RELIABLE AND ROBUST IMPLEMENTATION OF ATTITUDE DETERMINATION AND CONTROL SUBSYSTEM AND INITIAL FLIGHT OPERATION RESULTS OF 50-KG-CLASS INTERPLANETARY SPACECRAFT PROCYON

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Abstract: *This paper summarizes the development of and initial flight results relating to the attitude determination and control system of PROCYON, which is the world's first micro-spacecraft that is capable of deep space missions. Because success in the automatic attitude control sequence from separation to sun pointing on the launch day is critical for its survival, a highly robust and reliable implementation is required for the automatic sequence. In order to increase robustness, path planning of the attitude maneuver is implemented to ensure that the strong separation disturbance is tolerated, and and fail-safe logic is built into the system. Reliability is enhanced by utilizing simulators for software and hardware verification. PROCYON was launched on December 3, 2014, and the automatic control sequence successfully executed in orbit almost as designed, except for the hunting around the target attitude owing to the unexpected time lag of the reaction wheels. A rapid change in the body angular momentum was also observed for the first few days after the launch. The analysis suggests that the primary cause is outgassing, a hypothesis that is consistent with the mass rate of outgassing for space flight instruments and spacecraft.*

Keywords: *Micro-Spacecraft, Deep Space, Attitude Determination and Control, Fail-Safe Implementation, Flight Results*

1. Introduction

An increasing number of space agencies, universities, and companies are developing nano/micro satellites that are launched into space. One of the main reasons is their unique feature that enables economical and short-term development [1]. For example, the 50-kg-class satellite SDS-4 (Small Demonstration Satellite-4), which was developed by Japan Aerospace Exploration Agency (JAXA) and launched in 2012, successfully demonstrated its micro-satellite bus system with Japan's first zero momentum three-axis control [2]. Another example is PRISM (Pico-satellite for Remote-sensing and Innovative Space Mission) weighing only 8.5 kg, developed by the University of Tokyo and launched in 2009, which achieved its primary mission of taking earth's images with a resolution of 30 m [3]. However, most of these "small" but "reasonably-performing" satellites were in low earth orbit, and had not yet been applied to deep space missions.

PROCYON (PRoximate Object Close flYby with Optical Navigation), which was developed by the University of Tokyo in collaboration with JAXA, is a 50-kg-class interplanetary spacecraft (categorized as a "micro-sized spacecraft"). After an Electric Delta-V Earth Gravity Assist (EDVEGA), PROCYON is flying toward an asteroid. PROCYON has two main mission objectives. The first objective is to demonstrate indispensable technologies of a micro-spacecraft bus system for deep space exploration. This includes communications, attitude/orbit determination and control, heat control, and solar power generation in the deep space environment. The second objective is to demonstrate advanced deep space exploration technologies that range over communication using a high-efficient X-band amplifier, precise orbit determination by very long baseline interferometry (VLBI), and flying-by observation of an asteroid using optical navigation [4]. In September 2013, PROCYON was selected as one of the secondary payloads for the 26th H-IIA launch. After the short-term development, it was launched with Hayabusa-2 in December 2014.

This paper first introduces the mission objectives of PROCYON and the specifications of the attitude determination and control system (ADCS). Next, the implementation requirements of reliability and robustness are described, which are unique to a secondary payload. We then explain the design of the attitude determination and control subsystem in addition to its verification process. We finally review the initial flight operation results.

2. Overview of Attitude Determination and Control System

2.1. Specifications

In order to meet the mission requirements, the system architecture of ADCS is established. Table 1 summarizes the fundamental attitude control modes of PROCYON. The angular velocity control mode is used when the body spin rate should be controlled, for instance, during initial spin-rate dumping after the separation from the rocket. The sun search mode is also a part of the angular velocity control mode whose purpose is to detect the sun's direction, and the sun pointing mode is the attitude control for maintaining the solar panels perpendicular to the sun's direction. Three-axis stabilization is used when one wants to fix the attitude against inertial frame (i.e., keep middle gain antenna (MGA) pointing to the earth), while precise three-axis stabilization can achieve more precise inertial-fixed attitude control by applying filtering techniques such as Kalman filter and low pass filter.

To realize these control modes, appropriate components are selected. Table 2 shows ADCS sensors/actuators on PROCYON and Fig. 1 shows the mounted positions of the ADCS components. PROCYON has one fiber optic gyro (FOG: constituting a unit that can measure three-axis angular velocity), one star tracker (STT), and five non-spin (two dimensional) sun aspect sensors (SASs). In comparison with large missions, realizing a redundant system is more difficult to PROCYON owing to the resource restriction in terms of spatial and electric power issues. Hence, PROCYON

Table 1. Fundamental attitude control mode of PROCYON.

