A Reasonable and Robust Attitude Determination and Control System for Nano Satellite TSUBAME

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Abstract

TSUBAME is the fourth satellite developed in the Laboratory for Space Systems (LSS) at Tokyo Institute of Technology and Institute of Space and Astronautical Science (ISAS) in Japan Aero Exploration Agency. The 50kg TSUBAME is a demonstration microsatellite for Earth and astronomical observation technology. It is planned that TSUBAME will be launched into a sun-synchronous orbit at about 500km altitude. With the purpose to push the envelope of what can be achieved by university satellite projects, a control moment gyro (CMG) is used for high-speed attitude maneuvering. Although the CMG can produce larger torque comparing with reaction wheels or magnetic torques but it is easy to be trapped into singularity. This challenge makes attitude determination and control system (ADCS) becomes one of the key technologies of TSUBAME. This paper describes architecture of the ADCS, its components, algorithms, implementation and verification methods. Main aspects of the ADCS are new proposed attitude control and attitude determination algorithms, a multiplicity of sensors and actuators due to redundancy and/or power saving, and a health monitoring system to reduce the possibility of system failures. Finally, the paper shows some numerical simulated results of the main ADCS working modes. The numerical simulations are conducted based on a hardware-in-the-loop test platform, which was built for simulating on-orbit environment together with real flight-software and sensors, actuators hardware of the TSUBAME.

1 Introduction

Matunaga laboratory at ISAS/JAXA and Tokyo Tech has been developing a nano-satellite “TSUBAME” as shown in Figure 1, an earth and astronomical observation technology demonstration satellite, as the fourth satellite following CUTE-I, Cute-1.7 + APD and Cute-1.7 + APD II. Its size is about 50cm x 50cm x 50cm with a total mass of 50kg, and planned to be launched into sun-synchronous orbit at 500km altitude. The main missions of TSUBAME are as follows: 1) on-orbit demonstration of newly developed micro control moment gyros (CMGs), 2) polarized gamma-ray burst (GRB) observation using a hard X-ray Compton Polarimeter (HXCP) and wide field burst monitors (WBMs) with high-speed attitude maneuvering using Micro-CMGs, 3) Earth observation with a small high-resolution optical camera under high accurate pointing attitude control.

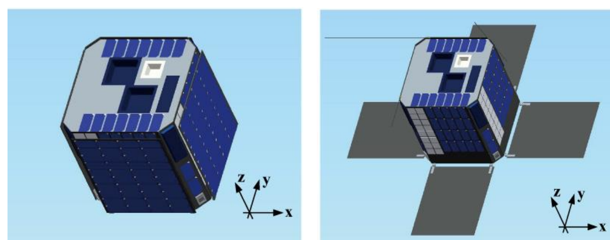

Figure 1 TSUBAME schematic diagram
(left: before solar panel deployment) (right: after solar panel deployment)

Now, our team is developing the flight model of TSUBAME aiming at the launch in the fiscal year of 2014. In Attitude Determination and Control System (ADCS), performance evaluation tests and calibration tests of the sensors, and magnetic test for measuring the residual magnetic moment of the satellite had been conducted. In TSUBAME, especially, the demonstration of high-speed maneuvering control, which requires a high accurate pointing and high generated torque, by using CMGs is a high risk mission. Therefore, the Attitude Determination and Control System (ADCS) needs to cope with the unique problems due to CMGs.

This paper shows the design, development and verification processes of the TSUBAME satellite. The ADCS’s components, algorithms, implementation and verification methods are described. Also embedded software and attitude simulation environments, such as Model in the Loop simulation (MiLS), Software in the

Loop Simulation (SiLS), and Hardware in the Loop Simulation (HiLS), are presented. Main aspects of the ADCS are new proposed attitude control and attitude determination algorithms, a multiplicity of sensors and actuators due to redundancy and/or power saving, and a health monitoring system to reduce the possibility of system failures. Finally, the paper also shows some numerical simulated results which are conducted based on a SiLS test platform.

