Acta Astronautica 145 (2018) 515–527

Contents lists available at ScienceDirect

Acta Astronautica

journal homepage: www.elsevier.com/locate/actaastro

Discussions on attitude determination and control system for micro/nano/pico-satellites considering survivability based on Hodoyoshi-3 and 4 experiences

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ARTICLE INFO

Keywords:
Micro-satellite
Attitude control system
Satellite survivability
Mode transition

ABSTRACT

The recent advancement of micro/nano/pico-satellites technologies encourages many universities to develop three axis stabilized satellites. As three axis stabilization is high level technology requiring the proper functioning of various sensors, actuators and control software, many early satellites failed in their initial operation phase because of shortage of solar power generation or inability to realize the initial step of missions because of unexpected attitude control system performance. These results come from failure to design the satellite attitude determination and control system (ADCS) appropriately and not considering "satellite survivability." ADCS should be designed such that even if some sensors or actuators cannot work as expected, the satellite can survive and carry out some of its missions, even if not full. This paper discusses how to realize ADCS while taking satellite survivability into account, based on our experiences of design and in-orbit operations of Hodoyoshi-3 and 4 satellites launched in 2014, which suffered from various component anomalies but could complete their missions.

1. Introductions

Thanks to the recent rapid advancement of miniature component technologies and ease of purchase, micro/nano/pico-satellites have become more and more important tools for space development and utilization [1], some of which even have realized high level missions such as space sciences, microgravity experiment, communications, or remote sensing. Examples include SNAP-1 [2], INDEX [3], SDS-4 [4], RISING-2 [5], BIRD [6] and BRITE constellation [7]. In order to carry out sizable missions, three axis stabilization is required in most cases, which has been realized to a sufficient level in these projects.

In contrast to these successful satellites, many "university satellites" which incorporated a challenging three axis stabilization design could not survive or fulfill their missions because of failure or low performance of their ADCS (Attitude Determination and Control System). The common tendency of these failed projects is that the system design was carried out assuming that all the components and software in the ADCS functions properly, which is a rarity for university-level satellites. For example, if a satellite with solar cells only on its large solar paddle cannot control its attitude to have sun light on its solar paddles, it eventually fails because of power shortage. Actually, three axis stabilization is a high level function which requires proper functioning of many sensors, actuators, and onboard software. Therefore, ADCS should be carefully designed, fully considering the satellite survivability even with unexpected performance. This is especially important for "university-level" satellites developed by less experienced development teams and lack redundant components. Recently it has become rather easy to buy miniature ADCS components. Even for CubeSats, various ADCS components including some basic software has been developed and used in many CubeSat projects such as QB50 [8]. Examples include "CubeADCS [9]" from "CubeSatShop," "SatBus CR [10]" from "nano avionics," or ADCS from "CubeSpace [11]." These components are considered "plug-and-play" and provide a certain level of attitude determination and control functions. This is one reason why many novice developers are challenged by three axis stabilization. The records on how such satellites behaved in space have not been published in literature thoroughly as developers do not want to make bad results open to the public.

A reliable and competent satellite cannot be built only by buying

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https://doi.org/10.1016/j.actaastro.2018.02.006
Received 27 May 2017; Received in revised form 15 December 2017; Accepted 4 February 2018
Available online 10 February 2018
0094-5765/

List of abbreviations

ADCS	Attitude Determination and Control System
AOBC	Attitude Control Onboard Computer
AOS	Acquisition of Signal
C&DH	Command & Data Handling
EPS	Electric Power System
FOG	Fiber Optical Gyroscope
GAS	Geomagnetic Aspect Sensor (i.e., Magnetic Sensor or Magnetometer)
GSD	Ground Sample Distance
HCAM	High Resolution Camera
HILS	Hardware In the Loop Simulator
H/K	House Keeping
IGRF	International Geomagnetic Reference Field
IRU	Inertial Reference Unit
LCAM	Low Resolution Camera
Li-Ion	Lithium Ion
LOS	Loss of Signal
LTAN	Local Time of Ascending Node
MCAM	Middle Resolution Camera
MOBC	Main Onboard Computer
MTQ	Magnetic Torquer
OBC	Onboard Computer
PCU	Power Control Unit
RCS	Reaction Control System
RW	Reaction Wheel
SAP	Solar Array Paddle
SAS	Sun Aspect Sensor (i.e., Sun Sensor)
SOI-SOC	Silicon On Insulator - System On Chip
SSO	Sun Synchronous Orbit
STT	Star Tracker (i.e., Star Sensor)
TLE	Two Line Element
UVC	Under Voltage Control

components and connecting them. For example, satellite developers should carefully design the attitude "mode sequence," that is, how the satellite can transit from initial tumbling mode to the final three axis stabilized mode in a safe way. In addition, "safe mode" and the transition sequence to safe mode should be properly designed to enable survival in various unexpected situations, even when the satellite cannot be contacted from ground.

This paper discusses the various considerations that should be made in order to design ADCS for university satellites, particularly considering satellite survivability. The discussions are based on our experiences to design and operate Hodoyoshi-3 and 4 satellites which were launched together in 2014 and have been successfully operated in orbit in three axis stabilization mode for more than one year. During in-orbit operations, several components' experienced temporal and permanent failures as well as other anomalies, which gave us much experience in tackling such anomalies, and the lessons learned taught us the importance of total system design combining the satellite subsystems and its ADCS. The objective of this paper is to share our experience with satellite developers, especially university teams who will start satellite development.

The discussion in this paper assumes the following types of satellites as targets.

1) Satellites are Hodoyoshi-3 and 4 class (50 kg) satellites which have three axis attitude control functions. Nano and pico-satellites which have similar functions can be included.
2) The attitude stability, determination, and control accuracy is for the level of remote sensing satellites with 3–30 m Ground Sample Distance (GSD) which requires Reaction Wheels (RWs) for precise control.
3) The satellite is in low Earth circular orbit with 500 km–800 km altitude.
4) Satellites have "timeline command" function, in which commands can be executed at the defined timing even when the satellite is not in direct contact with a ground stations.

Section 2 shows the overview of Hodoyoshi 3 and 4 satellites. Section 3 describes several considerations that should be made for three axis stabilization, and Section 4 discusses control system design strategy considering satellite survivability. Some of the lessons learned from our experiences through the operations of Hodoyoshi 3 and 4 are described in Sections 5 and 6, including additional ideas to improve the flexibility of ADCS.

2. Overview of Hodoyoshi-3 and 4 satellites

Hodoyosh-3 and 4 (Fig. 1 and Table 1) were developed by University of Tokyo together with several Japanese universities and small companies in the "Hodoyoshi Project" (2010–2014). The Hodoyoshi Project ("Hodoyoshi" stands for "just good") has been led by University of Tokyo and funded by Cabinet Office of Japan. The project aims to develop technologies and infrastructure for micro-satellite and to seek innovative utilizations [12]. Hodoyoshi-3 and 4's primary missions include Earth observation with 40 m and 240 m GSD (Hodoyoshi-3) and 6.3 m GSD (Hodoyoshi-4) optical cameras. New components developed through the Hodoyoshi Project were implemented for space demonstration, including a Silicon On Insulator, System On Chip (SOI-SOC) radiation hardened onboard computer, X-band transmitter with maximum 500 Mbps speed, reaction wheels, and ion thruster. "Store and Forward (low-power RF signal collection)" experiment and "hosted payload" business experiment were also tried as additional missions. At the beginning of the Hodoyoshi Project, the University of Tokyo had already developed and launched three satellites, including the world's first CubeSat "XI-IV" in 2003, "XI-V" in 2005, and 30 m GSD remote sensing satellite "PRISM" in 2009, but Hodoyoshi-3 and 4 were the first satellites for the University of Tokyo to attempt three axis stabilization using a full set of attitude sensors and actuators, including gyros, magnetic sensors, sun sensors, magnetic torquer, and reaction wheels. Fig. 1 shows the photos of Hodoyoshi-3 and 4, and some specifications of ADCS. Please take note that the moment of inertia matrix is almost diagonal by well-considered component placement.

The various specifications and attitude control requirements are summarized in Tables 1 and 2. These two satellites were launched by Dnepr launch vehicle on June 19 (UTC), 2014 from Yasny launch base, Russia. Though several anomalies were experienced, three axis stabilization was successfully achieved, and all the planned missions could be carried out. Fig. 2 shows an example of obtained images with 6.3 m resolution by Hodoyoshi-4.

3. Prerequisites of three axis stabilization

3.1. Required functions to realize three axis stabilization

The following requirements should be satisfied in order to realize three axis stabilization; if only one of these requirements is not satisfied, the three axis stabilization will not be realized.