No.	Control mode	Sensors/actuators	Usage example
(1)	Angular velocity control	FOG, RW	Spin-rate dumping
(2)	Sun search	FOG, RW	Sun-search
(3)	Sun pointing	FOG, SAS, RW	Sun acquisition for solar charging
(4)	Three-axis stabilization	STT, FOG, RW	MGA pointing to the earth
(5)	Precise three-axis stabilization	STT, FOG, RW	Scientific observation

Table 2. **ADCS sensors and actuators on board and their main specifications [5].**

Component	Role	Specifications
FOG (Sensor)	Angular velocity determination	Range: $\pm$ 10 deg/s Random noise : 4.7×10^{-4} deg/s Nominal bias : 2.8×10^{-3} deg/s
SAS (Sensor)	Sun direction determination	Accuracy : 1 deg Field of view : 100 deg $\times$ 100 deg
STT (Sensor)	Attitude determination against inertial frame	Accuracy (boresight) : 0.02 deg Accuracy (cross axes) : 0.002 deg Field of view : 8 deg $\times$ 8 deg
RW (Actuator)	Attitude control	Moment of inertia: 7.16×10^{-4} kgm^2 Maximum spin rate: 6000 rpm Delay : 0.5 sec
CGJ (Actuator)	Angular momentum control	Fuel: Xenon (used as cold gas) Thrust : 20.85 $\pm$ 0.85 mN

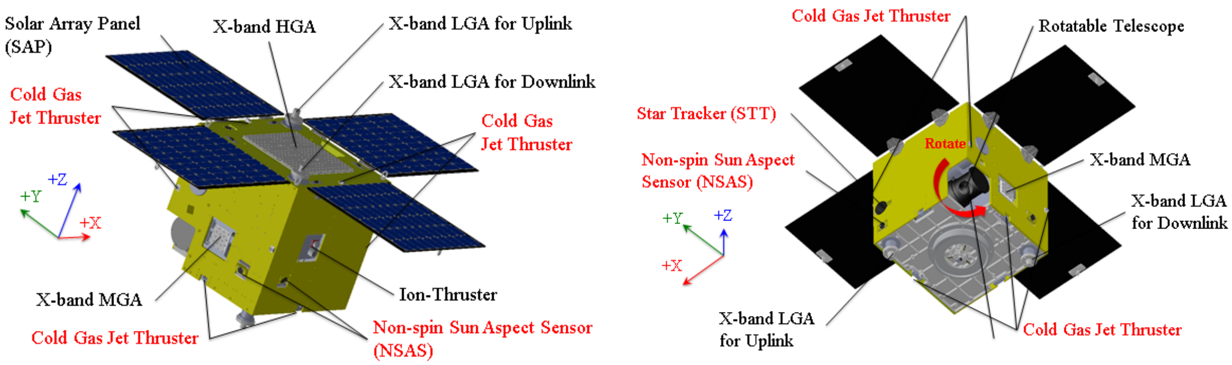

Figure 1. Mounted position of ADCS components of PROCYON.

has no backup FOG and STT on board. The actuators have four reaction wheels (RWs: one of which is aligned with the skew axis) and eight cold gas jet thrusters (CGJs). The CGJ is the only component that is used for RW unloading because magnetic torquers cannot be used for that purpose in the deep space environment. CGJ is a newly developed component that is unique to PROCYON, whose fuel tank is shared by the ion propulsion system [5] [6].

2.2. Difficulty on Launch Day

With regard to the operation, the critical time is the launch day. Table 3 shows the nominal sequence after PROCYON's separation from the rocket in relation to attitude control. On the launch day, PROCYON was estimated to experience the following two difficult constraints.

(Constraint 1) On the worst assumption, a separation disturbance of 10 deg/s (0.57 Nms) is added when PROCYON is released from the rocket.

(Constraint 2) Without sunlight, PROCYON survives for only one hour owing to capacity of its internal battery.

Table 3. Nominal sequence after the separation from rocket in relation to attitude control.

No.	Sequential event
1	Separation from rocket
2	Solar array paddle deploy
3	Spin-rate dumping
4	Sun search
5	Sun pointing

Regarding Constraint 1, PROCYON begins its attitude control for spin-rate dumping several hundred seconds after the separation from the rocket. PROCYON has to succeed in spin-rate dumping control even under the worst separation condition that is 10 deg/s (equivalent to 0.57 Nms in the angular momentum expression) in a certain direction. One possible measure to take is to use all RWs to absorb the body spin rate, but the maximum compensable angular momentum by four RWs is 0.50–1.02 Nms, which may saturate the RWs and finally cause the attitude to go out of control. Another possible measure is to use the CGJs for canceling out the excessive angular momentum. However, as mentioned in the previous section, the CGJ is a newly developed component and is being used in the space environment for the first time, so using the CGJs may entail an inherent and incalculable risk. Thus, a trade-off is necessary regarding whether to use the RWs or CGJs.

