CMG. This paper proposes a feedforward control logic using the optimization about CMG control inputs. In this study, Fourier Basis Algorithm (FBA) is used as a method of path planning. It approximates the CMG control input by Fourier series. In order to verify the usefulness of the proposed logic, the numerical simulations about a large-angle rapid maneuver mission is carried out. From the simulation result, it was confirmed that the proposed logic can control CMG in safety under the control requirements without rapid and furious CMG actuation to avoid singularities. Additionally, the energy consumption of CMG can be reduced by approximately 46% by comparing with the feedback control using GSRInverse steering logic.

コントロールモーメントジャイロによる人工衛星姿勢変更のエネルギー最適経路計画

概要
コントロールモーメントジャイロ（CMG）は効率的に高トルクを出力できるアクチュエータであるため、近年高機動性を有した人工衛星の姿勢制御アクチュエータとしての利用が期待されている。しかし、CMGを姿勢変更のためのアクチュエータとして用いる場合には、特別な問題が存在し、これが高速な姿勢変更を妨げるため、様々な特異点回避手法が提案されている。しかし、それらの手法ではCMGの消費エネルギーや堅牢性といった面が考慮されていない。そこで、本研究では、ステアリング則を用いずに、CMG駆動経路計画を用いたフィードフォワード制御則を提案する。本手法では、フーリエ級数近似を用いたフーリエ基底係数最適化法（FBA）により、CMGの消費エネルギーについて最適な経路計画を行うことで、制御要求を満たす中で効率的で安全な駆動を実現する。本研究では、大角度マニューバミッションについての数値解析を行い、ステアリング則を用いた従来手法と比較することでCMGの消費エネルギーや堅牢性の面について有効性を示した。

1. INTRODUCTION
The demand for an earth observation satellite equipped with a high-performance observing sensor has increased in order to acquire high-resolution images about specific ground targets. For precision observation the whole satellite body will turn rapidly rather than sweep the imaging system from side to side because pointing the entire satellite allows the imaging system to achieve higher definition and improve the resolution for its images. Moreover the cost and effectiveness of such agile satellite are greatly affected by the average maneuvering time and observing frequency. Therefore the development of large-angle rapid rotational maneuverability is required for the attitude control system of such agile satellite.

From the requirement, Control Moment Gyros (CMG) is expected to be applied to an attitude control actuator of the agile satellite, instead of Reaction Wheel (RW), because it can generate higher torque than RW effectively. In the past CMG had been employed as an attitude stabilization actuator of a large spacecraft such as International Space Station (ISS). And large-angle rapid maneuverability had not been needed for such large spacecraft. Then the variations of CMG gimbal angles are small and CMG can be controlled by a linear simple steering logic.

However, when CMG is used as the attitude control actuator of the agile satellite, CMG control logic can not be linearized because the variations of CMG gimbal angles are large. Therefore the CMG control logic for large-angle rapid maneuver is not linear time-invariant but nonlinear time-variant. In general the system is controlled by logics separated into a satellite attitude control system and CMG control system which is called CMG steering logic. CMG steering logic determines CMG gimbal rate commands in response to torque commands which come from the satellite attitude control system. Such CMG-based attitude control system of the agile satellite is shown in Fig. 1.

Fig. 1. Block diagram of existing control logic
But in case that the agile satellite uses four single gimbal CMGs of a pyramid arrangement for three axis attitude control, there is a problem that the dimension of the output torque degenerates from three-dimension to below two-dimension. This is because the direction of output torque depends on CMG gimbal angles. The problem is called singularity problem. At the singularity state, CMG can not generate a control torque to a particular direction. Then CMG system can not fulfill the requirement of rapid maneuverability. Therefore some CMG singularity avoidance steering logics are proposed such as Generalized Singularity Robust Inverse (GSRInverse) steering logic, Singularity Direction Avoidance (SDA) steering logic and so on in Refs. 1 to 4.

This research focuses on two following problems of existing CMG singularity avoidance steering logic. First problem is that it is impossible to use CMG in safety due to severe activation to avoid singularity state. This problem has a risk of a failure accident of CMG and relates directly to subsistence of the entire satellite system. Second problem is that little attention has been given to the optimality about energy consumption of CMG. As long as the attitude control system consists of two control systems including CMG steering logic, the optimality about the whole system can not be evaluated.