2 Micro Control Moment Gyro (CMG)

One of the TSUBAME missions is to demonstrate high-speed maneuver with micro Control Moment Gyro (CMG). A newly developed micro CMG is shown in Figure 2. A wheel, a gimbal ring, an angle sensor, and power supply section (collector ring) are sealed in one cylindrical structure. One atmosphere of inactive gas (nitrogen gas) is poured into the inside of CMG model, which prevents evaporation or bearing lubricating oil. The maximum length of this model is 150mm, and the mass per one unit is 1kg, which size and mass are reduced as much as possible. The pyramid type CMG configuration, which consists of four this CMG as shown in Figure 3, is mounted on TSUBAME. This nano satellite using the CMG system can realize high-speed maneuvers.

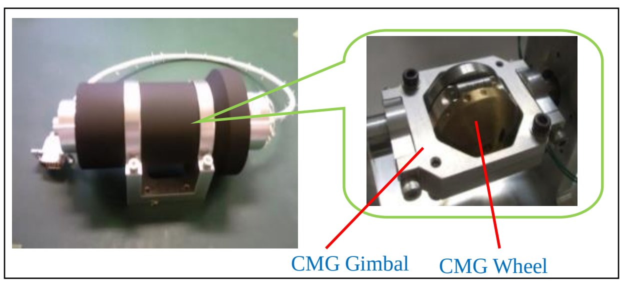

Figure 2 Micro control moment gyros (CMGs)

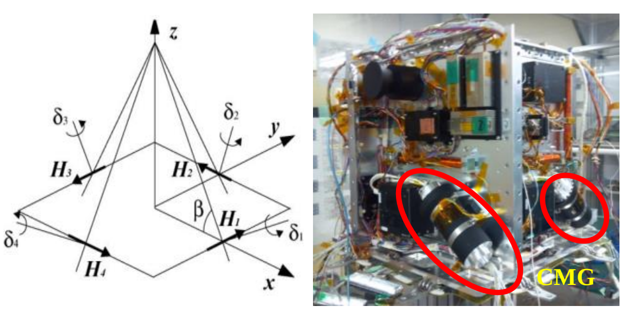

Figure 3 Pyramid type configuration of CMG system for Nano Satellite TSUBAME

3 Overview of ADCS

The attitude determination sensors of ADCS consist of six sun sensors, a 3-axis magnetometer, three MEMS Gyros, three Fiber Optic Gyros (FOGs), a GPS receiver, and two star trackers. On the other hand, the attitude control actuators of ADCS consist of three magnetic torques (MTQs) and four CMGs. These sensors and actuators are selected and used according to the specified operation mode as shown in Figure 4. The operation modes of the satellite are divided in two main groups: MTQ Control Mode and CMG Control Mode.

The MTQ Control Mode is also understood as power saving modes because only low-power sensors and actuators are used in this mode. While in the CMG Control Mode, all low-power sensors, actuators and three more high-power devices are used. The hardware information of sensors and actuators will be detail presented in next part. A block diagram of ADCS is shown in Figure 5.

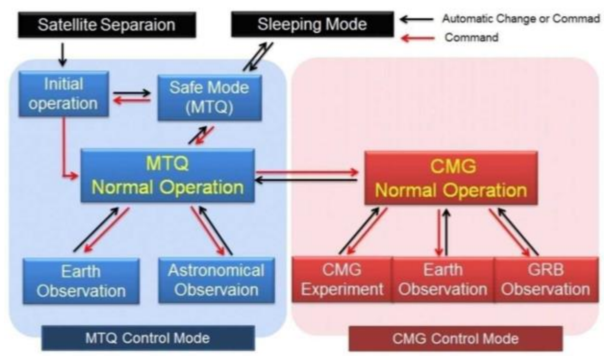

Figure 4 TSUBAME Operation modes

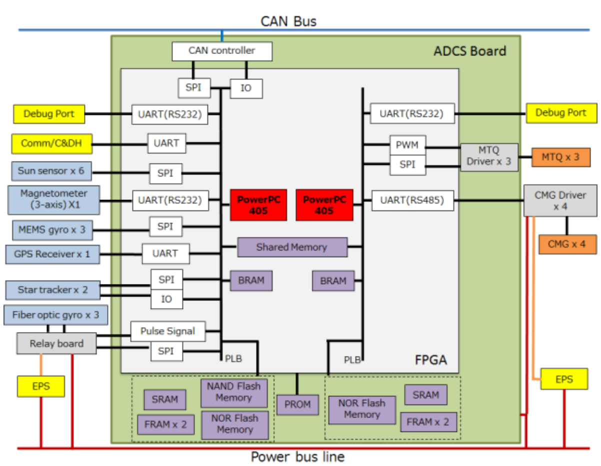

Figure 5 A Block Diagram of TSUBAME ADCS

3.1 Mission Requirements

The mission requirements of TSUBAME are shown in Table 1 with nine ADCS working modes. In which, the most difficult ADCS missions are in the mode number 1, 8 and 9.