Turning to Constraint 2, after completion of spin-rate dumping, PROCYON starts sun search until the sun is detected, after which sun pointing begins. At this point, PROCYON has to succeed in sun pointing control and start solar charging within one hour because without sunlight its internal battery can provide electricity for only one hour. Moreover, it has to continue sun pointing control for at least several hours as all earth stations in Japan performing telemetry-command operations for deep space missions are assigned to other missions, which means human-in-the-loop operations are impossible during that period. Therefore, the sequence from spin-rate dumping control to sun pointing control has to execute automatically. The difficulty is that although it is the first time PROCYON is working in the space environment, a highly reliable and robust implementation is required for this automatic sequence because its success is critical for the spacecraft's survival.

To solve these problems, the following strategies are established:

(Strategy 1) Avoid staying in or passing through the directions where the angular momentum capacity of four RWs is exceeded.

(Strategy 2) Verify the whole automatic sequence from spin-rate dumping control to sun pointing control with software in the loop simulator (SILS) and hardware in the loop simulator (HILS).

(Strategy 3) Build fail-safe logic in the system to handle the situation if at least a single ADCS component failure occurs during the automatic sequence.

Note that Strategies 1 and 2 directly correspond to Constraints 1 and 2 respectively, and Strategy 3 is added to enhance robustness.

3. Development Results

3.1. Capacity Analysis of Angular Momentum

Table 4 summarizes the worst disturbance conditions at the separation. In Tab. 4, the separation status "Anomaly" denotes the separation when a spacecraft is successfully separated using the backup method, not the nominal method, which causes a larger separation disturbance. To increase safety, we consider that the spacecraft has sufficient robustness against the worst disturbance even if a single failure relating to the separation occurs such as

- All four RWs work normally but anomalous separation is performed.
- Nominal separation is performed but one of the four RWs does not work.

The capacity of angular momentum in each direction is calculated by considering the alignment of the RWs and using the linear programming technique. Figure 2 (left) shows the calculation result when all four RWs work normally but an anomalous separation is performed, indicating that the maximum capacity of angular momentum in the weakest direction is 0.499 Nms, and the weak directions appear parallel to the X, Y, Z axes of the body coordinates. Figure 2 (right) shows the calculation result when two-color mapping is applied at the bound of 0.571 Nms (red if higher and blue if lower). Figure 2 (right) suggests that the spacecraft cannot maintain its attitude owing to saturation of RWs when the norm of the angular momentum is larger than 0.571 Nms and the direction is in the blue region. In other words, even under the worst disturbance the spacecraft can maintain sun pointing control by planning of attitude maneuver paths and attitude target, when the direction of the angular momentum avoids the blue region.

Table 4. Worst disturbance condition at the separation.

Separation Status	Worst disturbance
Nominal	6.47 deg/s (equivalent to 0.362 Nms)
Anomaly	10.0 deg/s (equivalent to 0.571 Nms)

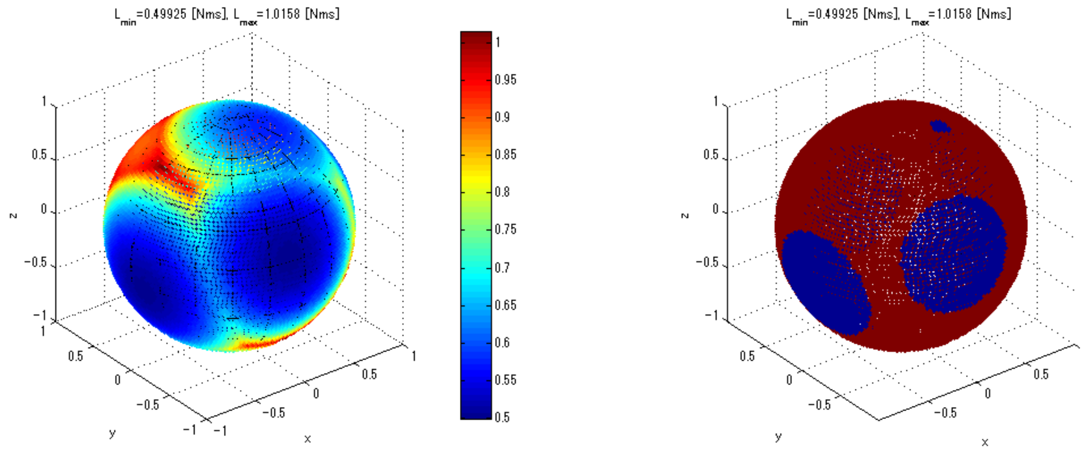

Figure 2. Capacity analysis results of angular momentum when all four RWs work normally but anomalous separation is performed.