From these points, this paper proposes the control logic of satellite attitude maneuver using CMG without CMG steering logic. This logic is a feedforward control logic using CMG optimal paths which has been planned in advance by the optimization about CMG energy consumption under the control requirement. This logic can control CMG effectively in points of energy consumption and safe use of CMG because path planning can consider about the optimality of energy consumption and there is no severe activation to avoid singularity state from no use of CMG steering logic. The method of the CMG path planning is Fourier Basis Algorithm (FBA) which seeks optimal solutions by Fourier series approximation of control inputs. This method can transform the optimization problem about control inputs to about Fourier series. Therefore the parameters of the optimization decrease and the amount of information handled at satellite on-board computers is reduced more than discrete-time optimization methods. Additionally the derivation of the control inputs in FBA is continuous function and this is effective in actual implementation because sin and cos functions are smooth and have no severe variations.

This paper will show the availability of the proposed logic by comparing with an existing control logic including CMG steering logic in points of energy consumption and safe use of CMG by numerical simulations.

2. ATTITUDE CONTROL USING CMG

2.1. Modeling

We are concerned with the attitude maneuver control of the agile satellite using CMG system which is four single-gimbal CMGs of a pyramid arrangement in the Fig. 2.

In this section, fundamental principles of the rigid satellite rotational equations and CMG dynamics are described, and the objective of this section is to organize the mathematical model of satellite attitude maneuver using CMG for the path planning in the next section.

This problem assumes a large-angle rapid maneuver. Therefore the expression of the satellite attitude is the following quaternion kinematics equation:

\[
\mathbf{q} = \frac{1}{2} \begin{bmatrix} 0 & \omega_3 & -\omega_2 & \omega_1 \\ -\omega_3 & 0 & \omega_1 & \omega_2 \\ \omega_2 & -\omega_1 & 0 & \omega_3 \\ -\omega_1 & -\omega_2 & -\omega_3 & 0 \end{bmatrix} \mathbf{\bar{\omega}} \cdot \mathbf{q} \tag{1}
\]

where \( \mathbf{q} \) is the quaternion vector, \( \mathbf{\omega} = (\omega_1, \omega_2, \omega_3) \) is the satellite angular velocity vector and \( \mathbf{\bar{\omega}} \) is the new defined vector.

From the principle of angular momentum conservation the equation of the rotational motion of the rigid satellite equipped with CMG is given by the following equation:

\[
\frac{d\mathbf{H}}{dt} + \mathbf{\omega} \times \mathbf{H} = 0 \tag{2}
\]

where \( \mathbf{H}_s \) is the total satellite angular momentum vector including CMG expressed in the satellite body-fixed control axes. Then the model is made on the assumption that the satellite body is rigid and external torques, such as gravity gradient and solar pressure and aerodynamic torques, can be regarded as zeros during a short time maneuver.

The total angular momentum vector consists of the satellite main body angular momentum and the CMG angular momentum; that is, we have

\[
\mathbf{H}_s = \mathbf{I} \mathbf{\omega} + \mathbf{h} \tag{3}
\]

where \( \mathbf{I} \) is the satellite inertia matrix and \( \mathbf{h} \) is the CMG angular momentum vector expressed in the satellite body-fixed control axes.

When Eq. (3) is substituted to Eq. (2), we obtain

\[
\mathbf{I} \mathbf{\omega} + \mathbf{\omega} \times \mathbf{I} \mathbf{\omega} + \dot{\mathbf{h}} + \mathbf{\omega} \times \mathbf{h} = 0 \tag{4}
\]

where Eq. (4) is the main rotation dynamic equation and \( \dot{\mathbf{h}} \) is the time-derivation vector of the CMG angular momentum. Then the satellite attitude can be controlled by changing the direction of CMG angular momentum.

The angular momentum of skew type 4CMGs expressed in Fig. 2 is in general a function of the gimbal angles \( \delta = (\delta_1, \delta_2, \delta_3, \delta_4) \); that is, we have

\[
\mathbf{h} = h_{CMG} \begin{bmatrix} -c\beta \sin \delta_1 - \cos \delta_1 - c\beta \sin \delta_1 + \cos \delta_4 \\ \cos \delta_1 - c\beta \sin \delta_1 - \cos \delta_1 + c\beta \sin \delta_4 \\ s\beta \sin \delta_1 + s\beta \sin \delta_1 + s\beta \sin \delta_1 + s\beta \sin \delta_1 \end{bmatrix}
\]
\begin{equation}
\mathbf{h} = h_{CMG} \begin{bmatrix}
-c\beta \cos \delta_1 & \sin \delta_1 & c\beta \cos \delta_2 & -\sin \delta_2 \\
-\sin \delta_1 & -c\beta \cos \delta_2 & \sin \delta_2 & c\beta \cos \delta_1 \\
s\beta \cos \delta_1 & s\beta \cos \delta_2 & s\beta \cos \delta_3 & s\beta \cos \delta_4
\end{bmatrix} \mathbf{\delta} = \mathbf{A}(\mathbf{\delta}) \mathbf{\dot{\delta}} \tag{5}
\end{equation}