	ADCS Mode	Attitudeel Control	Requirement					備考
			Pointing Target	Attitude Determination Accuracy	Pointing Accuracy	Stability	Rapidity	
1	Initial Operation before SAP Deployment	Detumbling Spin Up Sun Poiting	N/A	N/A	N/A	N/A	Sun Angle < 67deg within 30,000s	Separation Rate : 20deg/s (Maximum Value)
2	Initial Operation after SAP Deployment	Spin Stabilization	Sun	<3 deg (Sun Angle)	Sun Angle < 20 deg	N/A	N/A	Spin Rate ~5deg/s
3	MTQ Normal Mode	Spin Stabilization	Sun	<3 deg (Sun Angle)	Sun Angle < 20 deg	N/A	N/A	Spin Rate ~5deg/s
4	MTQ Science Observation	Spin Stabilization	Target Astronomical Object	<1 deg (Sun Angle)	Target Direction < 3deg	N/A	N/A	
5	MTQ Earth Observation	Spin Stabilization	Landmark on Earth	<1 deg (Sun Angle)	N/A	N/A	N/A	
6	CMG Normal Observation	Three-axis Stabilization	Sun	< 1 deg (Three Axis)	Sun Angle < 8 deg	N/A	N/A	
8	CMG Science(GRB) Observation	Three-axis Stabilization	Target Astronomical Object	< 0.3 deg (Three Axis)	Target Direction < 3deg	N/A	90deg /15s	
9	CMG Earth Observation	Three-axis Stabilization	Landmark on Earth	< 0.072deg (Three Axis)	Landmark on Earth < 0.72deg	< 0.25 deg/s	N/A	

Table 1: ADCS mission requirement of the TSUBAME

In the ADCS mode no. 1, because of the critical power generation, it requires the satellite should be stabilized and oriented to the Sun direction within 30,000 seconds from the deploying time. In the ADCS mode no. 8, the CMG Earth Observation mode, from the request of the camera system, the stability of satellite should be about 0.25 deg/s or less, and pointing accuracy should be smaller than 0.72 deg. In the ADCS mode no. 9, the CMG Science (GRB) Observation mode, it requires that satellite can perform a maneuvering up to 90 degrees within 15 seconds. This is because GRB abruptly occurs in the undefined direction and disappears in a short time. Therefore, this is the main reason why a high-speed position changing technology using the CMG is requested.

3.2 Attitude Determination and Control System strategy

The operation of TSUBAME is mainly divided into three phases: initial phase, mission phase and normal phase. After departure from the launch, the satellite enters the initial operation phase. The main objective of this phase is sun acquisition and solar panel deployment for power charging. The control requirement is to achieve a sun-pointing stabilization within 2200min (based on battery capability and power consumption during the initial phase). The pointing accuracy required is less than 20deg. Only MTQ will be adopted as active actuator. Previous studies have shown that the pure magnetic control can only realize sun pointing stabilization when the residual magnetism is small. It is vulnerable against the external disturbance. Thus a spinning sun acquisition method is adopted refer to the Reimei satellite design5). The large angular momentum introduced by spinning motion can provide gyroscopic stiffness effect against external disturbances.

After the deployment of solar panel, the satellite can either move into mission phase or remain sun-pointing stabilization as normal operation phase according to the ground command. During the X-Ray observation mission, because the duration of prompt emission is generally short (~ 40 s), a rapid maneuver capacity (~90 deg within 15 s) is required using CMGs. Then the observation can be carried out within 15s after the burst occurrence. Fig.3 exhibits the mission sequence of X-Ray observation. Due to the relative large power consumption of CMGs, the operation limitation of CMGs is about 1200min. And the required pointing accuracy is less than 4deg. For Earth observation mission, the required pointing accuracy is less than 2deg. Finally, after finishing the mission or the power is inadequate, the satellite will be switched back to normal sun-pointing phase for safe operation.