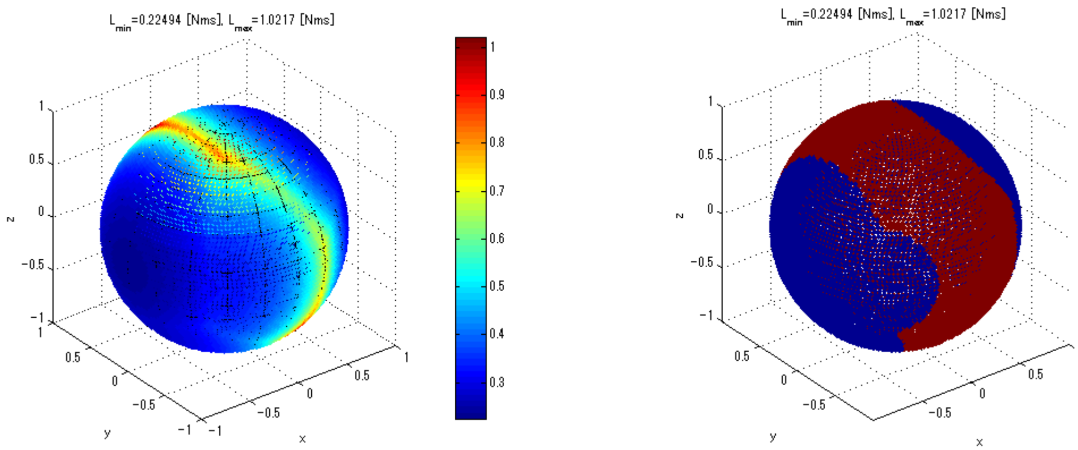

Figure 3. Capacity analysis result of angular momentum when nominal separation is performed but one of four RWs aligned with the body X axis does not work.

The same analysis is performed for the cases when one of the four RWs does not work under the nominal separation. Figure 3 show an example of the calculation result when nominal separation is performed but one of the four RWs aligned with the body X axis does not work. Figure 3 shows that the maximum capacity of angular momentum in the weakest direction of this configuration is 0.225 Nms and the weakest direction appears parallel to the X axis. By applying two-color mapping at the bound of 0.362 Nms, Fig. 3 (right) is obtained. Figure 3 (right) shows that although there is a blue region that is larger than that in Fig. 2 (right), it is still possible to avoid the hazardous region by attitude planning.

3.2. Fail-Safe Implementation

In order to increase robustness, fail-safe logic is built into the system. We define the fail-safe requirement on the launch day as “the automatic control sequence successfully executes and maintain sun pointing control even when at least a single ADCS component failure occurs”. We next design fail-safe actions and functions of fault detection, isolation and recovery (FDIR) corresponding to each component failure. The attitude control is then carefully investigated for each component failure situation in various separation conditions. Note that the implemented fail-safe logic increases safety not only on the launch day but also during all the operation phases. We introduce two possibly hazardous situations and the corresponding measures in the following.

Example 1: Failure of FOG

In this case, if an FOG failure occurs, spin-rate dumping under the automatic sequence cannot be carried out because of the absence of information on angular velocity. The implemented measure skips spin-rate dumping and initiates sun search directly when an FOG failure is detected. Whereas nominal sun search is a part of the angular velocity control mode because the information of FOG is used, the prepared sun search mode for the FOG failure searches the sun, using RWs by outputting constant feed-forward torques until the sun is detected.

Example 2: Failure of SAS on +Z body surface

In this case, if failure of SASs on the +Z body surface which is on the surface solar array paddle occurs, sun pointing control does not complete because the attitude information (sun direction) close to the target attitude is obtained only by SASs on the +Z body surface (the other SASs are located on the ±X or ±Y axes). The purpose of the implemented measure is to detect the sun's direction with any of the SASs on the sides first, and then start sun pointing by propagating the angular velocity information with the bias and noise of FOG. The attitude may drift as time passes owing to the bias and noise, but an attitude error can be detected by monitoring the level of solar power generation. When an attitude error is detected, the automatic control sequence restarts from spin-rate dumping. Thus, the spacecraft can survive until the human-in-the-loop operation begins.

3.3. Verification of Software and Hardware

For software/hardware verification, a simulation environment called SILS and HILS is used. SILS is the closed-loop simulator for software verification, which can verify not only ADCS software but also other subsystem software (which enables complete software verification). Figure 4 shows the structure of SILS. In contrast, HILS can verify software, hardware, and their communication with the on board computer and component drivers. Figure 5 shows structure of HILS. Detailed information on SILS/HILS is available in [5].