where \( h_{CMG} \) is the CMG wheel angular momentum, \( \beta \) is the skew angle of 4CMGs, \( \beta = \sin \beta \), \( \psi = \cos \beta \) and \( \mathbf{D} \) is the new defined vector of the CMG angular momentum.

The time derivative of the CMG angular momentum vector can be obtained as
\begin{equation}
\dot{\mathbf{h}} = h_{CMG} \begin{bmatrix}
-c\beta \cos \delta_1 & \sin \delta_1 & c\beta \cos \delta_2 & -\sin \delta_2 \\
-\sin \delta_1 & -c\beta \cos \delta_2 & \sin \delta_2 & c\beta \cos \delta_1 \\
s\beta \cos \delta_1 & s\beta \cos \delta_2 & s\beta \cos \delta_3 & s\beta \cos \delta_4
\end{bmatrix} \dot{\mathbf{\delta}} = \mathbf{A}(\mathbf{\delta}) \mathbf{\ddot{\delta}} \tag{6}
\end{equation}

where \( \dot{\mathbf{\delta}} = (\dot{\delta}_1, \dot{\delta}_2, \dot{\delta}_3, \dot{\delta}_4) \) is the gimbal rate vector and \( \mathbf{A} \) is the \( 3 \times 4 \) Jacobian matrix. One approach of designing CMG steering logic employs this differential relation among the gimbal angles, the gimbal angular rates and the derivation of the CMG angular momentum. This topic will be described in the next session.

Consequently the mathematical model of the satellite attitude maneuver using CMG is described by
\begin{equation}
\mathbf{x} = \mathbf{g}(\mathbf{x}) + \mathbf{h}(\mathbf{x}) \mathbf{u} \tag{7}
\end{equation}

where \( \mathbf{x} \) is the state variable vector which consists of the quaternion vector \( \mathbf{q} \), the satellite angular velocity vector \( \mathbf{\omega} \) and the CMG gimbal angle vector \( \mathbf{\delta} \). The control input vector \( \mathbf{u} \) is the CMG gimbal rate vector \( \mathbf{\delta} \) and \( \mathbf{I}_o \) is a unit matrix.

\subsection{2.2. Existing Control Logic}

In this section the existing control logic is introduced and the defects of the existing logic are indicated.

The existing control logic of the agile satellite using CMG is a feedback control logic which consists of the satellite attitude control system and the CMG steering logic in Fig. 1. The attitude control system calculates the control input torque from the satellite state variables and reference attitude commands. Then in order to realize the torque command CMG steering logic determines the CMG control input commands which are CMG gimbal rates.

The CMG control logic determines the CMG input command vector \( \mathbf{\delta} \) from the satellite attitude torque command vector \( \mathbf{u} \). The relation between the output torque of CMG and the gimbal angular rate vector is described by the kinematic equation in Eq. (6). In order to solve the inverse kinematics of Eq. (6), typically CMG steering logic uses the following pseudoinverse matrices:
\begin{equation}
\mathbf{\dot{\delta}} = \mathbf{A}' \mathbf{u} \tag{9}
\end{equation}
\begin{equation}
\mathbf{A}' = \mathbf{A}' (\mathbf{A} \mathbf{A}' \mathbf{A}^+)^{-1} \tag{10}
\end{equation}

where \( \mathbf{A}' \) is the minimum two-norm solution of the CMG gimbal angular rate under the constraint of Eq. (6). This is the most effective actuation of CMG in the all methods among CMG steering logics. But if \( \text{rank}(\mathbf{A}) < 3 \), then \( \text{det}(\mathbf{A} \mathbf{A}') = 0 \) and pseudoinverse matrix \( \mathbf{A}^+ \) diverges. The state is singularity. In the singularity state, the end time of maneuver delays because CMG can not generate the demanded control torque. Therefore some singularity avoidance methods were proposed so as to avoid the singularity state and to satisfy the control requirement of short-term large-angle rapid maneuver. Generalized Singularity Robust Inverse (GSRInverse) method in Ref. 1.

