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Attitude Control Simulator for the Small Satellite and Its Validation by On-orbit Data of QSAT-EOS

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ABSTRACT

The role of the microsatellites has become significant due to the progress of performance and miniaturization of the electronics devices. For the purpose of earth observation the constellation flight of the microsatellites has made it possible to obtain the continuous data of the designated location and it is expected that the data be utilized in wide variety of the filed such as science, agriculture, industries, and climate monitoring. As the microsatellites take the LEO, the time of observation on the designated location is limited. Therefore the agile and precise pointing performance is required so as the attitude control accuracies. This paper presents the development of the attitude control simulator for the microsatellite for the use of design of the attitude control subsystem. Quaternion feedback control is proved to be effective and stable. Then the on-orbit data of the QSAT-EOS (Kyushu Satellite for Earth Observation System Demonstration) is analyzed to confirm the effectiveness of the quaternion feedback control. QSAT-EOS was launched in 2014 and now is in the deorbit phase after two year operation. Results of the on-orbit data analysis have suggested some necessities of improvement of the simulator and control algorithm which are the future subject.

OVERVIEW OF THE SIMULATOR

The purpose of this simulator is to design the attitude control subsystem to fulfill the requirements of the microsatellites and to define the specifications for the equipment of the subsystem. This simulator is used in the preliminary design phase to confirm the realizability of the attitude control requirements derived from the system study. Also this simulator is used in the detailed subsystem design phase to derive the requirements for the components such as sensors and actuators. After selection of the components to be used, the actual characteristics and performance of the components are to be modeled and installed in the simulator to study under more realistic and precise conditions.

Configuration of the Simulator

This simulator is composed of the four major blocks, the Orbit Generation Block, Target Generation Block, On-orbit Environment Block, and Attitude Control Block as shown in the Figure 1.

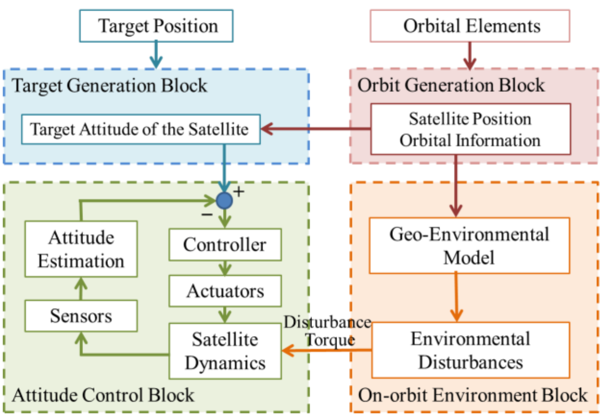


Figure 1: Major Components of the Simulator

The detailed features of these blocks are as follows.

Orbit Generation Block

This block computes the satellite position and velocities from the orbital elements and the results is to be sent to the target Generation Block and On-orbit Environment Block. Then the orbital elements are to be updated to the present time.

Target Generation Block

This block computes the vector from the satellite to the target and then computes the angular error between the pointing axis and the target vector. Pointing axis of the satellite is the optical axis of the camera for example. The target may be the subject on the surface of the earth such as the city, volcano, or constructions. Also the target may be the astronomical bodies such as the moon, the planets, or the stars.

On-orbit Environment Block

This block computes the disturbance forces and torques to be placed on the satellite on the orbit. Using the orbital data geo-magnetic moments, aerodynamic forces, and gravity gradient forces are computed. These disturbances are supplied to the Attitude Control Block and used to analyze the behavior of the satellite. Geo-magnetic model of the earth is IGRF model, atmospheric model of the earth is NRLMSISE model and gravitational field model of the earth is EGM96 model.

Attitude Control Block

This block includes the satellite dynamics model, control devises such as the actuator and the attitude sensors. This block generates the control forces from the error between the target attitude and present satellite attitude. The satellite dynamics is solved to obtain the satellite attitude and angular velocities.

Sensor models of the sun-sensor, the magnetic sensor, the star-sensor, and the Fiber Optical Gyro are installed. These sensors are selectively used as the requirements of control accuracy. The characteristics of each sensor such as the scale factor, bias offset, bias stability, etc. can be modeled

Actuator models of the reaction wheel devise and magneto-torquer are installed and selectively used as the requirements of control accuracy. The reaction wheel device is controlled its rotational speed and the maximum torque and the maximum angular momentum is limited.

Attitude Control Mode

This simulator deals with the following attitude control modes, which are the typical operation mode of QSAT-EOS. Immediately after the release from the launcher, the satellite will commence the de-tumbling control to reduce the angular motion of the satellite slow enough for onboard equipment to be operated properly. After completion of the de-tumbling control, the satellite will transit to the sun-acquisition mode to prepare for mission operation. As the major mission of QSAT-EOS is earth observation, the pointing of the instrument is the important attitude control mode.