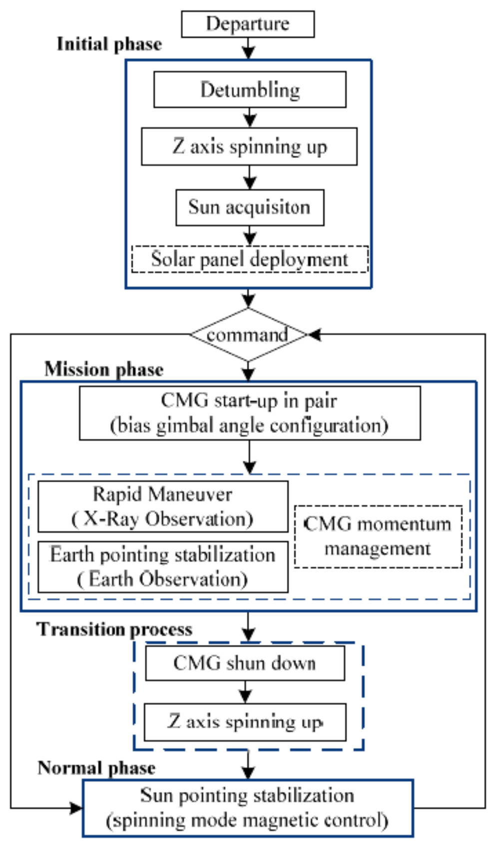

Figure 6: Attitude Control Flow

4 ADCS Algorithms Design

In this section the attitude control problem for mission phase is considered. Table 3 shows the algorithms that is used in ADCS

Subsystem	Algorithm	Input
ODS	EKF	GPSR
ADS	QUEST	TAM, SAS
	REQUEST	TAM, SAS, MEMS Gyro
	EKF, ASEKF, UKF, RTSEKF, SEKF	TAM, SAS, MEMS Gyro
	EKF, UKF	STT, FOG
ACS	B-dot	TAM, MTQ
	Spin with ANC	TAM,SAS, MTQ
	PD (q-feedback), SMC, HOSMC,MSMC	CMG, FOG, STT

Table 2: Attitude Determination and Control Algorithms

4.1 Attitude Determination System (ADS)

In TSUBAME, several new proposals algorithm are applied to increase the reasonably reliable property of ADCS.

Firstly, a real-time tuning separate-bias extended Kalman filter (RTSEKF) for robust spacecraft attitude estimation in the presence of measurement biases is applied. The adaptive mechanism applied during calculation of Kalman gain in the “bias” filter could help the attitude estimator system have a better response to the larger initial estimated error and unpredicted sensor bias models. The RTSEKF has higher accuracy and faster convergence speed than the conventional methods like separate-bias extended Kalman filter, extended Kalman filter, and even unscented Kalman filter. The computation cost of RTSEKF could be controlled by choosing the accuracy of the optimal process or using the filter reduction form, ASEKF [1] [2].

Secondly, an automatic and real-time tuning (RTUKF) procedure for the process noise Q and measurement noise R matrices of the UKF for the redundant filter is applied. The proposed filter uses the attitude data from the main estimator as the truth reference data to improve the convergence speed and estimated accuracy of the redundant estimator. Based on that, to control the system power consumption, the proposed filter is used with the selectable duration and frequent repeat of turn-on time of the main estimator data.

Finally, a residual-based adaptive unscented Kalman filter (AUKF) is applied in fault detection and diagnosis (FDD) to recovery when the variation of sensor noise magnitude is existing.

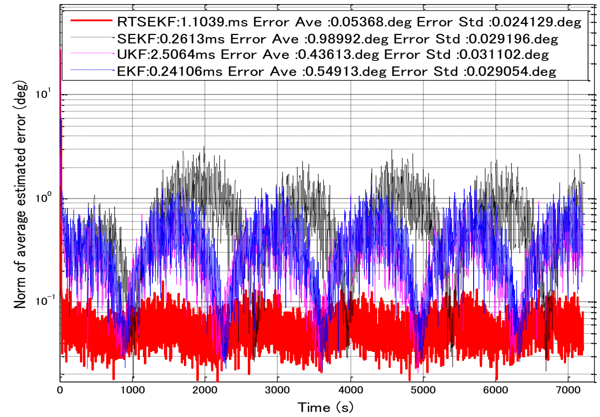

Figure 7: Simulation result of RTSEKF vs SEKF, EKF, and UKF for attitude and sensor bias estimation