is described as follows:
\begin{equation}
\mathbf{\dot{\delta}} = \mathbf{A}' (\mathbf{A} \mathbf{A}' + \lambda \mathbf{F})^{-1} \mathbf{u} \tag{11}
\end{equation}
\begin{equation}
\mathbf{F} = \begin{bmatrix} 1 & \epsilon_3 & \epsilon_2 \\ \epsilon_3 & 1 & \epsilon_1 \\ \epsilon_2 & \epsilon_1 & 1 \end{bmatrix} \tag{12}
\end{equation}

where the scalar \( \lambda \) and the off-diagonal element \( \epsilon_i \) are to be properly selected so as not to go into \( \text{det}(\mathbf{A} \mathbf{A}') = 0 \).

This method is based on the mixed, two-norm and weighted least-squares minimization about the torque errors and the gimbal angular rates. This method can escape any internal singularity through the influence of the additional term \( \mathbf{F} \).

But these actuations to avoid singularity are rapid and furious. Therefore it is emphasized that these rapid and furious actuations of CMG are thought to cause the failure accident and the energy loss of CMG. In other words, existing feedback logics using CMG steering logics are undesirable control logic in the points of energy consumption and safety use of CMG.

In the next section, the feedforward control logic using FBA will be proposed in order to solve these problems of the existing methods.

\section{3. PROPOSED LOGIC}

\subsection{3.1. Outline}

The purpose of the proposed logic is to control the system effectively in points of energy consumption and safe use of CMG without separating the whole control system into the attitude control system and the CMG steering system. Therefore the proposed logic aims to control the entire system directly without using CMG steering logic. The block diagram of the proposed logic is shown in Fig. 3.

In particular this paper proposes the logic using path planning of CMG gimbal angular rates that are control inputs. In the real-time implementation, this path planning can be executed before maneuver because earth observing satellites are thought to have the unperformed time at the other side of the ground objective points around the world.

By using the path as the feedforward inputs of the whole system, the problem about energy consumption of CMG can be settled through the consideration of CMG energy consumption in the path planning.

The problem about rapid and furious actuation of CMG also can be settled through no use of CMG steering logic.

\begin{figure}[h]
\centering
\includegraphics[width=0.8\textwidth]{fig3.png}
\caption{Block diagram of proposed logic}
\end{figure}
because the dangerous actuation is caused by CMG steering logic to avoid the singularity.

A feature of this control problem is that reference attitude and maneuver time are preliminarily given by the missions. Therefore the purpose of the path planning is not optimization about minimum-time maneuver but about energy consumption of CMG under the control requirement. In this study, the method of path planning is FBA, which is based on the approximation of the control inputs by Fourier series. This method has an advantage about on-board calculation resource and calculation time of path planning because optimal parameters in FBA are less than in other discrete-time optimization methods. Additionally in actual implementation control inputs approximated by the continuous function can make CMG to the smooth actuation. Therefore FBA is applied to the solution method of this path planning. In the following section, the detail explanation about FBA is described.

3.2. Fourier Basis Algorithm (FBA)

FBA is the method searching optimal Fourier coefficient parameters by approximating control inputs by Fourier series and optimization theory is based on Newton method in Refs. 5 and 6.

In general consider the motion equation given by Eq. (7) as the constrained condition, boundary conditions and the evaluation function given as follows:

\[
x(0) = x_0, \quad x(t_f) = x_f,
\]

\[
J(u) = \int_0^{t_f} \left( \frac{1}{2} \dot{x}(t)^T \dot{x}(t) + (x(t) - x_f)^T M(x(t) - x_f) \right) dt
\]  

(13)

where \( t_f \) is the end time of maneuver and Eq. (13) is boundary conditions of state variables and Eq. (14) is the evaluation function which consists of the control input norm \( u(t)^T u(t) \) and the state variables at the end of maneuver \( x(t_f) \). \( x_i \) is the initial state variable vector and \( x_f \) is the terminal state variable vector. \( M \) is the weighting matrix to be selected to satisfy the control requirement.

The control input vector \( u(t) \) minimizing the evaluation function \( J \) is the desirable control input vector which can control the system under the control requirement and the norm of control inputs becomes minimum. This solution is called optimal path in the following section.