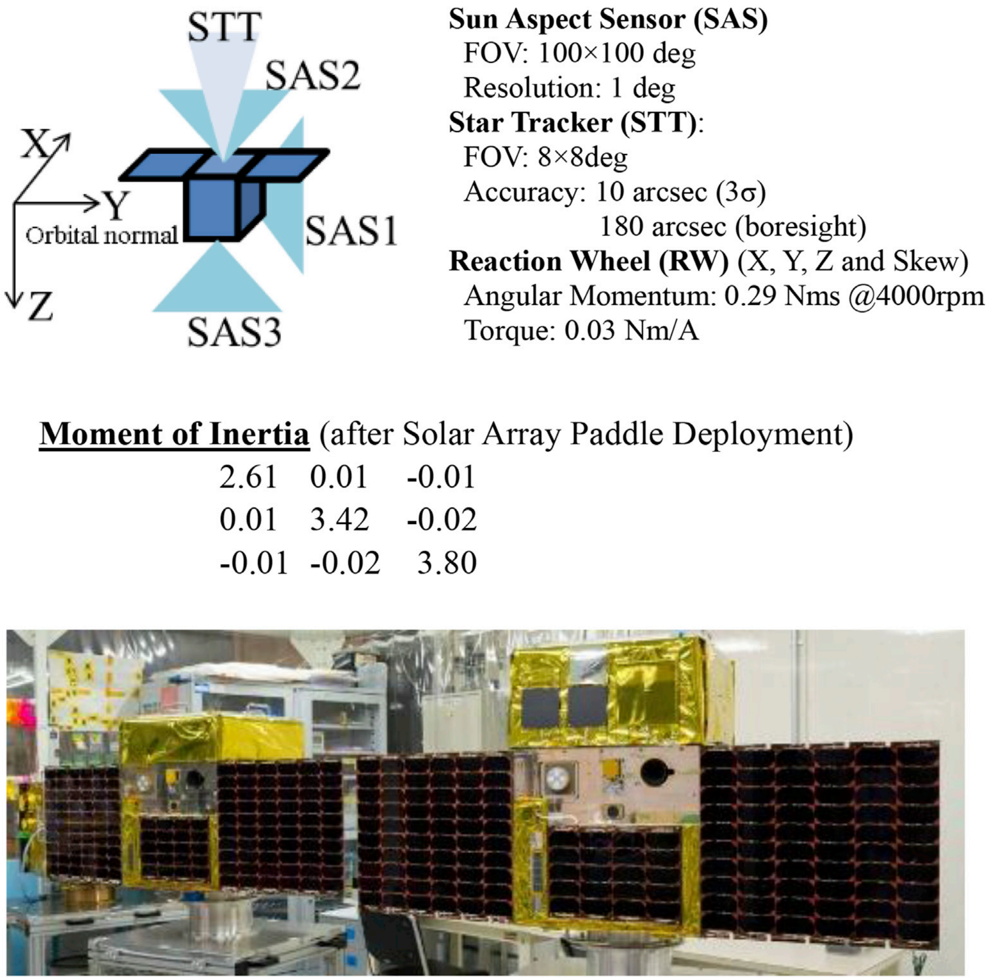

Fig. 1. Overview of Hodoyoshi-3 (left) and 4 (right) and its specifications related to ADCS.

Table 1
Specifications of Hodoyoshi-3 and 4 [13].

	Hodoyoshi-3 (H3) (2014-033-F) [40015]	Hodoyoshi-4 (H4) (2014-033-B) [40011]
Size	$0.5 \times 0.5 \times 0.7$ m	$0.5 \times 0.6 \times 0.8$ m
Mass	56 kg (including H_2O_2)	64 kg (including Xe gas)
Orbit	SSO: 612 × 665 km e = 0.0037, i = 97.97deg LTAN 10:30 a.m.	SSO: 612 × 650 km e = 0.0027, i = 97.97deg LTAN 10:30 a.m.
Life time	3 years (planned)	
EPS	Generation: max 130 W (GaInP2/GaGa/Ge SolarCell) Power Consumption: average 40–50 W, max 70 W Battery: 5.8 Ah, Nominal 28 V (24–32 V) (Li-Ion Battery)	
Attitude Modes	Initial, Spin-sun, 3 axis Sun pointing, 3 axis Earth pointing Actuators: RW, MTQs Sensor: FOG, STT, SAS, GAS Position is estimated by GPS.	
C&DH	OBC: SOI-SOC On-board computers (OBCs) for Main Data Handing (MOBC) and Attitude Control (AOBC) Software: Developed in HILS with “Hodoyoshi-Satellite Tool Kit”	

3.1.1. *Proper operation of attitude sensors*

Three axis attitude information obtained by a minimum of two reference vectors are required to obtain three-axes observability. Gyros will further improve the attitude estimation accuracy especially during a fast attitude maneuver and provide damping factor to attitude feedback control. Reference sensors include Geomagnetic Aspect Sensor (GAS), Star Tracker (STT), Sun Aspect Sensor (SAS), Earth Sensor and Gyros. GAS, SAS, or Earth sensors alone do not have three axis observability, and therefore a combination of them is required. Each sensor has some unobservable period. For example, SAS cannot be used during an eclipse period. Generally, gyro output is integrated to estimate the attitude continuously (this function is called “Inertial Reference Unit” or IRU), but the accumulated error due to gyro bias of IRU grows very rapidly over time, and this error is usually reduced occasionally using reference sensor outputs by way of a Kalman Filter.

3.1.2. *Proper operation of attitude actuators*

A Magnetic Torquer (MTQ) and Reaction Wheel (RW) are a popular

Table 2
Attitude control requirements by cameras and transmitter [14].

			Stability [deg/s]	Accuracy [deg]	
				Pointing	Determination
HODOYOSHI	3	Middle-Resolution Camera (MCAM)	0.4	0.8	0.04
		Low-Resolution Camera (LCAM)	2.0	4.0	0.2
		Small Camera	0.5	10	
	4	High-Resolution Camera (HCAM)	Pitch, Roll: 0.08 Yaw: 0.8	Pitch, Roll: 0.2 Yaw: 2	Pitch, Roll: 0.0048 Yaw: 0.048
		High-rate X-band Transmitter		Ground Station Tracking < 10	

combination as the actuator. MTQ cannot provide sufficient and precise torque, and RW alone cannot deal with its saturation problem. A Reaction Control System (RCS) with thrusters is sometimes used instead of a MTQ, but as they require complicated systems and fuel, RCS has rarely been used for university-level satellites.

3.1.3. Proper operation of software

Guidance, navigation and control functions require sophisticated software. In particular, the Kalman Filter, sometimes called a "Hybrid Navigation System," combines gyro based attitude calculation (IRU) and reference sensor based attitude measurements and is frequently used to track quick attitude motion as well as to prevent accumulation of errors over a long period.

3.1.4. Proper provision of electric power

Usually RWs require a large amount of electric power, and even when no torque is generated, RWs require certain power levels even at low speed. The satellite typically needs to provide significant electric power when three axis control is performed, such as during the mission phase. As the attitude in mission phase is frequently different from an attitude that would generate enough solar power, and as the mission sometimes needs to be carried out during an eclipse period (i.e., night time), proper functioning of Electric Power System (EPS) to provide enough power is indispensable, and power balance should be carefully considered.

Fig. 2. 6.3 m GSD image captured by Hodoyoshi-4.

3.2. Possible anomalies and consequences

In designing ADCS, possible anomalies which have an impact on the realization of the functions described in section 3.1 should be predicted, and special analysis on their influences to the survivability of the satellite should be carried out. Anomalies experienced in Hodoyoshi-3 and 4 are good examples for university-level satellites, which are summarized as follows.

3.2.1. Gyro

In Hodoyoshi-3 and 4, gyros sometimes could not be powered on by one uplink command and only after several trials it was powered on. Certain gyros showed unexpected behavior such that the output was stuck at a constant value, which jeopardized these satellites as discussed in Section 5. After around one year of operation, the gyros failed because of radiation damage. Gyro anomalies caused significant errors on attitude determination as well as on provision of "damping factor" for attitude stabilization. Some methods have been proposed which combines sun and magnetic sensors to estimate angular velocity [16], but this can only be done during day time. Hodoyoshi-3 and 4 did not employ this method.

3.2.2. Star sensor (STT)

In Hodoyoshi-3 and 4, STT was not used in-orbit as the STT parameters were not well tuned before launch. The engineer who was in charge of STT was an inexperienced person, and no one checked the appropriateness of his tuning. Unfortunately, these parameters were not included in the list of parameters which can be tuned by uplink command. This contains two lessons: have work checked by multiple people and make all tuning parameters available by uplink command. Therefore, during the eclipse period, only IRU and GAS could provide attitude information. However, in Hodoyoshi-3 and 4, as discussed later in 4.5.2, GAS was only used together with SAS, which means that IRU was the only source of attitude information during the eclipse period. The STT anomaly had a big impact on the attitude determination accuracy as well as gyro bias estimation accuracy. In the case of Hodoyoshi 4, during the first year's operations when the gyros are working well, attitude integration was carried out with high accuracy even during the eclipse period. After a gyro failed, attitude reference could not be determined during the eclipse period, and therefore the satellites were kept in safe mode most of the time to ensure their survival.

3.2.3. Sun Aspect Sensor (SAS)

In Hodoyoshi-3 and 4, SAS had two problems. First, it recognized the Earth albedo as direct sun light. Secondly, it suffered from "startup" problems; that is, a specific SAS would not power on if a different power-on sequence was used (compared to the ground tests). In such cases, we would power off all SASs and follow the same power-on sequence used in the ground tests. The former problem caused serious problems for Hodoyoshi-3 and 4 in the initial phase as described in Section 5. The latter anomaly was a big problem because we wanted to power on and off the specific SAS flexibly to avoid albedo misrecognition. This "power on sequence problem" was solved by changing the covariance value of measurement errors for SAS to be used in the Kalman Filter; that is, if a certain element of covariance matrix of the measurement errors of SAS is set as a very large value, the corresponding SAS does not contribute to the attitude update, which has the same effect as "turning-off" this SAS.

In Hodoyishi-3 and 4 the following anomalies described in sections 3.2.4 through 3.2.8 did not occur, but as there is potential for such anomalies, their effects are discussed as well.

