ATTITUDE CONTROL OF HAYABUSA2 BY SOLAR RADIATION PRESSURE IN ONE WHEEL CONTROL MODE

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Abstract: *The asteroid explorer Hayabusa2 was launched by Japan Aerospace Exploration Agency (JAXA) on December 3, 2014. During the cruise phase, Hayabusa2 controls its attitude by only one reaction wheel to bias the momentum around Z-axis of the body in order to save the operating life of reaction wheels for other axes. In this one wheel operation, the Sun-aspect angle is restricted within a certain limit angle in terms of the thermal condition of the spacecraft. Because the precession radius is determined by the initial attitude and the direction of the equilibrium point, we execute the attitude maneuver around Z-axis to change the equilibrium point in order to reduce the Sun-aspect angle and succeeded. This paper presents that the concept and theorem of the attitude keeping and maneuvers by utilizing precession control under the solar radiation pressure, and introduce the flight results of Hayabusa2 operation.*

Keywords: *One wheel Control, Solar Radiation Pressure, Attitude Control*

1. Introduction

The main mission of the probe is to sample pieces of asteroid, and bring it back to the Earth in order to conduct more advanced analysis on the ground. Hayabusa2 is planned to arrive at the target asteroid in 2018, and return to the Earth in 2020 [1], [2].

During the cruise phase, Hayabusa2 controls its attitude by only one reaction wheel to bias the momentum around Z-axis of the body. This is to save the operating life of reaction wheels for other axes, because we experienced that two reaction wheels of three equipped on Hayabusa were broken after the touchdown mission.

In this one wheel control mode, the angular momentum direction is slowly moved in the inertial space (generally called precession) due to the solar radiation torque. This attitude motion caused by the balance of the total angular momentum and solar radiation pressure is known to trace the Sun direction automatically with ellipsoidal and spiral motion around Sun direction. Based on the knowledge in the past, the attitude dynamics model for Hayabusa2 mission had been developed before the launch [3]. According to the newly developed attitude dynamics model of Hayabusa2, the precession trajectory is almost the ellipsoid around the attitude equilibrium point, and this equilibrium point is determined mainly by the phase angle around Z-axis of the body.

In the actual operation of Hayabusa2, the spacecraft already experience the one wheel control mode, and the attitude motion in this mode is almost corresponds to the expected motion based on the dynamics model developed before the launch. The precession trajectory is ellipsoid around the equilibrium point, and the attitude dynamics model is verified by the actual flight data. In this one wheel operation, the Sun-aspect angle is restricted within a certain limit angle in terms of the thermal condition of the spacecraft. Because the precession radius is determined by the initial attitude and the equilibrium point, the Sun-aspect angle almost exceed the limit angle due to the precession without change of the equilibrium point. At this operation, we execute the attitude maneuver around Z-axis to change the equilibrium point in order to reduce the Sun-aspect angle and succeeded. After that, we execute the maneuver again to change the equilibrium point to close point in order to make the small precession trajectory.

2. Overview of Cruise Phase Operation

As introduced in the section 1, Hayabusa2 controls its attitude by only one reaction wheel to bias the momentum around Z-axis of the body during the part of cruise phase. In this cruise operation, there are several limitations of the attitude.

One restriction is about attitude orientation with respect to the Earth direction, which is limited by the antenna configuration. Hayabusa2 has the X-band Middle Gain Antenna (XMGA) which is equipped on the +Z panel and the pointing direction can be driven by motor within hemisphere of +Z axis side. There are also three X-band Low Gain Antennas (LGA). LGA-A is equipped on +Z panel, LGA-B is equipped on +X panel, and LGA-C is mounted on –Z panel. The boresight direction of each LGA is corresponds to the normal direction of each panel. Although the MGA is primary antenna in the cruise phase, the LGA is also frequently used mainly in attitude control operation because of its wide antenna pattern compared to the MGA. Thus, the restriction toward the Earth direction is limited by the antenna pattern of LGA. If we can use the LGA-A, the aspect angle between Earth direction and boresight direction of LGA-A should be kept within less than 60 deg.