In order to ensure reliability, the whole sequence from the separation to sun pointing control is verified by SILS and HILS, changing the initial conditions such as initial attitude and disturbance. Figure 6 shows one example of simulation results from SILS when the initial attitude is precisely the sun's direction and the disturbance is given as [4, 4, 1] deg/s for each body axis. After the separation (0 s), the spacecraft starts spin-rate dumping from 22 s to 40 s. Then it begins sun search (+1 deg/s spin around X axis) which continues until the sun is detected (103 s). It reaches the sun pointing attitude at 190 s and maintains the control.

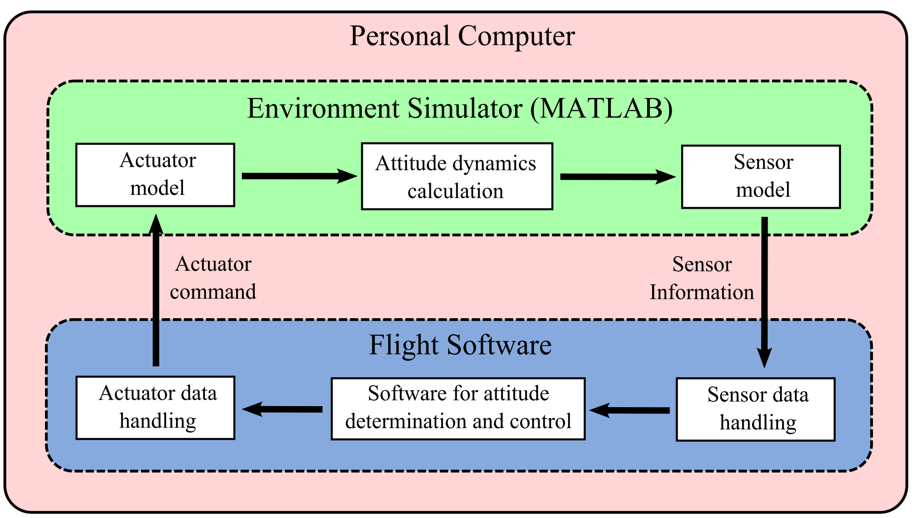

Figure 4. Structure of software in the loop simulator (SILS).

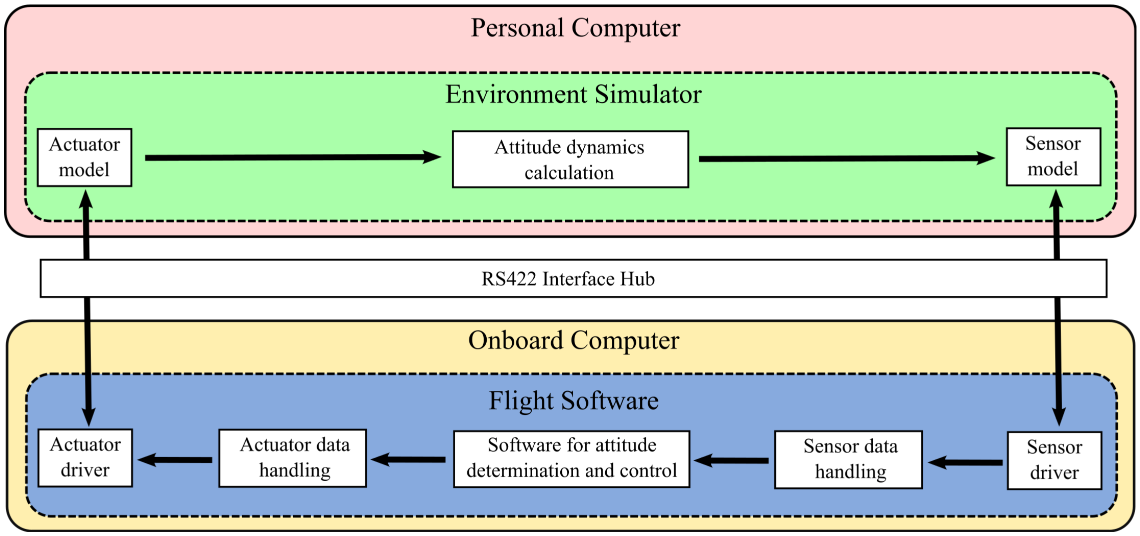

Figure 5. Structure of hardware in the loop simulator (HILS).

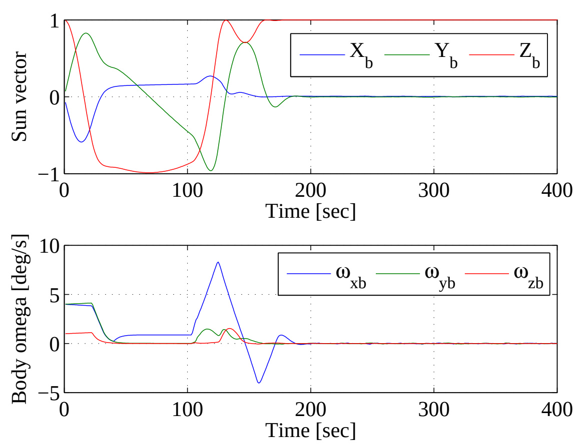

Figure 6. One example of simulation results from SILS for the entire sequence.