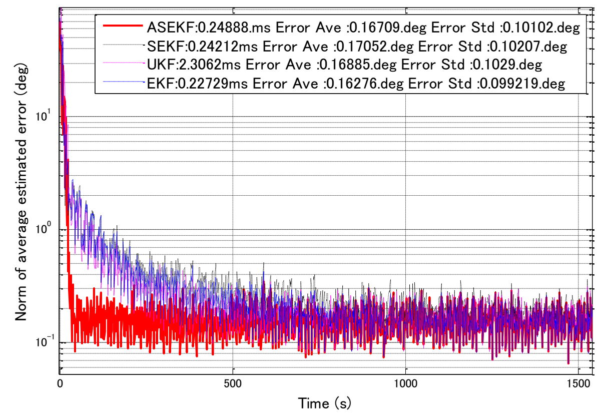

Figure 8: Simulation result of ASEKF vs SEKF, EKF, and UKF for attitude and sensor bias estimation

4.2 Attitude Control System

The attitude control system (ACS) controls the attitude of TSUBAME to orient its solar generators, its payload units (HXCP, WBMs), optical instruments and antennas. The operation of TSUBAME’s ACS is based on two kinds of actuators: low-power with MTQ and high-power, high-speed with CMGs. For simple and backup missions, the actuator MTQ is used. To control MTQ, the B-dot algorithm is used for detumbling process. B-dot algorithm only requires one sensor, TAM, and one actuator, MTQ. In the case of Normal Operation (MTQ) is performing, the TSUBAME need to be spin up and pointed to the Sun direction. ACS also uses Anti Nutation Control (ANC) to stabilize during spin at [0.2 0.2 0.2] deg/s.

For CMGs attitude control, because the mission requirement of TSUBAME is very high in maneuvering speed, so a new attitude control algorithm is proposed. The new proposed algorithm (modified sliding mode control) for CMGs is based on the higher order sliding mode control (HOSMC) [3]. It is useful in handling the attitude control system against external disturbance and unknown friction effects. Furthermore, HOSMC can remove the chattering effect if appropriate relative degree was chosen. To overcome the singularity problem inherent the control moment gyros, a modified steering law for Single Gimbal Control Moment Gyros (SGCMG), which aims at precisely exporting commanded torque is discussed based on the optimal output torque capability principle [4].

4.2.1 Modified Sliding Mode Control Law Development

First, we consider a general second system structure

$$\dot{x} = f(t, x, T_c) \tag{1}$$

In the Eq. (12), x is state variable that takes on values in a smooth manifold X. t is time. This equation is needed to be nonsingular. The design approach for sliding mode control law involves choosing the sliding manifold that achieves the control objective and designing the sliding conditions that satisfy the sliding manifold.

We can deduce the spacecraft equations of motion with control CMGs from Eq. (1), ω is angular velocity is body frame, T_c and T_d is control torque and disturbance torque, respectively.

$$\dot{\omega} = -I^{-1}\omega \times I\omega + I^{-1}T_d + I^{-1}T_c \tag{2}$$

where the part $-I^{-1}\omega \times I\omega + I^{-1}T^d$ is called the drift function and I^{-1} is the input matrix.

The modified rodrigues parameters can be transformed directly from the classical rodrigues parameters and it is a useful attitude representation for attitude control algorithm. In this paper, the MRP vector σ is defined in terms of the principle rotation elements $(\Phi, \hat{e})$ is.

$$\sigma = \tan\frac{\Phi}{4}\hat{e} \tag{3}$$

Modified sliding variable s is choosing as [5].

$$s = \Delta\omega + P_1\Delta\sigma + P_2\int_0^t \Delta\sigma dt \tag{4}$$

where $\Delta\omega$ is the difference in attitude angular velocity, P_1 is a 3×3 positive constant matrix, $\Delta\sigma$ is the error vector in the MRP form, P_2 is a symmetric positive-definite constant matrix. Suppose the system stays on the non-linear sliding manifold $s = 0$, and then the system will track the desired state. The attitude tracking error in MRP represents when on the sliding manifold can be transformed in the following expression:

$$\Delta\dot{\sigma} = -\frac{1}{4}\left[\left(1 - \Delta\sigma^2\right)I_{3\times 3} + 2\Delta\sigma\times + 2\Delta\sigma\Delta\sigma^T\right] \left(P_1\Delta\sigma + P_2\int_0^t \Delta\sigma dt\right) \tag{5}$$