Figure 2 shows the typical mode transition diagram.

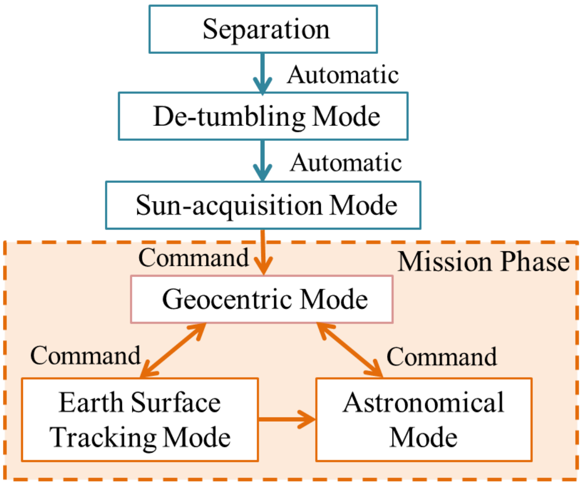


Figure 2. Typical Mode Transition Diagram

Attitude Control Mode in the Mission Phase

Geocentric Mode is to point the instruments axis such as of the optical camera to the center of the earth which means the camera observes the earth surface just below the satellite.

Earth surface tracking mode is to point the instruments axis to the designated location of the earth. As the control of this mode requires the agility and accuracy, the star-sensors are used to estimate the satellite attitude and the reaction wheel devices are used to steer the attitude of the satellite.

Astronomical mode is similar to the Earth Surface Tracking Mode except that the target is not on the surface of the earth but celestial bodies.

ATTITUDE CONTROL ALGORITHM

The attitude control modes are categorized into two modes. One is the angular rate damping and the other is attitude steering to the arbitrary direction. The de-tumbling mode is the former and the sun-acquisition mode and mission phase control modes are the latter.

De-tumbling Mode

The purpose of the de-tumbling mode control is to reduce the satellite rotational motion to the allowable level that the components and the instrument can operate properly. As no accurate attitude control is necessary, damping control using magneto-torquer, called B-dot control is applied. Control law is as follows.

$$M_i = -K \cdot \dot{B_i}^B \tag{1}$$

M_i : magnetic moment about the i-th axis of satellite
K : control gain
$\dot{B_i}^B$: rate of change of geomagnetism about i-th axis of satellite

$$\dot{B_{i,k}}^B \cong \frac{B_{i,k}{}^B - B_{i,k-1}{}^B}{\Delta t} \tag{2}$$

$B_{i,k}{}^B$: geomagnetism about i-th axis of satellite at time $k \cdot \Delta t$
$B_{i,k-1}{}^B$: geomagnetism about i–th axis of satellite at time $(k-1) \cdot \Delta t$
Δt : sampling time

Earth Surface Tracking Mode

The earth surface tracking mode is the typical mission mode and the geocentric mode and the astronomical mode are the special case of this mode. Only difference is if the target is on the surface of the earth or arbitrary location in space. When steering the specified axis of the satellite to the target point, the attitude of satellite can be arbitrary then the quaternion feedback control is selected to avoid the singularity problems.

Geometrical relationship between satellite and the target is illustrated in Figure 3.

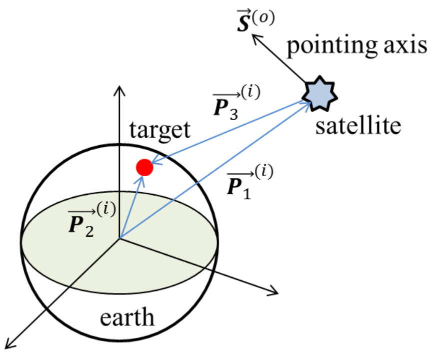


Figure 3. Geometrical Relationship between Satellite and Target

Quaternion Feedback Control Algorithm

The first step is to obtain the relationship between the pointing axis and the target direction. As shown in Figure 3 vector from satellite to the target is obtained by the following relation.

$$\overrightarrow{P_3}^{(i)} = \overrightarrow{P_2}^{(i)} - \overrightarrow{P_1}^{(i)} \tag{3}$$

$\overrightarrow{P_3}^{(i)}$: vector from satellite to target
$\overrightarrow{P_2}^{(i)}$: vector of target from the center of the earth
$\overrightarrow{P_1}^{(i)}$: vector of satellite from the center of the earth

Upper suffix (i) depicts the vector is defined in the inertial frame and (o) in the orbital frame.