FBA approximates this control inputs by Fourier series in order to search the optimal path to minimize the evaluation function given as follows:

\[
u = E \alpha
\]  

(15)

\[
E = \left[ \frac{1}{2} \sin \omega_0 t \sin 2 \omega_0 t \ldots \sin 2 \omega_j t \cos \omega_0 t \cos 2 \omega_0 t \ldots \right]
\]

(16)

where \( E \) is the Fourier base term matrix, \( \alpha \) is the Fourier coefficient parameter vector, \( \omega_b \) is the basic frequency of Fourier series approximation. When Fourier series approximation is set up N-order term and Eq. (15) is substituted into Eq. (14), by the orthogonality of the trigonometric function, the evaluation function Eq. (14) comes down to the function about only Fourier parameters in the following form:

\[
J(\alpha) = \alpha^T \alpha + (x(t_f) - x_f)^T M(x(t_f) - x_f)
\]  

(17)

In order to obtain the optimal parameters minimizing the evaluation function, Newton method is used as the searching algorithm. Newton method is based on the first and second-order derivatives of the evaluation function and has a feature that the convergence constant of searching the optimal solution is high.

The evaluation function Eq. (17) can be Taylor expanded around the present value of optimal parameter \( \alpha^* \) as follow:

\[
J(\alpha^* + \Delta \alpha) = J(\alpha^*) + \frac{\partial J(\alpha^*)}{\partial \alpha} \Delta \alpha + \frac{1}{2} \left( \Delta \alpha^T \frac{\partial^2 J(\alpha^*)}{\partial \alpha^2} \Delta \alpha + O(\Delta \alpha^2) \right)
\]  

(18)

where \( \Delta \alpha \) is the deviation vector of Fourier parameters. Then \( \Delta \alpha \) is determined by the basis of Newton method to satisfy \( J(\alpha^* + \Delta \alpha) - J(\alpha^*) < 0 \) as follow:

\[
\Delta \alpha = -\mu \left( I + y(t_f)^T M y(t_f) \right)^{-1} \left( \alpha^* + y(t_f)^T M (x(t_f) - x_f) \right)
\]  

(19)

where \( \mu \) is the constant to be properly selected and \( y(t_f) \) is obtained as

\[
y = \frac{\partial x}{\partial \alpha} \quad y(t_f) = \frac{\partial \dot{x}}{\partial \alpha} \sum_{i=0}^N \left( \frac{\partial \chi_i}{\partial \alpha_i} \right) y + hE \]  

(20)

by the numerical calculation of Eq. (20) and (21).

Eventually the updating equation of the Fourier parameters is the following form:

\[
\alpha = \alpha^* + \Delta \alpha
\]  

(22)

In this way, FBA can obtain the optimal paths by uploading according to Eq. (22).

3.3. Application

In this section two evaluations are added to FBA because aforementioned FBA is a general form.

First evaluation is energy consumption of CMG. Though the basic evaluation function of FBA estimates energy consumption at the norm of control inputs \( u(t)^T u(t) \), energy consumption of CMG can not be estimated exactly because satellite angular velocities can not be neglected with respect to CMG gimbal angular rates in agile maneuver. Therefore in order to evaluate the energy consumption of CMG strictly, the first term of Eq. (17) is modified in the follow forms:

\[
J(\alpha) = \sum_{i=0}^N \left( T_i^T M_i T_i dt \right) + (x(t_f) - x_f)^T M (x(t_f) - x_f)
\]  

(23)

\[
T_i = (\omega_{\text{CMG}} + \omega) \times h
\]  

(24)

where \( T_i \) is the output torque vector of \( i \)th CMG, \( h \) is the time-derivation vector of CMG angular rates, \( \omega_{\text{CMG}} \) is the angular velocity vector of \( i \)th CMG.

Second evaluation is restraint conditions of the control
inputs. A basic mission of earth observing satellites is Rest-to-Rest maneuver. Therefore this paper assumes the Rest-to-Rest maneuver mission. Then CMG should rest at the beginning and the end of the maneuver, in other words CMG gimbal angular rates should be zeros. Therefore FBA is adjusted to the form including this restraint condition in the following forms:

\[
\dot{\delta}(0) = 0 \\
\dot{\delta}(t_f) = 0
\]  

where Eq. (25) is the beginning restraint condition and Eq. (26) is the end restraint condition.

The penalty method can treat these restraint conditions as the terms of the evaluation function as follow:

\[
J = \sum_{i=1}^{4} \left[ \int_0^T T_i^r M_i T_i \, dt \right] + (x(t_f) - x_d)^\top M (x(t_f) - x_d)
\]

\[ + u^\top(0) S u(0) + u^\top(t_f) S u(t_f) \]  

(27)

where S is the weighting matrix to be selected significantly larger. Eq. (27) is the eventual evaluation function modified to strict energy consumption of CMG and restraint conditions of control inputs. Energy optimal paths can be obtained strictly by FBA using this evaluation function Eq. (27).