Linearizing the MRP kinematic differential equation, the following approximation is given

$$\Delta\dot{\sigma} \approx -\frac{1}{4}\Delta\omega \tag{6}$$

This paper uses the following Lyapunov function for the control system presented

$$V = 2\log(1 + \Delta\sigma^T\Delta\sigma) + \frac{1}{2}\left(P_2\int_0^t \Delta\sigma dt\right)^T \left(\int_0^t \Delta\sigma dt\right) \tag{7}$$

And

$$\begin{aligned} \Delta\sigma = 0 \quad &\rightarrow \quad V = 0 \\ \Delta\sigma \neq 0 \quad &\rightarrow \quad V > 0 \end{aligned} \tag{8}$$

Therefore, Eq. (7) is a valid Lyapunov candidate function. Differentiating Eq. (7) provides that

$$\dot{V} = \frac{4\Delta\sigma^T\Delta\dot{\sigma}}{1 + \Delta\sigma^2} + \left(P_2\int_0^t \Delta\sigma dt\right)^T \Delta\sigma \tag{9}$$

Using Eq. (9), $\dot{V}$ can be express as following

$$\dot{V} = -\Delta\sigma^T P_1\Delta\sigma - \Delta\sigma^T P_2\int_0^t \Delta\sigma dt + \left(P_2\int_0^t \Delta\sigma dt\right)^T \Delta\sigma \tag{10}$$

since $\Delta\sigma^T P_2\int_0^t \Delta\sigma dt = \left(P_2\int_0^t \Delta\sigma dt\right)^T \Delta\sigma$, Eq. (10) can transfer to the following form and result in

$$\dot{V} = -\Delta\sigma^T P_1\Delta\sigma \leq 0 \tag{11}$$

From the statement of differentiating of Lyapunov function, we can get the conclusion that the system is Lyapunov stable. The differential of sliding variable is deduced as

$$\begin{aligned} \dot{s} &= \Delta\dot{\omega} + P_1\Delta\dot{\sigma} + P_2\Delta\sigma \\ &= \dot{\omega} - R\dot{\omega}_r + \omega \times R\omega_r + \frac{1}{4}P_1 B(\Delta\sigma)\Delta\omega + P_2\Delta\sigma \end{aligned} \tag{12}$$

Substituting Eq. (5) to Eq. (12) and setting $\dot{s} = 0$, we can get the control expression T as following

$$\begin{aligned} T = I(&-I^{-1}\omega \times I\omega + I^{-1}T_d + R\dot{\omega}_r - \omega \times R\omega_r \\ &- \frac{1}{4}P_1 B(\Delta\sigma)\Delta\omega - P_2\Delta\sigma) \end{aligned} \tag{13}$$

This control of Eq. (13) is a sliding mode controller that keeps the sliding variable on the sliding surface. Since the sliding mode control always has a disadvantage which is called chattering effect. Here, a robust control algorithm which is formulated on the saturation function is introduced to prevent chattering on the controller as following

$$T_{sat} = \begin{cases} \dfrac{Ks_i}{\mu}, & (|s_i| < \mu) \\ sign(s_i), & (|s_i| < \mu) \end{cases} \qquad (14)$$

where K is a diagonal matrix and μ is a boundary thickness. Finally, the robust sliding mode control law based on the MRP is

$$T_c = T + T_{sat} \qquad (15)$$

There are three attitude control gains P_1, P_2, K should be determined. The gain selection method can be found in the reference 4.

4.2.2 Simulation

Table 3 shows the simulation parameter. Figure 8 and 9 show the comparison results with other classical algorithms. The new proposed method is not only faster but also reduces the singularity factor.