The other restriction is about attitude orientation with respect to the Sun direction, which is limited by the thermal condition. As the preliminary condition, the +Z-axis direction is nominally points to the Sun direction because the Solar Array Panel (SAP) is mounted as illustrated in Fig.1. Under this condition of SAP direction, the aspect angle between Sun direction and +Z axis is permitted to change within 11 degree which is determined from the result of thermal analysis. The attitude restriction is illustrated in Fig. 2.

Figure 1. Configuration of Hayabusa2

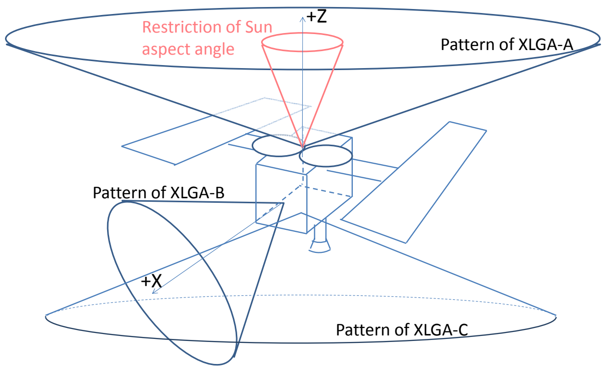

Figure 2. Attitude restriction

3. One Wheel Control Mode

In the Hayabusa operation, we experienced that two reaction wheels (RW) of three equipped RW's were broken after the touchdown mission. Even in that case, Hayabusa can return back to the Earth only with Z-axis RW by utilizing the solar radiation pressure. Therefore, Hayabusa2 has 4 reaction wheels for X, Y, Z1 and Z2 direction as shown in Fig.3. The redundancy concept for RW's clearly comes from Hayabusa lessons learned.

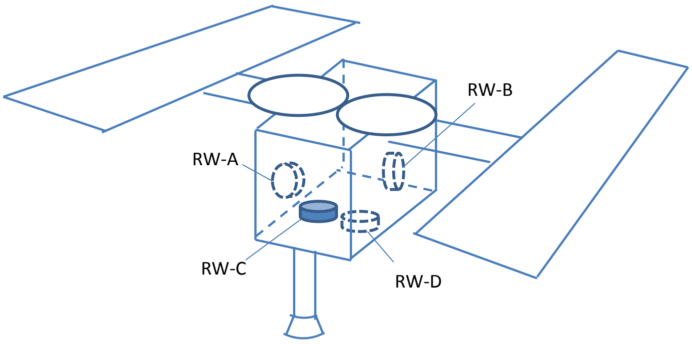

Figure 3. Location of mounted reaction wheel

During the cruise phase, Hayabusa2 controls its attitude only by one reaction wheel to bias the angular momentum around Z-axis of the body. There are two main reasons:

- To save the operating life of reaction wheels for other axes
- To save the fuel consumption.

First reason is from the redundancy concept learned from Hayabusa experience. The second reason is related to utilize the Solar Radiation Pressure (SRP). In this one wheel control mode, the angular momentum direction is slowly moved in the inertial space (generally called precession) due to the SRP torque. This attitude motion caused by the balance of the total angular momentum and SRP is known to trace the Sun direction automatically under the appropriate condition between SRP torque and angular momentum. The schematic of the Sun tracking motion is illustrated in Fg.4 and the geometry of the angular momentum vector and the SRP torque direction is shown in Fig.5.