3.2.4. Geomagnetic Aspect Sensor (GAS)

GAS is not only an attitude reference sensor but also frequently used for MTQ based control algorithms such as b-dot law [14] or cross-product law, and therefore a GAS anomaly has severe effects in many ways. Although the geomagnetic flux vector with respect to the satellite body can also be estimated by the International Geomagnetic Reference Field

(IGRF) model and satellite position/attitude information, this is only possible if position/attitude information is sufficiently accurate, which is not always the case. For example, during the initial de-tumbling control phase by b-dot law, TLE may provide position information which can be used to estimate the geomagnetic flux vector in the inertial frame. However, if the satellite does not have attitude information with respect to the inertial frame, the geomagnetic flux vector with respect to the satellite body (required for b-dot law) cannot be obtained.

3.2.5. GPS receiver

The GPS receiver is used to obtain the current position with respect to the Earth, which is required to calculate satellite attitude in inertial frame from GAS and to set target attitude in inertial frame for nadir or target pointing. The GPS receiver also provides the precise onboard clock. The failure of GPS receiver thus has impact on these functions. Orbit parameters such as TLE and OBC clock correction information can be uplinked, solving this problem to some extent, but the accuracy will be degraded from an order of several tens of meters to kilometers in position accuracy and from an order of milliseconds to seconds in time accuracy.

3.2.6. Reaction wheel (RW)

To prevent the zero-crossing problem (for detail, see 4.2) during the operation and to provide redundancy, Hodoyoshi-3 and 4 had four RWs (3 axis +1 skew). Therefore, even if one RW fails, control performance will not be degraded too much. An important consideration related to RWs is the total angular momentum kept within the satellite, which has large effect of attitude motion during safe mode. The capability to flexibly design this total angular momentum will be degraded by one RW failure, as discussed in 4.5.1.

3.2.7. Magnetic torquer (MTQ)

MTQ is frequently used for the initial attitude control phase, such as de-tumbling or Sun acquisition phase, and therefore anomaly of MTQ has severe damage to these mode transitions. Moreover, as MTQ is usually the only source of momentum unloading of RWs for satellites without thrusters, MTQ failure will make this unloading impossible and then the saturated RWs cannot absorb any more external disturbance torques. On the other hand, only one axis failure of MTQ can be endured to some extent as discussed in Miyata [15]. Fortunately, MTQ does not have any moving parts and is usually very robust.

3.2.8. Battery and power control unit (PCU)

If the battery voltage becomes lower than a certain threshold, it is advisable that the satellite moves to “safe mode,” in which battery is charged as quickly as possible by securing power generation while reducing the power consumption. The design of this “safe mode” and the way to transit to this mode is very important; deep understanding of the total satellite system is essential. In Hodoyoshi-3 and 4, three levels of software “Under Voltage Control” (UVC) are prepared; OBC checks the battery voltage and automatically transits to safe mode of three levels corresponding to the three thresholds of the battery voltages. The three levels are; (1) Mission components are turned-off, (2) Only OBC, RF transmitter and basic attitude sensors are powered on, and (3) Only OBC and transmitter are powered on. In either case, “spin-sun” control described in 4.4 is automatically triggered. If the voltage continues to decrease, OBC is automatically turned off and sleeps until a certain voltage is recovered, which is called “Hardware UVC.” In Hodoyoshi-3 and 4, these software and hardware UVC strategies worked well to survive in various severe situations until the crisis discussed in Section 5.

When designing this safe mode, an important consideration from the perspective of attitude control is that at the time when RWs are switched off to transit to safe mode, all of the internal angular momentum of the RWs becomes the angular momentum of the satellite body, which will start rotating with this angular momentum. Taking this into account, the total angular momentum of RWs during the three axis stabilization operations such as during the mission phase should be carefully designed. This issue will be discussed in detail in Section 4.

4. System and ADCS design considering survivability

The following considerations for survivability should be made in designing the satellite architecture and its ADCS.

4.1. Securing solar power generation

The top priority for survivability is that enough solar power is generated at a survivable level in all situations. The following categorizations would be adequate for further discussions.

1) If all the satellite surfaces have enough solar cells

If situation allows, this is the best solution; that is, the survivability does not depend on the satellite attitude [13]. This strategy is often used for 1U-3U CubeSat such as University of Tokyo's “XI-IV” which does not have enough attitude control capabilities. In such cases, the potential risks caused by ADCS need not be especially considered.

2) If almost all the surfaces have solar cells but some don't or some surfaces' solar cells are not enough for the satellite to survive.

In this case, avoid a situation where the satellite attitude is fixed in an inertial coordinate frame where the normal direction of the surface which has no or insufficient number of solar cells is towards the Sun. As there are various disturbance torques in low altitude, if the total angular momentum of the satellite is sufficiently small, the attitude will not be fixed in the inertial frame, and the above problem will not occur. The most dangerous situation is when the satellite is spinning around a certain axis such that it keeps no or insufficient solar cell areas directed towards the Sun (left figure in Fig. 3).

If energy dissipations occur, the spinning satellite will eventually spin around the maximum moment of inertia axis. Therefore, the maximum moment of inertia axis should not be perpendicular to this insufficient solar cell area in order to avoid pointing this area towards the Sun.

3) If all the solar cells are on a large solar paddle

This is not a good strategy in terms of satellite survivability, but sometimes this configuration is required for other reasons. If this is the case, then it is advisable to direct the maximum moment of inertia axis perpendicular to this solar paddle and to prepare a simple and reliable “sun acquisition” (realizing an attitude such that the sun is shining on this solar paddle) control algorithm using only primitive attitude control components such as SAS, GAS and MTQ, not using RWs. Once this attitude is acquired (right side figure in Fig. 3), the maximum solar power generation can be kept without control, during which components checkout or preliminary mission operation can be carried out.

4.2. Discussion of total angular momentum

During the mission phase when the satellite is three axis stabilized,

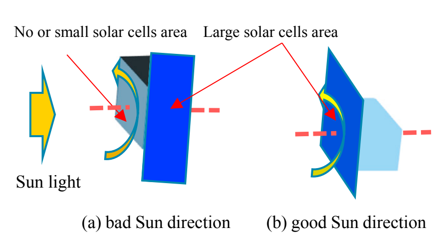

Fig. 3. Relationships of spin axis, solar cell attached surfaces and Sun direction.

there are several choices regarding the total angular momentum of the satellite including internal angular momentum by RWs. The most important consideration is that when a certain anomaly occurs, RWs may be powered off to transit to safe mode, but the total angular momentum is kept during this transition, which will determine the satellite attitude motion in safe mode.

In the case 3) or even in case 2) in Section 4.1, where solar cells are almost attached on one surface of the satellite, the following strategy is advisable in three axis stabilization phases to assure the maximum survivability of satellites during safe mode.

4.2.1. *Strategy A (bias momentum)*

a) In this strategy, the rotation of RWs are to be controlled such that the total angular momentum is along the line connecting the Sun and the satellite (called "Sun-satellite line" hereafter).
b) When it transits to safe mode and RWs are powered off, the satellite will start tumbling while keeping its angular momentum. This tumbling will eventually converge to one axis rotation around the maximum moment of inertia axis. As the total angular momentum is kept for this transition, this final rotational axis is the same as the initial total angular momentum vector realized in a).
c) If the satellite is designed such that its maximum moment of inertia axis is perpendicular to the solar paddle, then the final one axis rotation will keep the perpendicular direction of the solar paddle along the initial total angular momentum vector which is the Sun-satellite line. As a result, the final attitude will be either of the two figures in Fig. 4.
d) If the left side attitude in Fig. 4 is realized, the satellite can keep the maximum sun angle on the solar paddle in safe mode. Of course, some nutation may occur, but if these are not so large or suppressed by energy dissipation, this attitude is very safe in terms of power balance.
e) The only concern is that the satellite may transit to the right side attitude in Fig. 4. This situation can be avoided by carefully finding the initial total angular momentum in a). Further study is required to find out the conditions to avoid this worst attitude.

If in the above strategy (ii) in Fig. 4 may occur with a certain probability, then another choice in step a) is to design the total angular momentum vector to be perpendicular to the Sun-satellite line as in Fig. 5, and design the maximum moment of inertia axis to be parallel to the solar paddle surface. In this strategy, the solar paddle can get sunlight for some period during one rotation no matter which attitude may be taken in the final one axis rotation. If power balance is still acceptable in this strategy, this would be safer.

4.2.2. *Strategy B (zero momentum)*

The second case is that the satellite has solar cells on several surfaces, and the simulation study assures that it would be safe if the satellite tumbles randomly. Then it is advisable to keep the total angular momentum of the satellite including RWs as small as possible, with the following method.

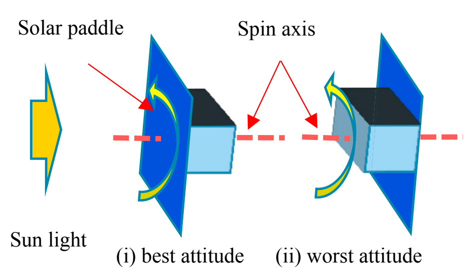

Fig. 4. Best (left) and worst (right) attitude of satellite, spinning around its maximum moment of inertia axis.