4. NUMERICAL SIMULATIONS

4.1. Simulation Conditions

The purpose of numerical simulations is to show the availability of the proposed logic in large-angle rapid maneuver in points of energy consumption and safe use of CMG. Therefore a mission of the simulations assumes a 60deg Roll-to-Roll maneuver around roll axis in 20s. The satellite body is rigid with the following inertia matrix:

\[
I = \begin{bmatrix}
1500 & 0 & 0 \\
0 & 1500 & 0 \\
0 & 0 & 1500
\end{bmatrix} \text{kg} \cdot \text{m}^2
\]  

(28)

The other parameter values are shown in Table 1. The state variables at the beginning and at the end of this Rest-to-Rest maneuver are constrained as follows:

<table>
<thead>
<tr>
<th>Parameters</th>
<th>Values</th>
</tr>
</thead>
<tbody>
<tr>
<td>Inertia moment of CMG wheel</td>
<td>( I_{CMG} ) = 0.11 kgm²</td>
</tr>
<tr>
<td>Rate of CMG wheel rotation</td>
<td>( \omega_{CMG} ) = 6000 rpm</td>
</tr>
<tr>
<td>Rotational rate limit of CMG gimbal</td>
<td>( \delta_{max} ) = \pm 1.0 rad/s</td>
</tr>
<tr>
<td>Rotational acceleration limit of CMG gimbal</td>
<td>( \ddot{\delta}_{max} ) = \pm 3.0 rad/s²</td>
</tr>
<tr>
<td>Skew angle</td>
<td>( \beta ) = 54.7 deg</td>
</tr>
<tr>
<td>Dimension of Fourier approximation</td>
<td>( N ) = 4</td>
</tr>
</tbody>
</table>

Table 1. Parameters and values

The simulation results for 60deg roll maneuver in 20s are presented in Figs. 4 to 7 and Table 2. Figures 4 and 6 are results of the proposed logic. Figures 5 and 7 are results of the existing logic which is GRSInverse steering logic.

4.2. Simulation Results

Figures 4 and 5 show that both logics can achieve the desired maneuver. The time histories of CMG gimbal angles are shown in Figs. 6 and 7. As can be seen in these figures, though GRSInverse logic is influenced by singularity problem in 7-10 seconds, the proposed logic is not affected by singular problem. Therefore the proposed logic can control CMG more smoothly and in safety without rapid and furious actuation than GRSInverse logic.

As shown in Eq. (30), there is a restriction in terminal CMG gimbal angles. In general terminal gimbal angles have no effect on the maneuverability of the agile satellite. Therefore terminal gimbal angles have never been considered by existing logics, such as GRSInverse steering logic. Nevertheless in case of the multitarget mission terminal gimbal angles are related to the singular problem of the next maneuver because the terminal gimbal angles become the initial gimbal angles at the next maneuver. Therefore the terminal gimbal angles are set to return to zeros in Eq. (30). This constraint is natural under the ideal circumstance in the respect of the conservation principle of angular momentum.

The numerical simulations are carried out under these conditions. Moreover numerical simulations of the existing logic, which is a feedback control logic using GRSInverse method in Eqs. (11) and (12), are also carried out as the comparison object of the proposed logic.
Especially the rapid and furious actuation of the existing logic near the singularity state reaches the limits of CMG gimbal angular rates. Therefore existing logic has disadvantage in point of safe use of CMG. As can be seen in Table 2, energy consumption of the proposed logic can be reduced by approximately 46% in comparison with the result of GSRInverse logic. Therefore it was shown that the proposed logic is more effective than GSRInverse logic in the point of energy consumption of CMG. Additionally, from the result in Fig. 6 it can be seen that the terminal gimbal angles of the proposed logic become all zeros and this property is the advantage in the actual implementation.

5. CONCLUSIONS

In this paper the feedforward control logic based on the energy optimal path planning was proposed for the large-angle and rapid maneuver control of the agile satellite using CMG. The method of path planning is FBA and this method can consider energy consumption, the control requirement and the terminal gimbal angles. Numerical simulation results show the availability of the proposed logic in points of energy consumption and safe use of CMG during the rapid maneuver. The future work is to design a feedback control system including the proposed feedforward control logic in order to obtain the robustness about various errors and disturbance.

REFERENCES