Control quaternion from pointing axis ($\vec{S}^{(o)}$) to target vector ($\overrightarrow{P_3}^{(i)}$) is derived by following procedure.

Quaternion direction (the first three elements) is derived from cross product of $\vec{S}^{(o)}$ and $\overrightarrow{P_3}^{(i)}$. Rotation angle (the fourth element) is given by the dot product of them.

$$\vec{m}^{(o)} = \frac{\vec{S}^{(o)} \times \overrightarrow{P_3}^{(o)}}{\left\| \vec{S}^{(o)} \times \overrightarrow{P_3}^{(o)} \right\|} = \left(m_1{}^{(o)}, m_2{}^{(o)}, m_3{}^{(o)}\right)^T \tag{4}$$

$$\cos\Psi = \frac{\vec{S}^{(o)} \cdot \overrightarrow{P_3}^{(o)}}{\left\| \vec{S}^{(o)} \cdot \overrightarrow{P_3}^{(o)} \right\|} \tag{5}$$

$$\begin{aligned} \boldsymbol{q}_c &= (q_{c1}, q_{c2}, q_{c3}, q_{c4})^T \\ q_{c1} &= m_1{}^{(o)} \cdot \sin\frac{\Psi}{2} \\ q_{c2} &= m_2{}^{(o)} \cdot \sin\frac{\Psi}{2} \\ q_{c3} &= m_3{}^{(o)} \cdot \sin\frac{\Psi}{2} \\ q_{c4} &= \cos\frac{\Psi}{2} \end{aligned} \tag{6}$$

The second step is to generate the control torque to rotate the satellite in order to align the pointing axis to target vector. The procedure is as follows.

Proportional error feedback control is obtained by the following equation.

$$\boldsymbol{u}_p = -K_p \cdot \boldsymbol{e}_p \tag{7}$$

$\boldsymbol{u}_p$: proportional feedback control
K_p : proportional feedback gain
$\boldsymbol{e}_p$: proportional error vector

Proportional error vector $\mathbf{e}_p$ is the first three elements of error quaternion $\mathbf{q}_e$, which is obtained by the following equation.

$$\boldsymbol{q}_e = \boldsymbol{Q}_c \cdot \boldsymbol{q}_s \tag{8}$$

$$\boldsymbol{Q}_c = \begin{bmatrix} q_{c4} & q_{c3} & -q_{c2} & -q_{c1} \\ -q_{c3} & q_{c4} & q_{c1} & -q_{c2} \\ q_{c2} & -q_{c1} & q_{c4} & -q_{c3} \\ q_{c1} & q_{c2} & q_{c3} & q_{c4} \end{bmatrix} \tag{9}$$

$\mathbf{q}_S$ is the satellite attitude quaternion derived from Euler angles or direct output from the star sensor.

Angular rate error feedback control is simply derived by the following equation.

$$\boldsymbol{u}_d = -K_d \cdot (\boldsymbol{\omega}_s - \boldsymbol{\omega}_d) \tag{10}$$

$\boldsymbol{u}_d$: diferential error feedback control
K_d : differential error feedback gain
$\boldsymbol{\omega}_s$: angular rate of satellite
$\boldsymbol{\omega}_d$: target angular rate of satellite

$\boldsymbol{\omega}_d$ is the first three elements of derivative of satellite quaternion error obtained by the following equation.

$$\dot{\boldsymbol{q}}_e = \boldsymbol{Q}_c \cdot \dot{\boldsymbol{q}}_s \tag{11}$$

$$\dot{\boldsymbol{q}}_s \cong \frac{\boldsymbol{q}_{s,k} - \boldsymbol{q}_{s,k-1}}{\Delta t} \tag{12}$$

Integral error feedback control is given the following equation.

$$\boldsymbol{u}_i = -K_i \cdot \boldsymbol{e}_i \tag{13}$$

$\boldsymbol{u}_i$: integral error feedback control
K_i : integral error feedback gain
$\boldsymbol{e}_i$: integration of proportional error

$$\boldsymbol{e}_{i,k} \cong \boldsymbol{e}_{i,k-1} + \boldsymbol{e}_{p,k} \cdot \Delta t \tag{14}$$

Total control torque is the summation of the proportional error control, differential error control, and integral error control. As the maximum torque of the actuator is limited by its physical characteristics, then the control is to be normalized by the maximum torque limit.

$$\boldsymbol{u}_{total} = -K_p \cdot \boldsymbol{e}_p - K_d \cdot (\boldsymbol{\omega}_s - \boldsymbol{\omega}_d) - K_i \cdot \boldsymbol{e}_i \tag{15}$$

EFFECTIVENESS OF QUATERNION CONTROL.