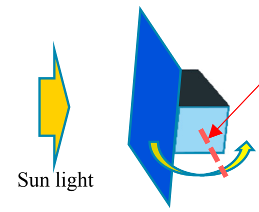

Fig. 5. Total angular momentum vector to be perpendicular to the Sun-satellite line.

a) During the mission phase when the satellite is three axis stabilized, total angular momentum of RWs is kept as small as possible using the following b) or c).
b) If four RWs (x, y, z, and skew), are used the nominal speed of the four RWs can be aligned such that the total angular momentum of the RWs is zero, while keeping each nominal speed at a certain non-zero level. If the deviation of the speed of RWs from this nominal speed becomes larger than a certain threshold, momentum unloading logic is triggered to suppress this deviation. When an attitude maneuver is being performed, this unloading logic is not triggered. There are two objectives to set such target nominal speeds. One reason to keep lower speeds for each RW is to avoid large power consumption and fatigue of RW bearings. The other reason is to keep speed of each RW at a non-zero level in order to avoid "zero-crossing" phenomenon. "Zero-crossing" is the problem that when the rotational speed of the RW crosses zero, the large and unsteady friction of the wheel makes a large deviation of the generated torque from the intended torque.
c) If only three RWs can be used, the target rotational speed of each RW is set at a low but non-zero level such as 100 rpm in order to keep low total angular momentum, while avoiding the zero-crossing phenomenon. If the actual speed deviates significantly from this target speed, then momentum unloading logic is triggered to keep the speed near this target speed, except when attitude maneuver is being performed.

In Hodoyoshi-3 and 4, case b) of Strategy B was adopted as four wheels can be used during all of the operation period.

The momentum unloading logic in case b) is as follows; If the speed of a certain RW deviates from its nominal speed, MTQ is activated with the following cross product law;

$$\boldsymbol{M_c} = \boldsymbol{M_p}\,\mathrm{sign}(\boldsymbol{B} \times \Delta\boldsymbol{L}) \tag{1}$$

Here, $\boldsymbol{M_c}$ is the magnetic moment vector to generate, $\boldsymbol{M_p}$ is the maximum magnetic moment vector which MTQ can generate, $\Delta\boldsymbol{L}$ is the target angular momentum change to make as;

$$\begin{aligned}\Delta\boldsymbol{L} &= -\boldsymbol{I}_{RW}\boldsymbol{\omega}_{RW} - I_{skew}\omega_{skew}[\cos\alpha\cos\beta \quad \sin\alpha\cos\beta \quad \sin\beta]^{\mathrm{T}} \\ &-\boldsymbol{I\omega}\end{aligned} \tag{2}$$

Here, $\boldsymbol{I}_{\mathrm{RW}}$ is the inertia matrix, and $\boldsymbol{\omega}_{RW}$ is the rotational rate of the non-skew axis RWs. I_{skew} and ω_{skew} are the moment of inertia and rotational rate, respectively, of the skew axis RW. α is the offset angle between the x-y plane and skew RW axis. β is the offset angle between the z-axis and skew RW axis. $\boldsymbol{I}$ and $\boldsymbol{\omega}$ are the moment of inertia and angular rate of the satellite. Actually, equation (2) corresponds to the minus of the current total angular momentum of the satellite and RWs, and the control law (1) tries to negate the total angular momentum. This momentum unloading is not triggered when the satellite is performing a large angle attitude maneuver or precise attitude control for mission such as taking a picture of the Earth.

Fig. 6 shows the total angular momentum including both the satellite body and internal RWs, and the rotational rate of each of four RWs. As in the figure below, the nominal speed of the skew RW is set at around 610 rpm, and the other three diagonal wheels have a nominal speed of 352 rpm (610 divided by square root of 3), and their speeds fluctuate around this rate. In Fig. 6, the momentum unloading logic is activated

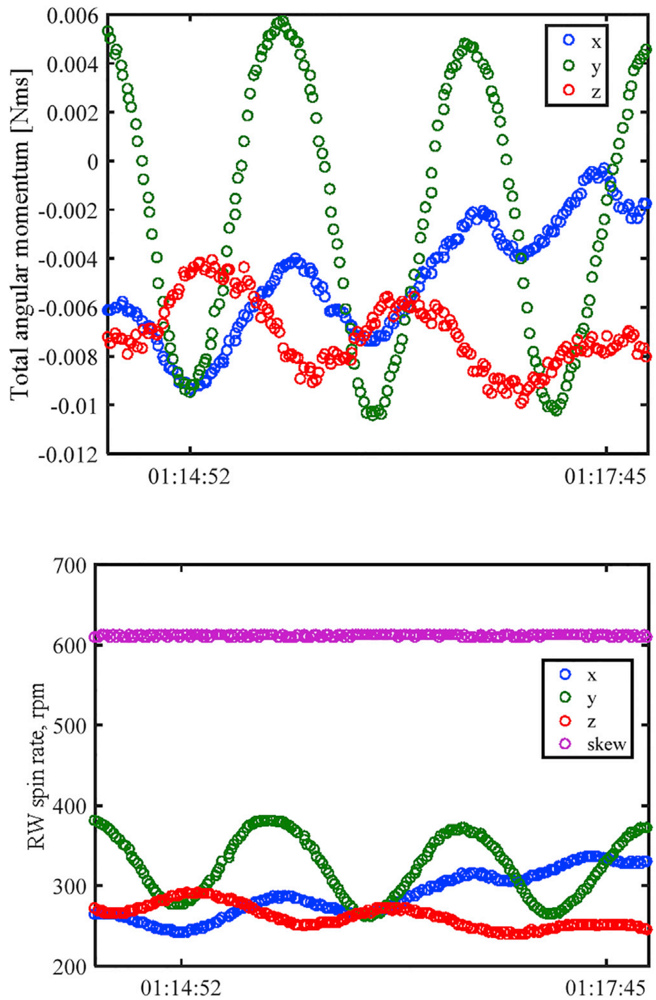

Fig. 6. Total angular momentum during three axis stabilization phase and the trajectory of rotational rate of four RWs (Hodoyoshi-4 case, the speed of skew RW is opposite direction of the other RWs).

and therefore the deviation from the nominal speed of RWs does not become large. Momentum unloading logic is deactivated during large angle maneuvers, but if the RW rate gets larger than 4000 rpm, which is near the speed limit of RWs, automatic unloading is activated. The figures show that the total angular momentum is kept at a very small value while avoiding “zero crossing” of each RW.

4.3. *Considerations for mode transitions*

The definition of modes and design of proper transitions between modes are very important in ADCS design. In designing modes and their transitions such as in Fig. 7, the following factors should be considered.

a) Initial mode is the situation just after release from the rocket. The satellite is usually tumbling with an unpredictable rate and attitude. Usually launch providers will provide the information of the maximum satellite tumbling rate after release, which is sometimes called the “tip-off rate.”
b) It is usually dangerous to transit directly from this initial mode to the final operation mode (“mission mode”) because before doing so the components required for three axis stabilization should be checked out. It is advisable to carry out these checks in such a mode that the satellite can get enough solar power, which is usually called “Sun acquisition mode.” If this mode is well designed so that the maximum solar power generation is assured without control (passive type) or with simple and reliable control logic (active type), then satellite operators, once this mode is achieved, can carry out checks of components, various experiments including attitude determination and control or even preliminary trials of the satellite missions without much concern about the power balance.
c) “Mission standby mode” is usually defined as a nominal attitude for Earth observation satellites; this mode should be designed so that positive power balance can be assured and that the transition from this to mission mode can be done quickly. This mode is required in cases where the power balance in mission mode is negative and therefore a certain “way-point” is required between Sun acquisition mode and mission mode.
d) “Safe mode” should be carefully designed as the last resort for a satellite which has critical problems. This mode should be so designed that transitions from any mode to this mode are safely and quickly

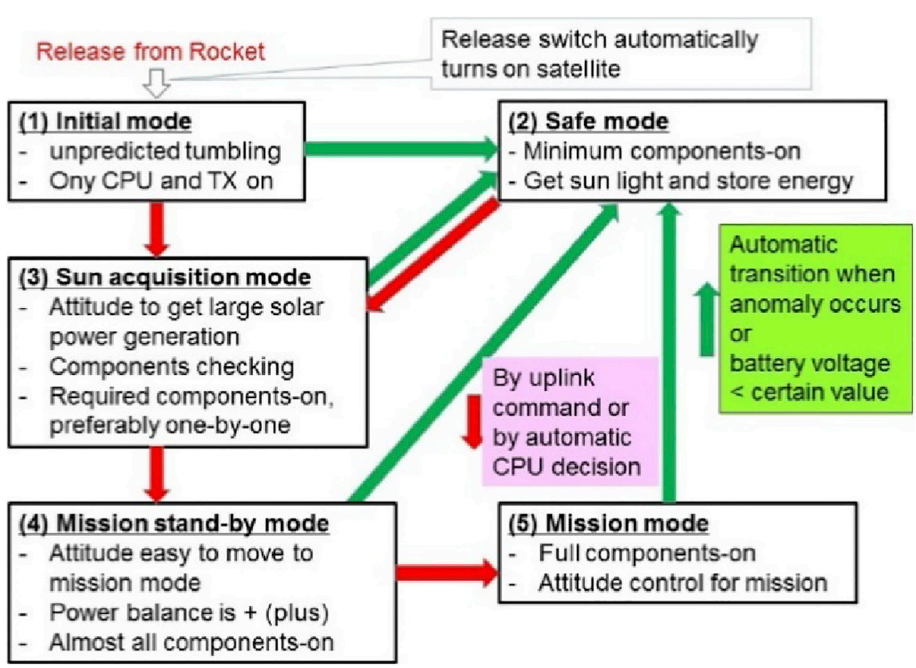

Fig. 7. Typical modes and mode transition diagram.

carried out even when the ground stations cannot communicate with the satellite, and therefore this transition should be made by onboard autonomous function. Also, it is advisable to prepare an uplink command to manually trigger this transition. The most important requirement is that during this mode the power balance should be positive so that the satellite can survive for long time to wait for the ground support. In order to realize that, the active, powered components and the total angular momentum during mission standby and mission modes, as discussed in 4.2, should be carefully designed. In the Hodoyoshi-3 and 4 cases, in the safe mode, some components are turned off and “spin-Sun” control as described in 4.4 is automatically triggered.

e) Transitions between modes can be manual (by uplink commands) or automatic (by OBC software functions or by hardware). In either case, the criterion to trigger each transition should be well considered and carefully designed. Especially for automatic transitions, special care should be taken so that the satellite can surely survive during and after the transitions.