Furthermore, off-nominal cases under the whole sequence are carefully verified by SILS and HILS. Two examples of simulation results from SILS are shown in Fig. 7 and Fig. 8. In Fig. 7, the initial attitude is precisely the sun's direction and the disturbance is given as [4, 4, 1] deg/s for each body axis. Soon after the separation (0 s) the spacecraft finds the FOG cannot be used, therefore it skips spin-rate dumping and initiates sun search directly by delivering feed-forward torque through the RWs. After about 170 s, the attitude maintains sun pointing by using the information of only the +Z body SAS (though the spin rate of the Z body axis remains 1 deg/s owing to lack of body angular velocity information from the FOG).

It is clear from Fig. 8 that the initial attitude is precisely the sun's direction and disturbance is given as [4, 4, 1] deg/s for each body axis. After the separation (0 s) the spacecraft finds that the SAS on the +Z body surface cannot be used, but the sequence followed is almost the same as the nominal

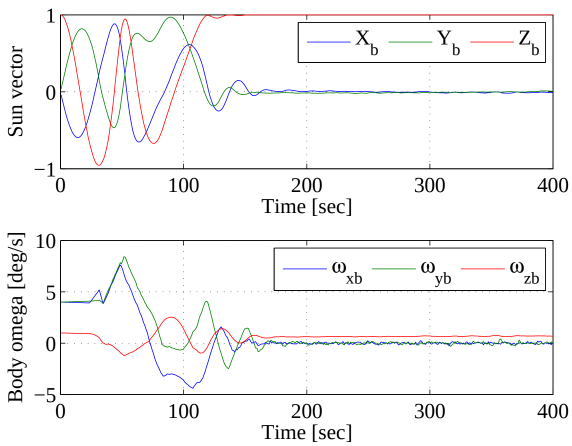

Figure 7. Simulation result from SILS (failure of FOG, corresponding to Example 1 in Section 3.2.).

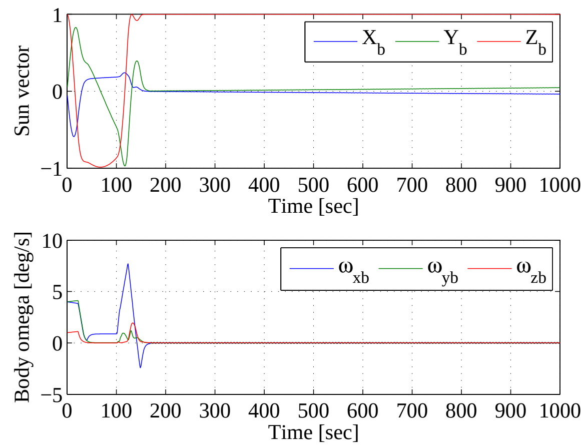

Figure 8. Simulation result from SILS (failure of SAS on +Z body surface, corresponding to Example 2 in Section 3.2.).

one. During sun pointing control, the attitude is maintained by propagating the information of the FOG instead of directly using the +Z body SAS's information. The attitude gradually drifts from the true target owing to the propagation error, but the error can be detected by monitoring the level of solar power generation; therefore electricity depletion of the internal battery can be avoided by restarting the sequence when the monitored level of the generation becomes lower than the threshold level.

4. Flight Results

4.1. Initial Attitude Control Result

PROCYON was launched in December 3, 2014, and the telemetry-command operation started 5.5 h after the separation. Figure 9 shows the initial flight result of the attitude control. In Fig. 9 (left), after the separation with light disturbance, the automatic attitude control executes successfully from spin-rate dumping to sun pointing. Sun search is skipped because the sun is within the field of view of the SAS on the +Z body surface when the status of the SAS becomes ON.

On the other hand, the result also shows that one unexpected result is the hunting around the target attitude seen in Fig. 9 (right). The primary cause is that RWs have a dead time of 0.5 s in addition to the delay time of 0.5 s while the control period is 1.0 s. This negatively couples the calculation of torque distribution of the RWs whose law can prevent each RW from canceling out the total angular momentum [7], resulting in unstable attitude control. However, this instability soon disappeared after another torque distribution law was applied, which calculates by simply multiplying the pseudo-inverse matrices of the four RWs' alignment and that of the required three-axis body torques.