Simulation Case 1: Geocentric Mode

Using the simple model of the microsatellite the effectiveness of the quaternion feedback control was examined. Conditions of the simulation are shown in the Table 1

Table 1. Simulation Conditions for Evaluation of Quaternion Control Effectiveness

Item	Values
Initial attitude	(5.0 , 5.0 , 5.0) [degrees]
Target attitude	(0.0 , 0.0 , 0.0) [degrees]
Initial Angular Rate	(0.1 , 0.1 , 0.1) [degrees/seconds]
Proportional Gain	K_p = (0.01 , 0.08 , 0.02)
Differential Gain	K_d = (0.26 , 0.74 , 0.35)
Moment of Inertia	$\begin{bmatrix} 1.69 & 0.0 & 0.0 \\ 0.0 & 1.71 & 0.0 \\ 0.0 & 0.0 & 1.56 \end{bmatrix}$ [kgm^2]

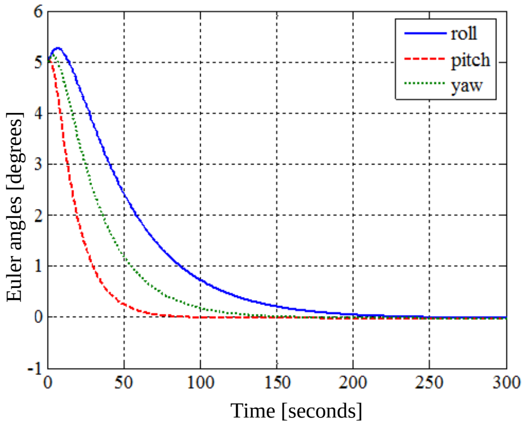


Figure 4 : Time history of Euler Angles

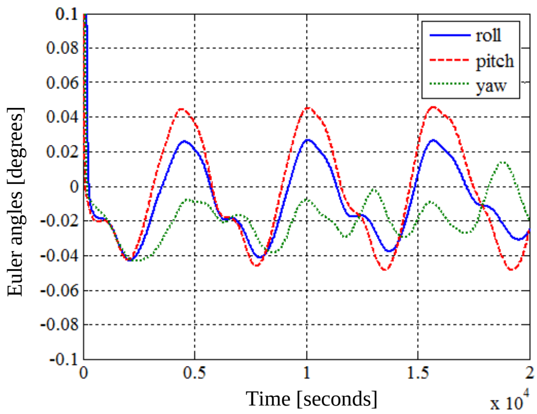


Figure 5 : Time history of Euler Angles Enlarged

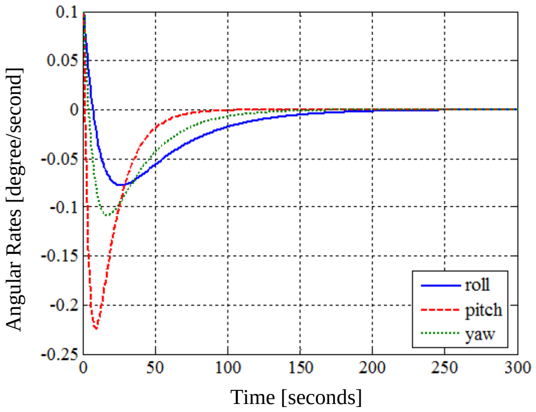


Figure 6: Time History of Angular Rates

Simulation case 1 was conducted in order to evaluate the convergence of attitude and steady state error of pointing performance in geocentric mode. Simulation commenced from the initial attitude and terminated at convergence of the attitude. Results of the simulation case 1 are shown in the Figure 4, Figure 5, and Figure 6.

Figure 4 shows the change of Euler angles of the model satellite after the attitude control of the geocentric mode was applied and indicates that the quaternion feedback control has successfully performed to steer the satellite to the target attitude within 250 seconds, four minutes. Figure 5 shows the residual error in the steady state. Required accuracy in pointing is less than 0.1 degrees and the steady state error is within this requirement. Figure 6 shows the time history of angular rate of the model satellite and indicates that the angular rate is less than 0.01 degrees/seconds and the quaternion feedback control satisfies the requirements of accuracy of 0.02 degrees/seconds. The time histories of the Euler angles and angular rates are of normal behavior and the quaternion feedback control is effective and stable.

Simulation Case 2: Earth Surface Tracking Mode

Another example of attitude control of microsatellite is of the surface tracking mode. The simulation conditions are shown in the Table 2. The target is the arbitrary point on the surface of the earth.