In Fig. 7, green arrows show automatic transition to “safe mode” in such cases that some anomalies (such as malfunction of components or low voltage of battery) occur. This transition should occur at any time without time delay by OBC software if OBC is active, or by certain hardware mechanisms, if the battery voltage is too low to keep OBC active. Red arrows can be triggered by OBC software when certain criterion are satisfied, or by uplink commands.

4.4. Designing sun acquisition mode

In this mode, the satellite keeps its attitude so that a large enough amount of solar power can be generated. It is advisable that this mode can be achieved and kept without control or with simple control with minimum sensors and actuators, not using RWs or Kalman Filter.

Considering these requirements, the following attitude control strategies are promising to achieve and keep this mode.

4.4.1. Case 1) maximum moment of inertia axis is perpendicular to the surface which has the largest solar cell area

In this case, spinning around this axis is stable in the long run, and if this axis is directed towards the Sun, then the satellite can maintain this good attitude to obtain large solar power generation without control such as in left figure of Fig. 4. Hodoyoshi-3 and 4 adopted this strategy named “spin-sun” control, and this was achieved by a series of two control laws (3) and (4) using SAS, GAS and MTQ;

i) Spinning-up the satellite by cross product law;

$$\boldsymbol{M_c} = \boldsymbol{M_p}\,\mathrm{sign}(\boldsymbol{B} \times \Delta\boldsymbol{\omega}) \tag{3}$$

ii) Changing the Spin axis towards the sun;

$$\boldsymbol{M_c} = \boldsymbol{M_p}\,\mathrm{sign}\left(\boldsymbol{B} \times [s_x \quad s_y \quad 0]^T\right) \tag{4}$$

Here, $\boldsymbol{M_c}$ is the magnetic moment vector to generate, $\boldsymbol{M_p}$ is the maximum magnetic moment vector which MTQ can generate, $\Delta\omega$ is the desired change of angular velocity vector and $\boldsymbol{s} = [s_x, s_y, s_z]^T$ is the sun direction vector. All the vectors in the equations above should be represented with respect to the body frame of the satellite, and it is assumed that the desired direction of the sun with respect to the body frame is -z, i.e., $[0, 0, -1]^T$ direction. As to the definition of the body frame, please see Fig. 1.

First, the spin rate of the satellite is checked and the satellite starts spinning-up if the rates are not sufficient by formula (3). After reaching appropriate spin, the satellite performs the next control of changing the spin axis to the sun by (4). During the second control, if the spin rate becomes below acceptable value, spinning-up is carried out again by (3), and the sequence is repeated until the spin axis becomes within acceptable deviation from the sun direction and the spin rate is within a certain range. In order to get enough stability of the spin axis, 3 deg/s rotation along the z-axis (the maximum moment of inertia axis as shown in Fig. 1) is targeted in Hodoyoshi-3 and 4. This control is completed when the –z direction becomes within a 5 deg deviation from the sun direction.

The important consideration is that this control should not be started if the battery voltage is low, as this transition is sometimes dangerous as the additional power required to operate SAS, GAS and MTQ may be more than the generated solar power during this transition. That is, there may be a “Death Valley” of battery voltage during this transition. In Hodoyoshi development phase, Monte Carlo simulations were carried out to check whether the satellite can survive this Death Valley from varied initial conditions. (Similar analysis was carried out in Ref. [17].). The initial maximum tumbling rate just after release from the rocket was predicted to be 15 deg/s based on the data from the launch provider, and initial attitude is unknown. The mode transition from initial mode to this mode is triggered by an uplink command, and therefore the question is “in what condition the transition to this mode should be triggered?”

Fig. 8 shows the maximum battery usage during this transition as a function of the initial angular rates. In simulations, all the predicted disturbances in-orbit including atmospheric drag, solar radiation pressure, magnetic, and gravity gradient disturbances are included. Various initial conditions concerning initial attitude with respect to the Sun as well as initial angular velocity (less than 15 deg/sec) are randomly generated. Based on this figure, in most cases the amount of battery energy that would be consumed during this transition can be determined. From this figure, the acceptable battery energy consumption during this transition was decided to be 100 Wh, which means that the success rate to achieve the sun acquisition mode within this energy usage is 85% for the Hodoyoshi-3 (H3) and 87% for Hodoyoshi-4 (H4) respectively. This is not perfect (100%) but good enough, as safe mode is prepared as the last resort if the consumption gets larger. If the satellite transits to safe mode because of low battery voltage, then the satellite will store additional solar power in safe mode and after that it will try the transition to sun acquisition mode again. As 100 Wh was decided to be the nominal energy consumption during this transition, a battery voltage above 27 V was decided to be the threshold whether to start the transition to sun acquisition mode, assuring that the battery Depth of Discharge (DOD) will not be larger than 40% even if 100 Wh energy is used during this transition.

However in the real in-orbit operation of Hodoyoshi-3 and 4, both the satellites could not survive this “Death Valley” because of a component anomaly during this phase and almost died, which will be described in

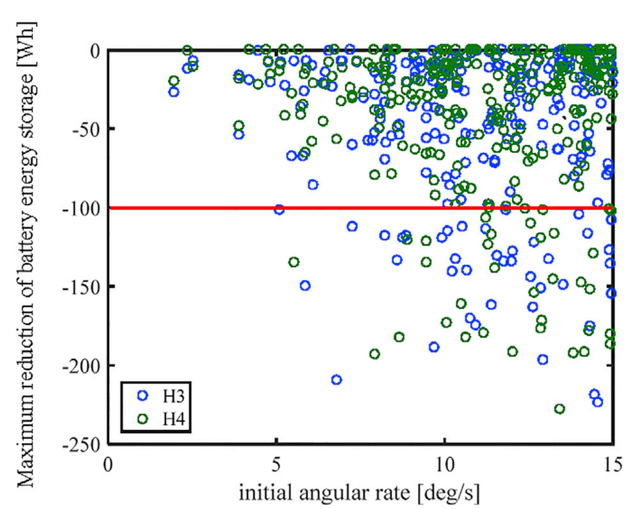

Fig. 8. Maximum reduction of battery energy storage during spin-sun control (Monte Carlo simulations with 270 trials).

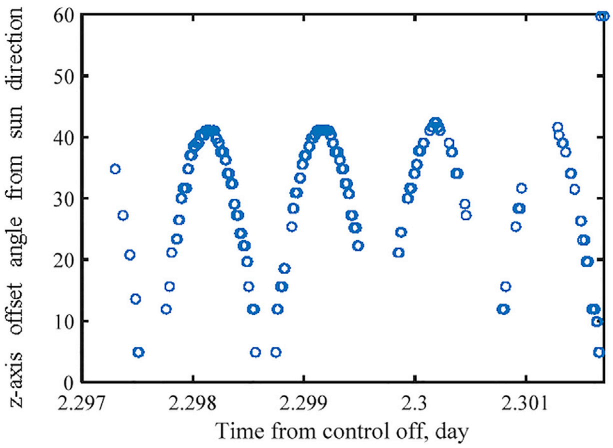

Fig. 9. z-axis deviation from Sun direction after achieving spin-sun mode without control (in-orbit result).

more detail later in Section 5.

Once this sun acquisition mode is achieved, as in Fig. 9, the z-axis did not deviate more than 40° from the Sun direction even without control, which means the satellite could get enough solar power on average. During this mode, check-out of various components as well as Kalman Filter and control algorithms could be carried out without much concern for the satellite attitude. Besides, in Hodoyoshi-3 and 4, preliminary missions were also conducted, such as taking low resolution (such as 240 m or 40 m GSD) Earth pictures by field sensor cameras, a low-power RF signal collection experiment, and "rental space" (hosted payload) missions.

Hodoyoshi-3 and 4 both have only three sun sensors (SAS-1 to SAS-3) in Fig. 1. This is why spin-up was required as the first step in order to detect the Sun direction by only three sun sensors. If the satellite has 6 sun sensors on all the surfaces, the following control law is adequate.

$$\begin{aligned} \boldsymbol{T} &= k_{angle}(-\boldsymbol{Z}\times\boldsymbol{S_b}) - k_{vel}(\dot{\boldsymbol{S}}_{\boldsymbol{b}}\times\boldsymbol{S_b}) \\ &+k_{spin}\left(\boldsymbol{\omega_{ref}} - \frac{\dot{\boldsymbol{B}}_{\boldsymbol{b}}(\boldsymbol{B_b}-\boldsymbol{S_b})}{|\mathbf{B_b}\times\mathbf{S_b}|}\right)\boldsymbol{S_b} \end{aligned} \tag{5}$$

where $\boldsymbol{Z}$ is desired Sun direction, $\boldsymbol{S_b}$ is the measured Sun direction and $\boldsymbol{B_b}$ is the measured Geo-magnetic flux density in body frame, and $\boldsymbol{\omega_{ref}}$ is the target rotational rate around the Z axis. Gains "k_{angle}," "k_{vel}" and "k_{spin}" should be tuned through simulations. The output $\boldsymbol{T}$ is the desired torque in body frame, and "Cross Product Law" was used to provide such torque which is nearest to this $\boldsymbol{T}$ using three axis MTQ.