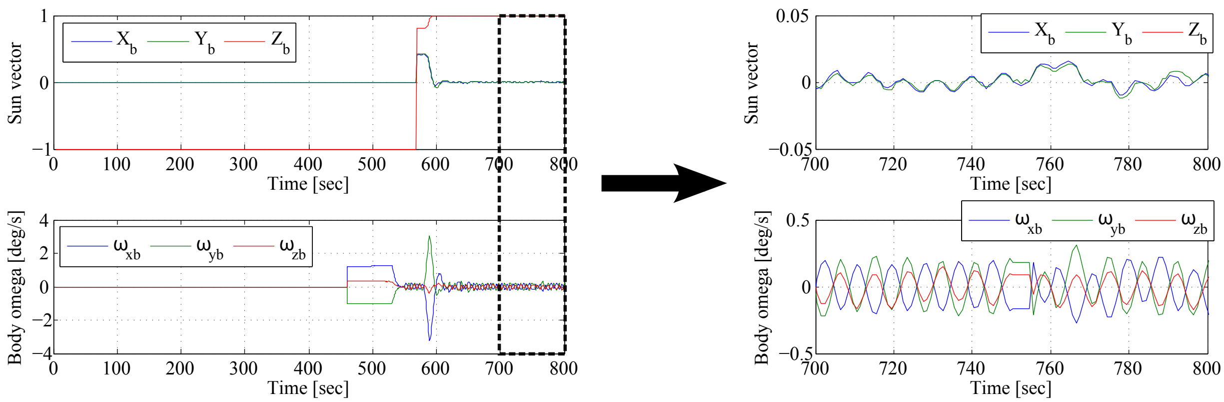

Figure 9. Initial flight result of attitude control after the separation. The statuses of FOG and SASs become ON at approximately 460 sec and 570 sec, respectively.

4.2. Disturbance Effect for the First Few Days

Another significant phenomenon is a rapid change in the body's angular momentum for the first few days after the launch. Figure 10 shows the body angular momentum between December 3 and 5 in 2014. The norm of the angular momentum changes by 0.03 Nms for the first 5.5 h after the launch and the change consistently becomes slower until the beginning of December 6.

In order to understand this phenomenon, five possible disturbance causes are identified: (1) earth's magnetic field, (2) earth's gravity, (3) moon's gravity, (4) solar radiation pressure, and (5) outgassing. In the case of PROCYON, the magnitude of the disturbance effects–except outgassing–can be approximately estimated. The orbital effect is also taken into account, which is the result of an exchange of the body's inner angular momenta by continuing the sun pointing attitude under orbital movement (hence it is an apparent torque). In the following, the torque on the Z axis is calculated because this torque is not more than at the level of the orbital effect unless a force is added, whereas large apparent torques are seen in the X and Y axes because their inner angular momenta are exchanged under sun pointing control with a slight rotation around the Z axis.

Figure 11 shows the calculation results of a given torque on the Z axis of PROCYON due to possible disturbance. In Fig. 11, no proposed cause agrees well with the flight result; the earth's magnetic field is responsible for significant disturbance only for a few hours after the separation, and solar radiation pressure, orbital effect, and gravity are too small to explain the flight result. Outgassing is thus the only possible remaining cause. Another fact is that the decrease rate of the disturbance torque is in good agreement with the mass rate of outgassing for space flight instruments and spacecrafts (i.e., in proportion to t^{-1} where t denotes time in hours [8]). Therefore, we conclude that outgassing is the most likely primary cause for the disturbance in the initial flight result.