Inertia matrix. kg.m2	diag[1.751 1.788 1.836]
Skew-angle. Deg	54.75
Gimbal rate (MAX). rad/s	±1
Gimbal rate aceleration. rad/s2	1
Gimbal contrl step. s	0.1
Residual magnetic disturbance.A.m2	0.1
Initial attitude Euler angle. deg	[17.8,42.6,76.9]
Initial angular velocity, deg/s	[1.5, 1.5, 1.5]
Initial gimbal angle, deg	[0, 0, 0,0]
Initial gimbal rate, deg/s	[0, 0, 0,0]

Table 3: Simulation Parameters

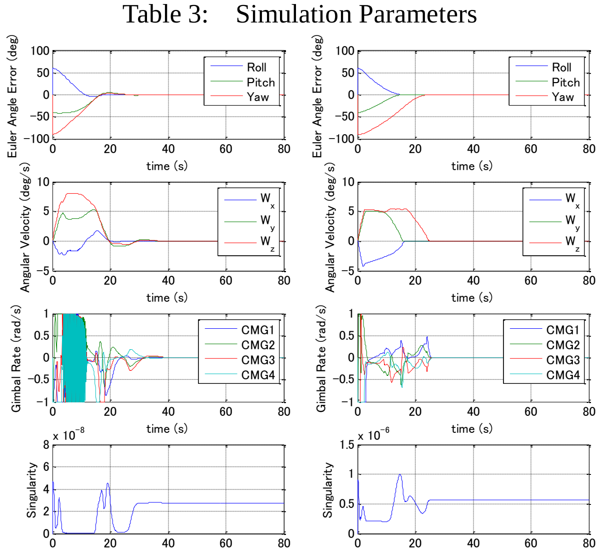

Figure 9: Simulation result of PD vs. M-SMC

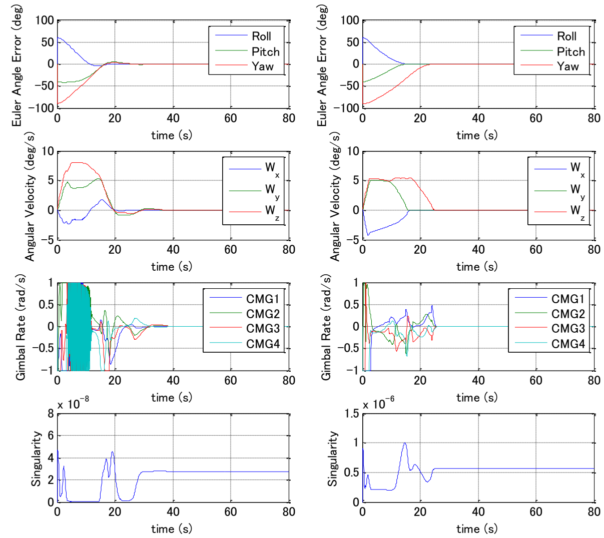

Figure 10: Simulation result of SMC vs. M-SMC

4.3 Hardware in the Loop Simulation (HiLS)

Hardware in the Loop Simulation (HiLS) realized by communications between the attitude simulator and On Board Computers (OBCs) is also being developed as shown in Figure 11. The purpose of this simulation is checking the performance of the software on the real system. This simulator consists of the OBCs, the actuators mounted on the satellite and the external computer which imitates the space environment and the sensors output. By confirming the performance of the software with the OBCs, reliability of the ADCS is also much improved.

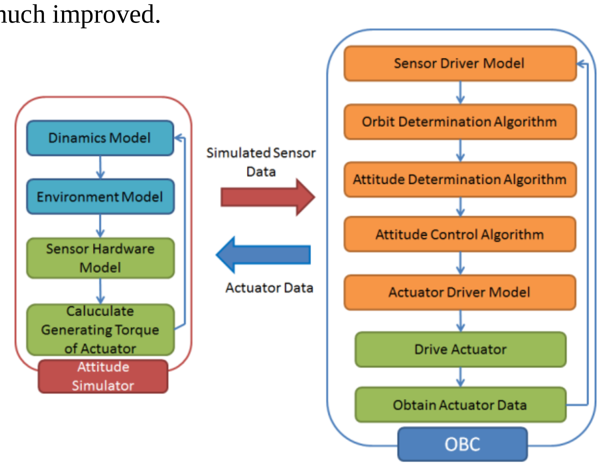

Figure 11: A Block Diagram of HiLS

HiLS environment should have variable settings according to the simulating purpose, because OBC works in real time and its operating speed is generally slower than one of MATLAB/ Simulink, and communication rates of variable sensors and actuators are also different, so that if simulations of such a communication is needed, a general real-time controller should be required as shown in Figure 12. In this environment, electrical signal level of the sensors and the actuators can be simulated and evaluated.