4.4.2. *Case 2) other scenarios*

As there is no method to keep a good attitude for solar power generation without control, the above control law (5) should be constantly activated with $\boldsymbol{Z}$ being the desired Sun direction with respect to the satellite body frame ($\boldsymbol{Z}$ does not have to be "z-axis" of the satellite body-frame). The appropriate value of ω_{ref}, the target angular velocity around Z-direction, should be searched for by simulations.

4.5. *Countermeasures against component anomalies*

During the satellite operation, various anomalies of components may occur as discussed in 3.2, but appropriately prepared countermeasures can minimize their effects. In this section, such countermeasures are proposed against possible anomalies of components, some of which actually occurred in Hodoyoshi-3 and 4. Generally speaking, the important lesson learned was that "freedom to change logic" onboard should be prepared in order to flexibly deal with such anomalies. Of course software upload would solve many algorithmic or parameter related problems, but it requires communications of large amount of data between the satellite and the ground and its confirmation which sometimes is difficult or even impossible because of instable satellite attitude or shortage of battery power. Therefore, simple and yet effective flexibility should be implemented onboard.

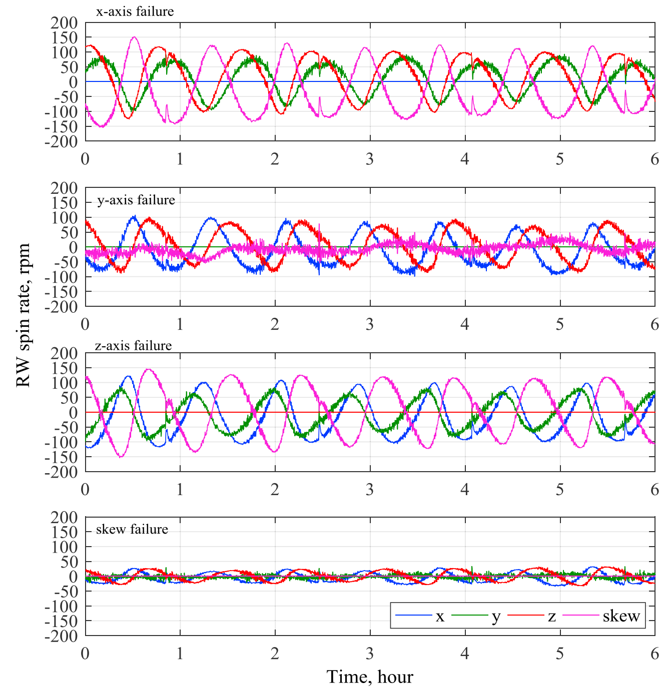

Fig. 10. Simulation of RWs behavior in one RW fail case (simulation results).

4.5.1. *Reaction wheels (RW)*

Even if one of four RWs fails, changes of torque allocation matrix can realize required torque easily, but the total angular momentum control becomes difficult. As discussed in 4.2, by appropriately setting the threshold to trigger momentum unloading logic, total angular momentum can be kept small to some extent. It is advisable that several sets of these threshold values should be stored in OBC memory to be switched corresponding to which RW fails, or that they are further tuned by uplink commands seeing their behaviors. In Hodoyoshi-3 and 4, such RW anomalies did not occur, though such thresholds was prepared. Fig. 10 shows the predicted behavior of RW in one RW failure case obtained by simulations.

The top figure of Fig. 10 shows the behavior of y, z-axis, and skew-RWs in the case that x-axis RW fails, and other figures show the similar behavior with one RW failure. It is observed that low angular momentum is kept but sometimes zero crossing phenomenon occurs which does not happen in the case all of the four RWs are active (Fig. 6). Keeping low total angular momentum and avoiding zero crossing phenomenon are contradicting requirements, for which proper trade-off is required.

If more than two RWs fails, ADCS becomes "under actuated system," but still three axis control is possible. Hayabusa-1 experienced this situation, but successfully carried out the planned mission [18].

4.5.2. *Attitude reference sensors (STT, GAS and SAS)*

In Hodoyoshi-3 and 4, STT could not be used as mentioned in 3.2.2. As a result, only SAS, GAS and gyro could be utilized as attitude sensors, and the following strategy using these sensors were taken, which worked very effectively.

During the day time, attitude reference is provided by the combination of SAS and GAS output, which has three axis attitude information. Gyro was used to integrate attitude quaternion as IRU, which is periodically updated by Kalman Filter at the timing when SAS and GAS measurement is provided. During the night (eclipse) time, only integration of

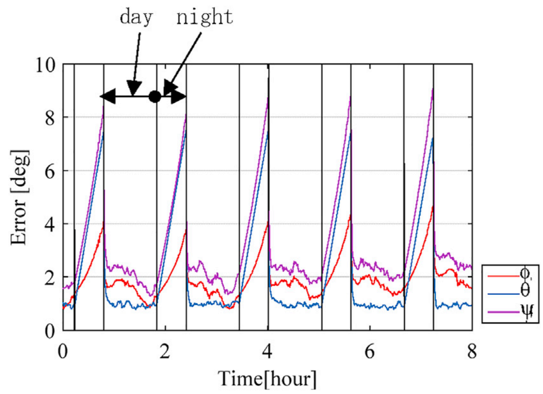

Fig. 11. Attitude estimation error during day and night time for Hodoyoshi-3 and 4 (simulation results).

quaternion by gyro output (IRU) was performed, and therefore the error usually grows rapidly. However, as the implemented gyro has a good bias stability, this bias could be estimated off line on the ground and uplinked to the satellite to be used for several days, and using such bias data, IRU only yields maximum 12° (in average, 8°) attitude estimation error at dawn (the end of night time). Fig. 11 shows the simulation results of attitude estimation error with the real setting of parameters, which almost coincides with in-orbit performance.

In Hodoyoshi-3 and 4 case, SAS and GAS information was utilized only when both of them can be used, that is, only during day time. But as Kalman Filter can interpolate such information having insufficient observability, it would be better to use GAS also in night time even if SAS cannot be used. GAS has only two degree of freedom observability instantaneously, but fortunately the direction of Geo-magnetic flux vector changes over time in inertial frame as the satellite flies around the Earth except for the case that the orbit's inclination is zero degree. Therefore, the continuous measurements of GAS will eventually provide three axis attitude observability. Fig. 12 shows the comparisons of attitude errors depending on GAS usage in night time. (A) is the same profile as in Fig. 11 (nominal case) and (B) is the case when the gyro bias stability gets worse. (C) and (D) show the case when GAS is used even in night time with good gyros (C) and bad ones (D) in IRU. In A), during night time, because of integration error of gyro measurements, the attitude estimation error quickly grows, whose tendency is further enhanced when using low specification gyros as shown in (B). If GAS is used during night time, this integration error is much suppressed as in (C), and even with low specification gyros, the error is suppressed to some extent as in

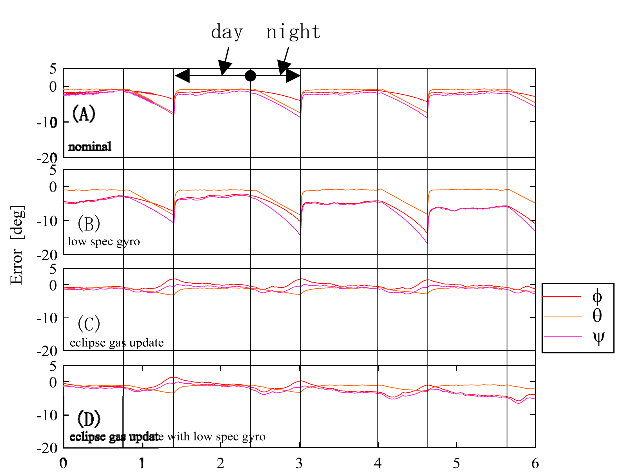

Fig. 12. Comparisons of attitude determination error in case GAS is used in night time.

(D). Soon after the dawn, which occurs every 96 min, the accuracy gets much improved because sun sensors are used to update the attitude information. It shows that the usage of GAS in night time can improve the long time accuracy of the attitude determination.

4.5.3. Gyroscope

Gyros are usually used in IRU for integration of attitude quaternion. When a gyro cannot be used, the integration of attitude quaternion should be made based on attitude dynamic equation such as "Euler equation of rotation motion" and all the torques exerted on the satellite should be input to this equation. This dynamic equation requires precise information on moment of inertia matrix, whose estimation errors quickly generate large attitude errors. Also it would be usually difficult to measure the external disturbance torques, whose estimation errors also generate errors in attitude estimation. These errors are usually very large as compared to the case when gyros can be used, but can be deleted to some extent by Kalman Filter with such reference sensors as SAS, GAS and/or STT.

If accurate moment of inertia matrix cannot be obtained or if disturbance torques are difficult to estimate, it is advisable to use only GAS + SAS or STT as the reference sensors to estimate attitude when such observations are made, without using integration of the attitude dynamic equation through interpolation. If angular rate information is required for damping terms of feedback control, then it can be generated by differentiating the two consecutive three axis attitude estimations obtained by these reference sensors, though it is less accurate than rate sensor measurements.