Table 2: Simulation Conditions of Surface Tracking Mode Control Demonstration

Item	Value
Orbital Elements	Semimajor Axis : 6909.137 [km] Eccentricity : 0.004 Inclination : 97.4 [degrees] Right Ascertion : 195.0 [degrees] Argument of Priapsis : 161.0 [degrees] True Anormaly : 0.0 [degrees]
Initial Attitude	(5.0 , 5.0 , 5.0) [degrees]
Initial Angular rates	(0.1 , 0.1 , 0.1) [degrees/seconds]
Proportional Gain	(0.3 , 0.42 , 0.3)
Differential Gain	$\begin{bmatrix} 11.8 & 0.28 & 0.28 \\ 0.28 & 12.0 & 0.28 \\ 0.28 & 0.28 & 11.0 \end{bmatrix}$
Integral Gain	(0.0036 , 0.0096 , 0.0048)
Moment of Inertia	$\begin{bmatrix} 1.69 & 0.0 & 0.0 \\ 0.0 & 1.71 & 0.0 \\ 0.0 & 0.0 & 1.56 \end{bmatrix}$ [kgm^2]
Target Location	Latitude : 33.0 degrees North Longitude : 130.0 degrees East

Simulation case 2 commenced at the initial state and terminated at the time 100 seconds after the passage over the target point on the surface of the earth. Results of the simulation case 2 are shown in Figure 7, Figure 8, Figure 9, and Figure 10.