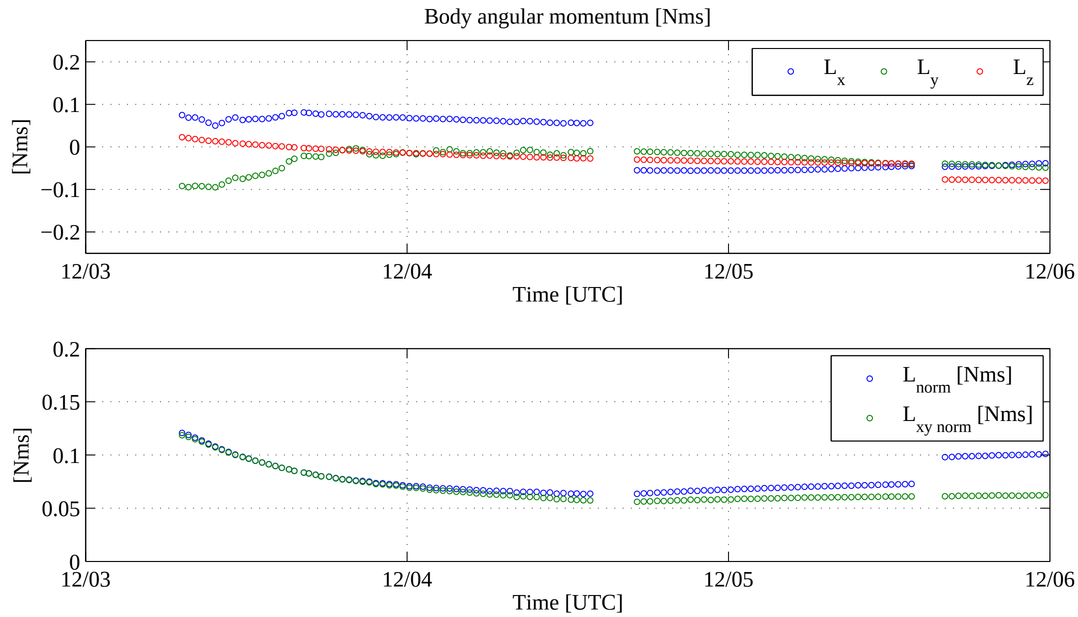

Figure 10. The body angular momentum between December 3 and 5 in 2014. The plotted period shows that PROCYON maintains a sun pointing attitude and the blank periods show that the attitude of PROCYON is not sun pointing. Unloading was executed at 16:00 on December 5.

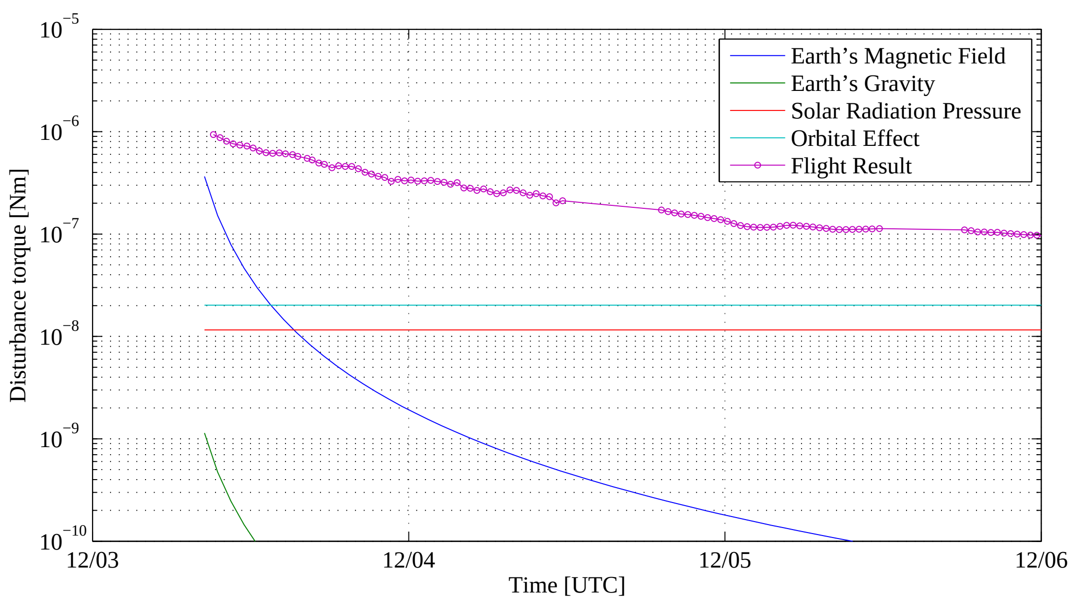

Figure 11. Calculation results of a given torque on the Z axis of PROCYON due to a possible disturbance. Some parameters specific to PROCYON are assumed: the remaining magnetic moment is 0.5 Am^2, the difference in the moment of inertia $(I_{zz} - I_{xx})$ or $(I_{zz} - I_{yy})$ is 0.2 kgm^2, the torque on the Z axis by solar radiation pressure is a little ($\approx 10^{-8}$ Nm) under the sun pointing attitude. The orbital effect is calculated as the maximum change rate in the angular momentum on the +Z axis when the norm of the angular momentum is 0.1 Nms. The effect of the moon's gravity is not shown in this figure because it is small (a maximum of 10^{-13} Nm) compared to the other candidates.

5. Conclusion

PROCYON is the world's first micro-spacecraft that is capable of deep space missions. In terms of attitude control, the critical time is the launch day. The difficulty is that although it is the first time PROCYON is working in the space environment, a highly robust and reliable implementation is required for this automatic sequence because its success is critical for its survival. To increase robustness, path planning of the attitude maneuver is implemented to ensure that the strong separation disturbance is tolerated, and fail-safe logic is built into the system. Reliability is enhanced by utilizing simulators: SILS and HILS. After the short-term development, PROCYON was launched in December 3, 2014. The automatic control sequence successfully executed in orbit almost as designed, except for the hunting around the target attitude due to the time lag of the RWs. A rapid change in the body's angular momentum was also observed for the first few days after the launch. The analysis suggests that the primary cause is outgassing, a hypothesis that is consistent with the mass rate of outgassing for space flight instruments and spacecraft.

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