5. Other attitude anomalies experienced in Hodoyoshi-3 and 4

This section further describes actual anomalies of Hodoyoshi-3 and 4 caused by more than one consecutive failures. Because of such anomalies, these two satellites experienced very low battery voltage for about a week and almost died. This example shows how multiple causes of small anomalies yielded a critical situation, and how the problem could be solved.

About three weeks after launch, in early July 2014, both the satellites already established three axis stabilization and started mission operations. On one Friday night, we checked and confirmed that the satellite was properly working in three axis stabilization mode with its solar paddle directed towards the Sun, i.e., in "Mission Standby Mode," and we decided that we would not operate these satellites during the weekend. Therefore, we expected the satellite was keeping the attitude as in the left figure of Fig. 13 by autonomous control during the weekend. But on the contrary, in the Monday morning when we started the weekly operation, we were very surprised to see that Hodoyoshi-4 was fast spinning (about 10 deg/sec) with its solar paddle directed to the opposite direction of the Sun such as the right side figure of Fig. 13, and the battery voltage was extremely low. While we were analyzing the cause and discussing countermeasures, Hodoyoshi-3 fell into the same symptom in Tuesday with a spinning rate of about 3 deg/sec.

This is the worst attitude motion, as the satellite can receive only little solar power and this attitude is difficult to change because of the large total angular momentum of the satellite. As a result, the battery voltage actually became too low and "hardware UVC" already occurred and main OBC was powered on and off reflecting the fluctuations in battery voltage. If the situation cannot be recovered quickly, frequent low battery voltage (i.e., very high DOD) would damage the battery so that it would never be recovered.

By analysing the very small amount of stored data, as OBC already lost lots of stored H/K data by turning-off, we could estimate that the following scenario occurred for these satellites, as shown in Fig. 14.

1) When the satellite flew over the Antarctic Ocean where its large ice area provides strong albedo, SAS misrecognized this albedo as direct sun light (1 in Fig. 14)

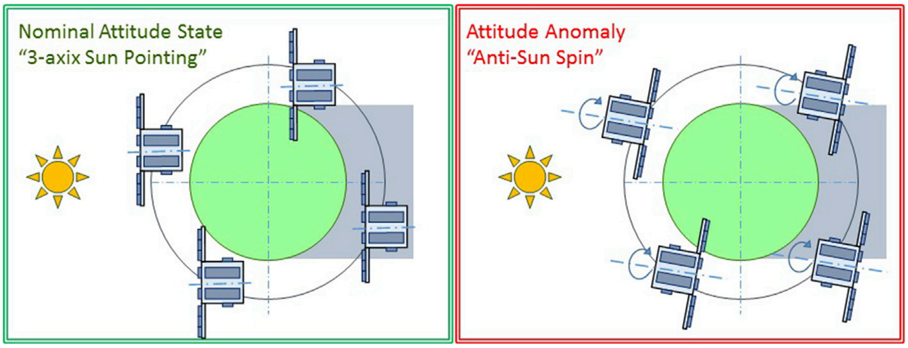

Fig. 13. Normal (left) and abnormal attitude of Hodoyoshi-3 and 4.

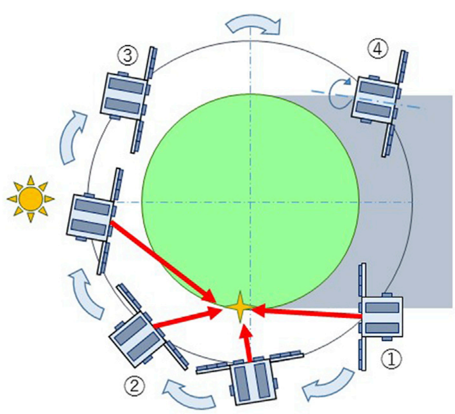

Fig. 14. Estimated scenario to lead to the attitude anomaly.

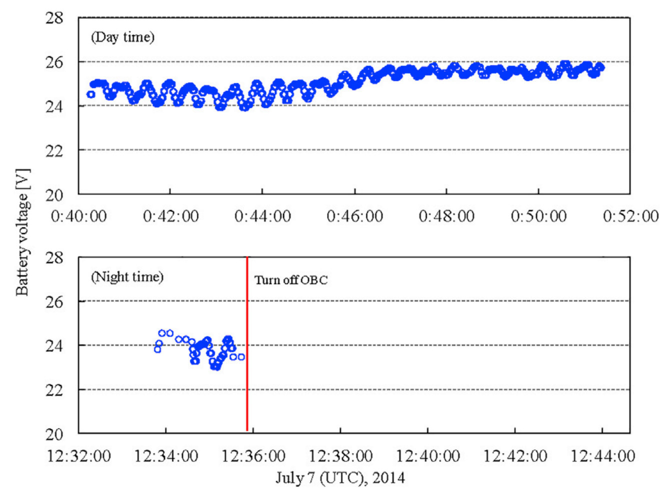

Fig. 15. Battery voltage in-orbit history during anomaly (in-orbit result of Hodoyoshi-4).

2) ADCS tries to track the misrecognized "Sun" and the satellite attitude was changed so that the solar paddle was directed towards this albedo (2 in Fig. 14).
3) As the battery voltage got lower, "software UVC" was activated and the satellite transited to safe mode and "spin-sun" control was automatically triggered.
4) During the spin up phase in spin-sun control, the satellite kept tracking the Earth albedo because real sun was opposite side of the SAS-2 and never was detected.
5) In Hodoyoshi-4 case, long duration of anti-sun pointing attitude caused the further lower battery voltage. Maybe because of such low voltage, the output of z-axis gyro was stuck, and as a result the satellite kept spinning up the rotation as there were no stopping signal coming from this gyro. At that time when the spin rate got about 10 deg/sec, hardware UVC occurred because of further lower battery voltage and main OBC was turned off. Unfortunately, the solar paddle was directed towards almost opposite direction from the Sun at this time, while spinning around z-axis with fast speed.
6) For Hodoyoshi-3 case, the first step of spin-sun mode was correctly terminated at 3 deg/sec, but during the process of changing the spin axis direction towards the Sun, the same hardware UVC occurred and OBC was turned off.
7) In both cases, the final spin axis (z-axis) direction was unfavorable in terms of solar power generation. Fortunately, a small number of solar cells were implemented on the other satellite surfaces, which could generate a little electric power, and the main solar paddle also could generate a little using Earth's albedo. Using this small generated power, during the day time, the battery voltage got a little higher which powered on OBC, but in night time, it became low and OBC was turned off again (Fig. 15).

In this critical situation, the only possible control is to change the direction of the spinning axis slowly towards the sun, using MTQ of the z-axis. Fortunately, z-axis, which is aligned with the spinning axis, almost keeps the same direction in inertial frame, and therefore continual activation of z-axis MTQ can generate the torque to rotate this spinning axis every time towards the same direction. We carefully powered on and off this z-axis MTQ, watching the change of battery voltage.

As a result, Hodoyoshi-3 was recovered after two days, while Hodoyoshi-4 required 10 days to recover as its spinning rate was about three times faster than Hodoyoshi-3's case. During this 10 days, Hodoyoshi-4's battery experienced more than 400 times of hardware UVC. It was feared that battery damage could have occurred, but fortunately Hodoyoshi-4 has been continuing its operation until now (October 2017).

We realized that three phenomena jointly caused this severe situation, which each resulted from our mistakes in the satellite basic architecture design.

a) Sun sensor misrecognized albedo as a direct sun light; this came from the inflexibility that the parameter to control sun sensor's sensitivity could not be tuned onboard.
b) Gyro's output was stuck: this was not observed during the ground test, but this possibility especially under low voltage should have been recognized, and appropriate countermeasures such as pre-filter to detect output freeze should have been prepared.
c) Spin-sun control was used in safe mode so that the satellite automatically transits to the sun acquisition mode: this is controversial. Spin-sun control is sometimes risky as it requires long time (such as several hours) during which the satellite should survive "Death Valley" of battery voltage. However, once spin-sun attitude is realized, we don't have to care much about the attitude stability.

After these crisis, we gave up spin-sun control to automatically transit from safe mode to sun acquisition mode. As zero-momentum strategy has been employed during mission standby and mission modes, the satellite is expected to have very small total angular momentum in safe mode. Then the satellite's attitude will be varied with various disturbances. In that case, with almost all the components turned-off, positive power balance can be realized, which was already assured in simulations before launch. From these considerations, the control during safe mode was switched after this crisis so that in safe mode the satellite will stay without doing anything automatically, and that a certain simple manual control algorithm using SAS will be employed to directly move from safe mode to mission standby mode.

6. Some hints to improve flexibility

As discussed in the previous sections, the satellite and its components will sometimes behave unexpectedly, in which case the satellite or ground operators should take appropriate actions to compensate for the unexpected events. The key factors to do so is "freedoms to change logic" implemented in the onboard software or hardware systems. In this section, some hints to improve such flexibility will be given, based on our experience in Hodoyoshi-3 and 4.

6.1. Parameters in OBC software are tunable by uplink commands

In Hodoyoshi-3 and 4, almost all the parameters used in OBC software can be updated by uplink commands, not by software upload. Examples include feedback gains, covariance of measurement noises for Kalman Filter, moment of inertia matrix, various thresholds, RWs torque distribution matrix, alignment matrix of attitude related sensors and actuators, orbit parameters, sensor/actuator control parameters. They can be tuned before launch through simulations or ground tests, but sometimes they have to be modified because of several reasons such as; (1) the space environment may be different from the ground tests (such as brightness of sun), (2) ground tests cannot derive appropriate values (such as moment of inertia matrix, residual magnetic momentum), (3) components characteristics may be changed from the ground test (such as component alignment variations because of launch load) or (4) even the documents from component vendors have wrong information (which actually occurred in our previous project).