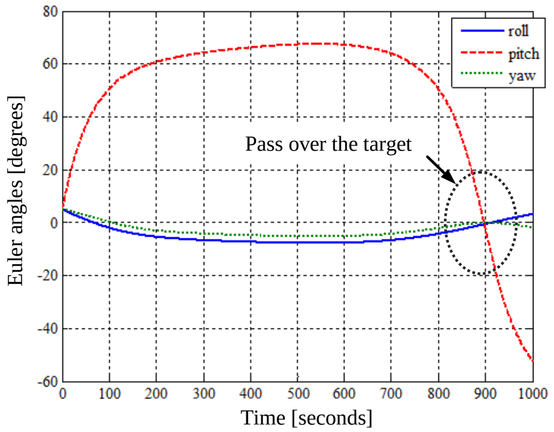


Figure 7: Time History of Euler Angles

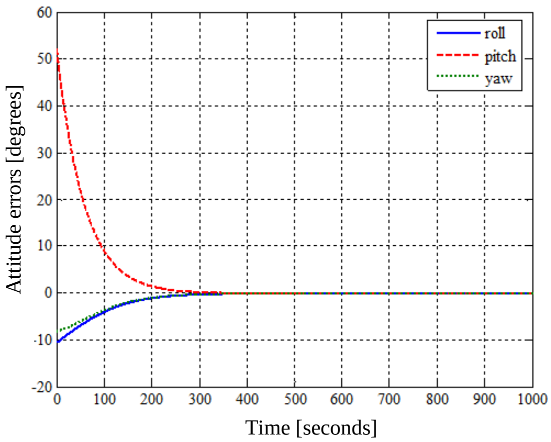


Figure 8: Time History of Attitude Errors

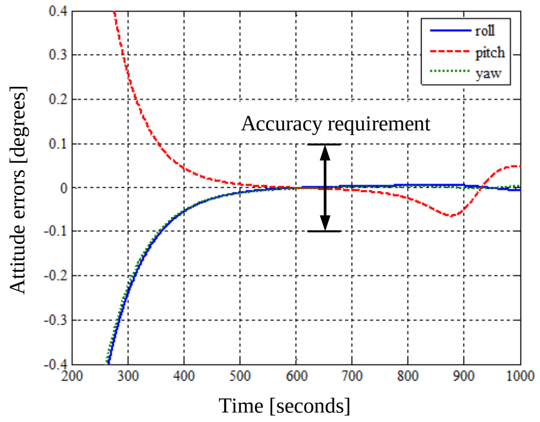


Figure 9: Time History of Attitude Errors Enlarged

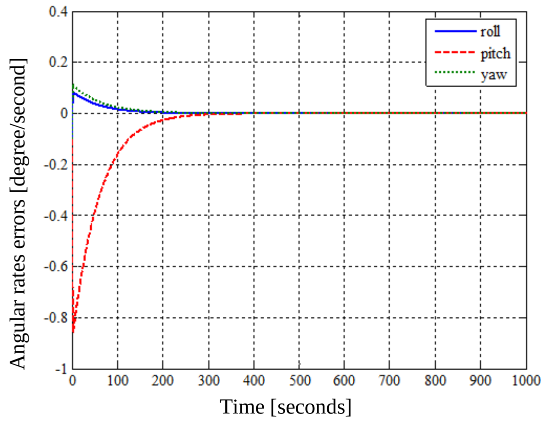


Figure 10: Time History of Angular Rate

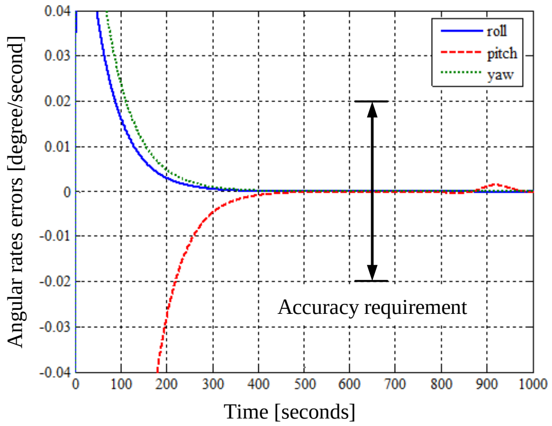


Figure 11: Time History of Angular Rate Enlarged

As the Figure 7 indicates, the attitude of the satellite changes rapidly when it passes over the target on the surface of the earth. The rapid change in pitch attitude occurs when the altitude of the satellite is low. At this time the errors in attitude and angular rate increase but still be within the accuracy tolerances by selecting the control gains properly. Using the orbital information the gain can be adapted properly and the errors in attitude and angular rate can be reduced low.

Simulation Case 3: Study on Effects of Parameter Uncertainty

Attitude control subsystem is designed under consideration of the properties of the satellite such as the mass properties or remanence. Some of the properties are estimated in the early phase of design and should be taken care of its uncertainty into design margin. Simulation case 3 was performed to study of the effects of uncertainty of the satellite properties. Simulation conditions are shown in the Table 3.

Table 3. Simulation Conditions for Study on Effects of Parameter Uncertainty

Item	Values
Initial attitude	(5.0 , 5.0 , 5.0) [degrees]
Target attitude	(0.0 , 0.0 , 0.0) [degrees]
Proportional Gain	K_p = (0.01 , 0.08 , 0.02)
Differential Gain	K_d = (0.26 , 0.74 , 0.35)
Moment of Inertia	$\begin{bmatrix} 1.69 & 0.04 & 0.04 \\ 0.04 & 1.71 & 0.07 \\ 0.04 & 0.07 & 1.56 \end{bmatrix}$ [kgm^2]
Remanence	(-0.514 , 0.042 , 0.093) [Am^2]

Simulation case 3 was conducted with the parameter of moment of inertia and moment of remanence to be deviated from their nominal values by 10%. Results of the simulation case 3 are shown in Figure 12 and Figure 13.

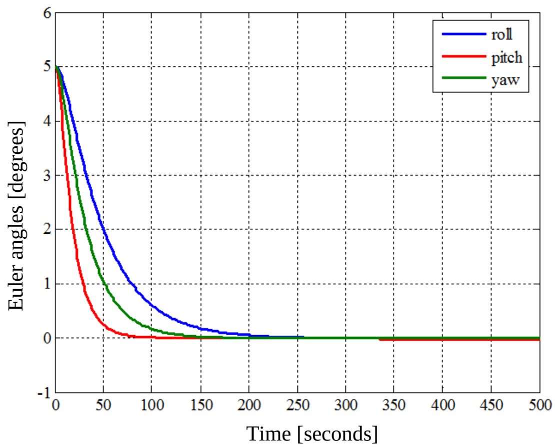


Figure 12: Time History of Attitude

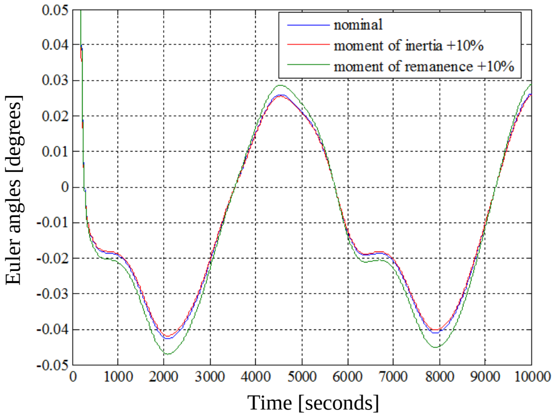


Figure 13: Effects on Satellite Attitude of Parameter Uncertainty (roll)

Figure 12 shows the time history of the satellite attitude and indicates no irregular behaviors in its motion. Figure 13 shows the satellite attitude with parameters deviated by 10%. Maximum difference in attitude occurs at the time of 2,000 seconds from the start of the simulation. Deviation from the nominal condition is 0.001 degrees for the case of moment of inertia and 0.005 degrees for the case of moment of remanence. The effect of remanence is five times larger than that of moment of inertia. As it is very difficult to measure the remanence of the satellite on the ground, some correlation should be made on orbit if more precise control is required.