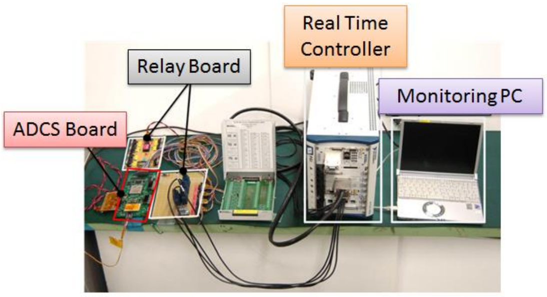

Figure 12: HiLS Using Real-time Simulator

4.4 Simulation Result using Simple HiLS

The simple HiLS environment is shown Figure 13. In this environment, two processors are used independently in OBC (FPGA) mounted on the satellite. One of the processors calculates the proposed control method, steering law of CMGs, and the estimated torque. It also drives the real CMG system, and each control algorithm of CMG is calculated using measurement data of wheel rate and gimbal angle obtained from sensors inside CMG in real time. This processor corresponds to the processor in the actual system. On the other hand, the other processor calculates the dynamics of the satellite, and it corresponds to an attitude simulator. In simulation loop, the estimated torque, quaternion, and angular velocity are transmitted and received between the two processors.

Figure 13: Photo of Simple HiLS

Currently, the HiLS is on developing. Therefore, this part shows some simulation results using SiLS. Firstly, satellite is simulated from detumbling process to Sun pointing process. Assume that the velocity of TSUBAME after deployment is 20 deg/s around z axis. Figure 14 shows the three-axis angular velocity of the satellite during in detumbling from $\omega = \begin{bmatrix} 0 & 0 & 20.0 \end{bmatrix}^T$ deg/s, and process of spin-up to $\omega = \begin{bmatrix} 0 & 0 & -5.0 \end{bmatrix}^T$ deg/s processes.

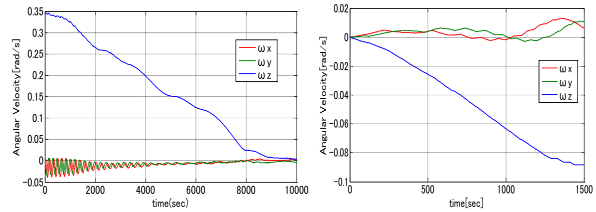

Figure 14: B-dot for detumbling and spin-up

After that, TSUBAME performs the sun pointing process together with ANC. The initial angular velocity after departure is assumed to be 3deg/s about each axis. Only angular velocity and sun angle curves are presented for brevity. Figure 11 show the Sun angle from the simulation starting point. This simulation is the worst case when the solar panel point to the opposite direction of Sun vector. It took about 7200 seconds to reduce Sun angle under 67 degree, as requirement. Therefore, the total time which TSUBAME need from deployment time to point to the Sun direction is around 18,700 second. It also is smaller than the requirement, 30,000 second. The pointing accuracy is about 5deg during the sun lighting zone while drifted up to 20deg during the eclipse. Thus the control performance fulfills the mission requirement even with large residual magnetism disturbance (0.1A m2).

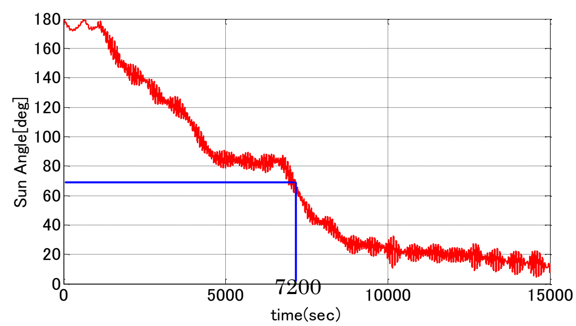

Figure 15: Sun angle during the Sun pointing process

5 Conclusions

This paper focuses on the attitude control system design for TSUBAME. Based on mission requirements under different modes, a hybrid control flow is proposed to guarantee a robust and safe operation against external disturbance. Currently, towards the launch, which is expected this year, TSUBAME development team is gearing the FM development in software development center. In the near future, using the attitude simulator described in this paper, a fully integrated FM system testing is performed to enhance the completeness of the satellite.

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