"Software upload" requires significant efforts, and therefore simple methods to modify these parameters should be prepared. In Hodoyoshi-3 and 4 case, for example, covariance values of sun sensors to be used for Kalman Filter was changed between a very large value and the nominal value, which had the same effect as to turn off and on the specific sun sensor. As STT could not be used, gyro bias values were estimated in offline process on the ground and uplinked to OBC, which contributed to better accuracy in attitude propagation in night time. On the other hands, sufficient mechanisms to change some sensor control parameters were not prepared, such as thresholds for sun sensor to distinguish direct sun light from albedo or star sensors sensitivity parameters. As a result, these sensors could not be fully utilized, which made attitude determination and control much more difficult. From these experience, it is advisable to prepare as much flexibility of tuning parameters as possible.

6.2. Flexibility of change the guidance, navigation and control software codes

In Hodoyoshi-3 and 4, various "navigation", "guidance" and "control" software are modularized as "software parts," and their combination can flexibly be chosen by uplink command. For example, navigation parts include "quaternion calculation by SAS + GAS output," "quaternion calculation by STT," "coarse Kalman Filter using SAS + GAS" or "precise Kalman Filter using STT." Guidance parts, which define targets of attitude and angular rate, include "Sun pointing," "nadir pointing," or "target position pointing." Control parts include "b-dot to suppress angular velocity," "PD feedback by RWs to aim at target attitude and angular rate" or "MTQ control to spin-sun mode." By combining these navigation, guidance and control "software parts," together with the flexibility to tune their parameters, various customization of ADCS functions can be done without software uploading, which was really useful in Hodoyoshi-3 and 4 operations.

One example sequence which was not implemented initially but was generated by uplink commands and frequently used was "communication system operation mode transition at AOS and LOS" (AOS is "Acquisition of Signal" and LOS is "Loss of Signal" which correspond to the timings when the ground station starts and ends getting signal from the satellite), "covariance matrix change sequence." Safe mode definition was changed from "spin-sun" to "no control with small angular momentum," whose change was also easily implemented by using this customization function.

6.3. ADCS simulator

In order to verify such new guidance, navigation and control sequences or new parameters before actually trying on the real satellites, ADCS simulators on ground has been frequently utilized. In the Hodoyoshi Project a (a) MATLAB based simulator, (b) software simulator and (c) Hardware In the Loop Simulator (HILS) were developed. The MATLAB based simulator was used for initial trial and error to develop ADCS software, which was translated to C-code to be used for software simulator, which can carry out "fast time" Monte Carlo simulations to obtain statistical data of the ADCS performance. And in HILS, the onboard software can be tested in the real onboard computer with "real time simulation" using "hardware models" of various ADCS sensors and actuators. All the predicted disturbances are modeled in each of these simulators, and the software verified in HILS can directly be implemented on the OBC in the satellite Flight Model (FM). These simulators can be utilized not only for the onboard software development before launch but also for verifying new ADCS algorithms or parameters before uploading them to the satellites or trouble shooting of ADCS of in-orbit satellites, which contributed a lot to the flexibility of the ADCS.

7. Conclusions

Design of ADCS for satellite requires many considerations, among which the survivability of the satellite should be the primary focus. In Hodoyoshi-3 and 4 operations, many lessons learned were obtained, with which we had valuable opportunities to seriously consider how we can deal with various components' failures and safely operate our satellites. This consideration also gave us many hints toward desirable designs of ADCS architecture for university level microsatellites, including the following;

1) Components sometimes do not behave as expected, and therefore it is advisable to consider how the satellite can survive in case of failure of one component at any time. In order to do so, the effects of each component failure and how to compensate for it should be extensively studied and verified by simulations. These lessons should be reflected not only to ADCS design but also to the overall satellite architecture such as solar cells attachment strategy.
2) Whether to keep large angular momentum or keep almost zero momentum in the mission standby and mission modes needs fundamental research considering various factors including maximum moment of inertia direction, solar cells attachment surfaces, and power balance during safe mode.
3) Design of modes and mode transitions is very important. Especially the definition of the sun acquisition mode and how to transit to this mode should be carefully designed. Sometimes there is a "Death Valley" of battery voltage on its way to sun acquisition mode, and in that case, transition to this mode should be carefully triggered considering the current battery voltage.

4) Definition of and transition to the safe mode is also very important to make the satellite survive in severe situations. Power balance design during safe mode should be carefully made, for which the satellite total angular momentum has significant effects.
5) Implementation of onboard flexibility is very important. Some hints to change the satellite behavior without uploading new software, such as tuning control sequences or parameters by uplink commands, are explained based on Hodoyoshi-3 and 4 experiences.

Acknowledgements

HODOYOSHI project (2010–2014) was granted by the Japan Society for the Promotion of Science through the FIRST Program, initiated by the Council for Science and Technology Policy (CSTP)

Appendix A. Supplementary data

Supplementary data related to this article can be found at https://doi.org/10.1016/j.actaastro.2018.02.006.

References

[1] S. Nakasuka, From education to practical use in University of Tokyo's nano-satellite activities, Acta Astronaut. 66 (2010) 1099–1105.
[2] W.H. Steyn, Y. Hashida, In-Orbit attitude performance of the 3-axis stabilized SNAP-1 Nanosatellite, in: Proceedings of the AIAA/USU Conference on Small Satellites, Logan, UT, Aug. 13–16, 2001. SSC01-V-1.
[3] S. Sakai, Y. Fukushima, H. Saito, Design and on-orbit evaluation of attitude control system for small scientific satellite REIMEI, in: Proceedings of 25th ISTS (International Symposium on Space Technology and Science) and 19th ISSFD (International Symposium on Space Flight Dynamics), Kanazawa, Japan, June 4–11, 2006, 2006-d-84.
[4] Y. Nakajima, N. Murakami, T. Ohtani, Y. Nakamura, K. Hirako, K. Inoue, SDS-4 attitude control system: in-light results of three axis attitude control for small satellites, in: Proceedings of 19th IFAC Symposium on Automatic Control in Aerospace, 2013, pp. 131–136.
[5] Y. Sakamoto, N. Sugimura, K. Fukuda, T. Kuwahara, K. Yoshida, J. Kurihara, T. Fukuhara, Y. Takahashi, Development and flight results of microsatellite bus system for RISING-2, Trans. JSASS. Aero. Technol. 14 (ists30) (2016). Pf_89-Pf_96, 2016.
[6] K. Brieß, W. Bärwald, T. Gerlach, H. Jahn, F. Laura, H. Studemund, The DLR small satellite mission BIRD, Acta Astronaut. 46 (2–6) (2000) 111–120.
[7] J.C.T. Cheng, J. Lifshits, C.C. Grant, M. Barbu, R.E. Zee, The BRITE constellation space telescope design and test of a wide field, high resolution, low noise optical telescope for a nanosatellite constellation, in: Proceedings of the 25th Annual AIAA/USU Conference on Small Satellites, Logan, UT, 2011. SSC11-I-1.
[8] J. Thoemel, F. Singarayar, T. Scholz, D. Masutti, P. Testani, C. Asma, R. Reinhard, J. Muylaert, Status of the QB50 cubesat constellation mission, in: 65th International Astronautical Congress, Toronto, Canada, IAC-14–B4.2.11, 2014.
[9] "CubeADCS" https://www.cubesatshop.com/product/cube-adcs/.
[10] "CubeSat Attitude Control System" http://n-avionics.com/cubesat-components/attitude-control-systems/.
[11] "CubeSpace ADCS" https://www.cubespace.co.za/.
[12] S. Shirasaka, S. Nakasuka, Realization of the concept of reasonably reliable systems engineering in the design of nano-satellites, Trans. JSASS. Aero. Technol. 10 (ists28) (2012). Tt_7-Tt_12.
[13] Y. Tsuruda, Y. Aoyanagi, T. Tanaka, T. Matsumoto, S. Nakasuka, S. Shirasaka, M. Matsui, I. Mase, Demonstration of innovative system design for twin micro-satellite: Hodoyoshi-3 and -4, Trans. JSASS. Aero. Technol. 14 (ists30) (2016). Pf_131-Pf_140.
[14] K. Miyata, S. Nakasuka, On-orbit evaluations of the three axis stabilization performances of standard micro-satellite buses, Trans. JSASS. Aero. Technol. 14 (ists30) (2016). Pf_17-Pf_24.
[15] K. Miyata, S. Nakasuka, ADCS design with effective FDIR considering mission constraints for a small satellite, Trans. JSASS. Aero. Technol. 12 (ists29) (2014). Td_7-Td_13.
[16] A. Kutlu, Ch Haciyev, O. Tekinalp, Attitude determination and rotational motion parameters identification of a LEO satellite through magnetometer and sun sensor data, in: Recent Advances in Space Technologies RAST '07, 2007.
[17] R.J. Patton, F.J. Uppal, S. Simani, B. Polle, A. Monte Carlo, Analysis and design for FDI of a satellite attitude control system, IFAC Proceedings 39 (13) (2006) 1318–1323.
[18] M. Uo, K. Shirakawa, T. Hashimoto, T. Kubota, J. Kawaguchi, Attitude control challenges and solutions for Hayabusa spacecraft, in: Proceedings of 16th Workshop on JAXA Astrodynamics and Flight Mechanics, 2009, pp. 179–184.