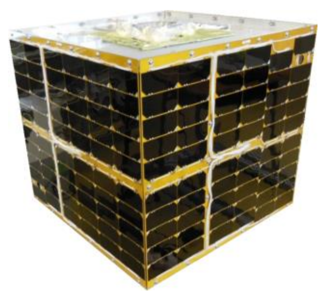
QSAT-EOS ON-ORBIT DATA ANALYSIS

The data from the QSAT-EOS (Kyushu Satellite for Earth Observation System Demonstration) launched in 2014 is analyzed to confirm the effectiveness of the quaternion feedback control and to find out improvements on the simulator.

Over View of QSAT-EOS

QSAT-EOS was developed by the Department of Aeronautics and Astronautics, Kyushu University and Institute for Q-shu pioneer of Space, Inc. (i-QPS). QSAT-EOS was launched on 6th November, 2014 from the Yasny Launch Base of Russia by Dnepr Launch Vehicle. It had been operated for about two years on orbit and now is in the de-orbit phase using expandable deorbit sail to accelerate the descent. Major missions of QSAT-EOS were earth observation by optical camera, measurement of geomagnetic field and water in the earth's atmosphere, and micro-debris measurement by thin film debris sensing device. Major specification and feature of QSAT-EOS is shown in the Table 4.

Table 4: Characteristics of QSAT-EOS

Item	Specification
Size	50 x 50 x 50 (cm) 50 x 50 x 350 (cm) : de-orbit sail expanded
Mass	50.6 (kg)
Orbit	Sun-synchronous (altitude 530km)
Mission	Earth observation (optical camera) Geomagnetic field (scientific magneto-sensor) Water in atmosphere (RF time delay) Micro-debris (thin film debris sensing device)
Attitude control	Type: Three-axis stabilized Sensors: Sun-sensor, Magneto-sensor, Star-sensor, Fiber Optical Gyro (FOG), GPS Actuators: Reaction wheel, Magneto-torquer
Appearance	

On-orbit Data Analysis of QSAT-EOS

Data Analysis Case 1: Geocentric Mode

The geocentric mode data was obtained during 02:04 and 02:15 o'clock UTC, 15th September, 2015. In this case attitude of QSAT-EOS was estimated using data from the sun-sensors, magneto-sensor, and FOG's. The ground tracking trajectory is shown in the Figure 14. The red markers are the projection of the satellite onto the surface of the earth and the blue markers are optical line of sight onto the surface of the earth. The gray regions depict the coverage of the Ku-band communication antenna.

Telemetry data was obtained with three second intervals but there were some fails of receipt of the data.

Figure 15 shows the attitude change during the geocentric control. Figure 16 and figure 17 show the time history of the angular rates of the satellite and the reaction wheel rotation speed respectively. QSAT-EOS commenced the geocentric control at 02:10:47 UTC and completed steering at around 02:12:05 UTC. Rotation speeds of each reaction wheels are coincident with the commands.

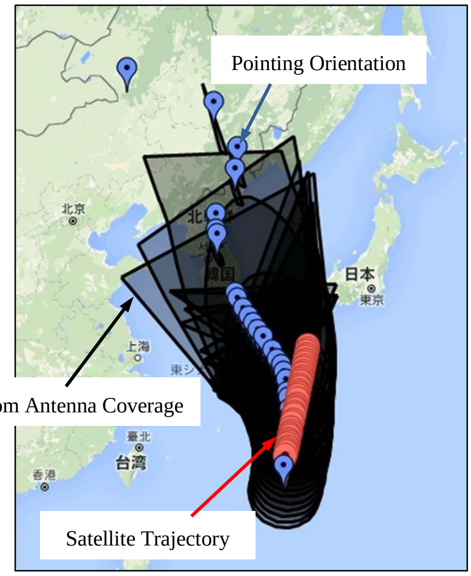


Figure 14: Trajectory of QSAT-EOS (Geocentric Mode)

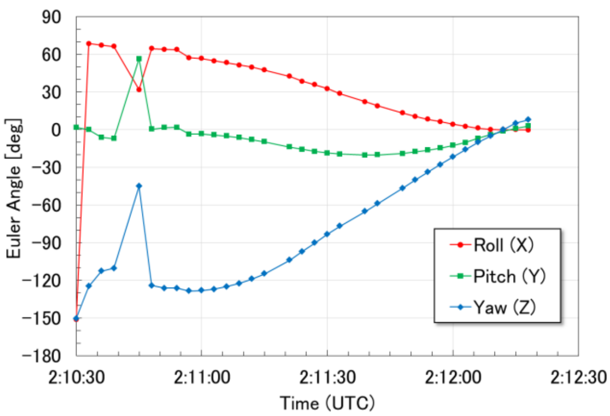


Figure 15: Time History of Euler Angles of QSAT-EOS (Geocentric Mode)

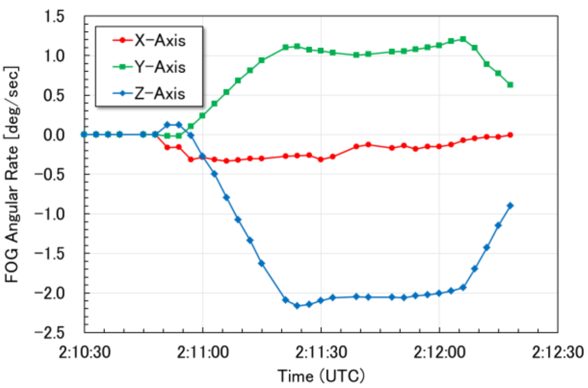


Figure 16: Time History of Angular Rate of QSAT-EOS (Geocentric Mode)

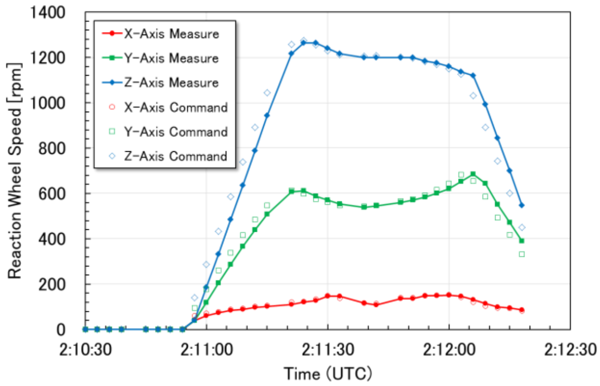


Figure 17: Time History of Rotation Speed of Reaction Wheels and Commands (Geocentric Mode)

Data Analysis Case 2: Earth Surface Tracking Mode

The earth surface tracking mode data was obtained during 03:02 and 03:06 o'clock UTC, 28th October, 2015. The attitude of QSAT-EOS was estimated using data from the sun-sensors, magneto-sensor, and FOG's as of the case 1. The ground tracking trajectory is shown in the Figure 18.

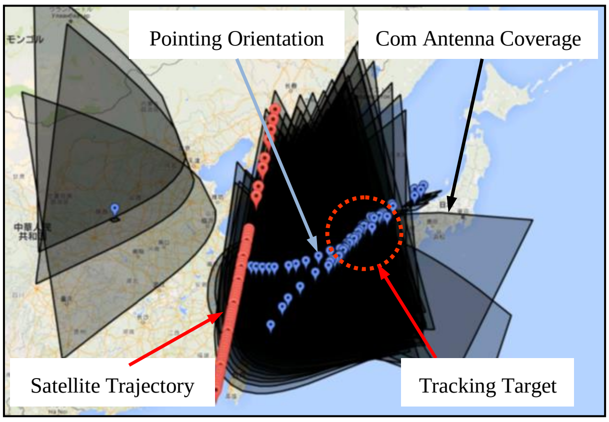


Figure 18: Trajectory of QSAT-EOS (Earth Surface Tracking Mode)

In this case the tracking target was Fukuoka city where the ground tracking station is located. QSAT-EOS commenced the earth surface tracking control at around 03:04:30 UTC and steered from the west to the east orienting to the target. At 03:05:00 UTC tracking was overshoot to the east and then QSAT-EOS steered backward to the west. At 03:06:00 UTC QSAT-EOS terminated the earth surface tracking mode and transferred to the stand-by mode orienting the 30 degree of roll attitude as planned.

CONCLUSION

Simulator to be utilized for the attitude control subsystem design has been developed and evaluated. Simple model simulation has indicated quaternion feedback control is effective to the small satellite attitude control and provides the stable results. Through the analysis of the on-orbit data of QSAT-EOS some issues to be improved for the simulator have been found.

Conclusions are the followings.

- Quaternion feedback control is effective to the small satellite attitude control.
- Uncertainty of satellite remanence is major error source of attitude control. Actual remanence should be measured on orbit using correlation maneuver or other sophisticated method such as the model predicting control.
- QSAT-EOS data has indicated the quaternion feedback control is effective on orbit but control gain adjustment has not been properly selected. More study is needed to obtain optimal control for the earth surface tracking mode.
- To construct the realistic model of the sensor and the actuator is needed to study the specification for the small satellite attitude control subsystem.

Based on the knowledge learned from the data analysis of QSAT-EOS, the attitude control system simulator is planned to be improved.

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