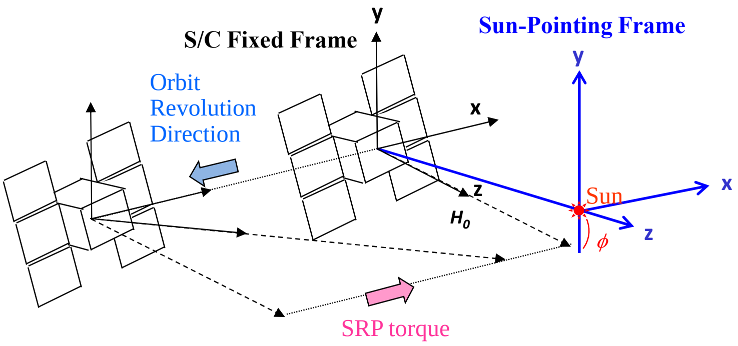

Figure 4. Sun tracking motion

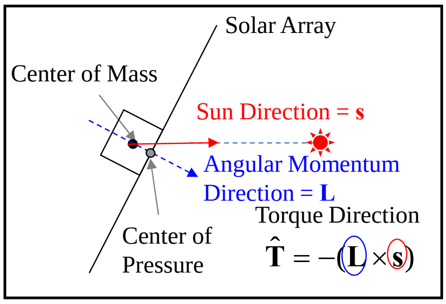

Figure 5. SRP torque direction

In Hayabusa and IKAROS mission [4], [5], this attitude motion was actually observed in the flight operation, and we have accumulated the experience and knowledge of the attitude dynamics under the solar radiation pressure. Based on this knowledge in the past, the attitude dynamics model for Hayabusa2 mission had been developed before the launch [3]. The detail about the dynamics is introduced in the section 4.

In general three axis control operation, Hayabusa2 should follow the Sun direction in order to keep the Sun aspect angle within a certain restriction determined from the thermal condition. It takes fuel to keep Sun aspect angle because the Sun direction automatically moves about 1 degree/day due to the orbit motion. However, by using the attitude motion due to the SRP, the angular momentum vector can trace the Sun direction automatically and fuel free to keep the Sun-aspect angle. The attitude motion in the inertial frame and Sun-pointing frame is illustrated in Fig. 6. As shown in Fig.6, the angular momentum makes circle trajectory below the Sun direction around the equilibrium point in the Sun-pointing frame.

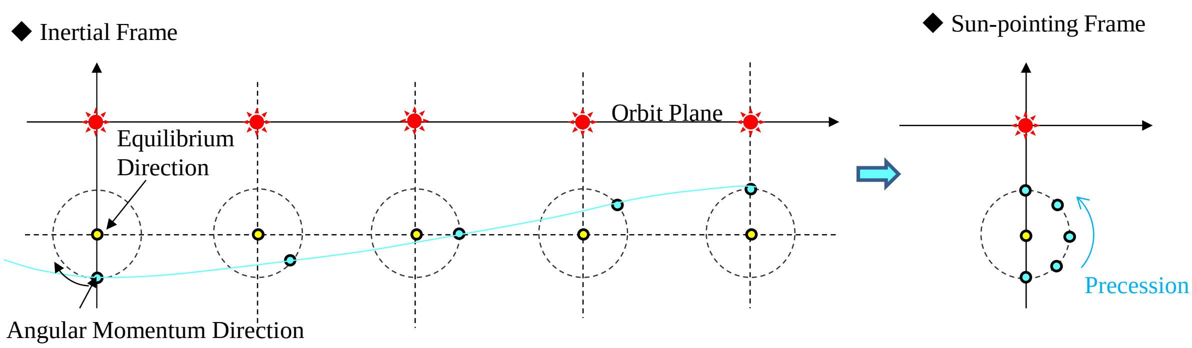

Figure 6. Sun tracking motion in inertial frame (left) and Sun-pointing frame (right)

In the actual operation, we should consider about the transition of the control mode. The 3-axis attitude of Hayabusa2 is nominally controlled by three RW's as bias-momentum. Thus, the momentum of the X and Y axis should coast down before spacecraft transits to the OWC mode. If the momentums are coasted down without control, however, the reaction torque affects the spacecraft attitude as the disturbance and the attitude starts tumbling. In order to avoid this, the attitude control mode is firstly transit to the 3-axis control mode by the thrusters called R3AX (RCS three-axis) control mode. In this R3AX mode, the thrusters are ignited when the attitude or the angular rate of the spacecraft are over the limits of the state (few degrees for the attitude and few 0.1 degree/sec for the angular rate). Therefore, the attitude is kept by thrusters when the RW's are coasted down, and after that the control mode transits to the OWC mode. Indeed, there are few degrees residual angle error and few 0.1 degree/sec residual angular rate, so the initial orientation of the angular momentum vector of RW-Z is affected by these residual states just after the transition to the OWC mode.

4. Dynamics Equations of One Wheel Control Mode

It is known that the attitude dynamics of the spacecraft in the deep space is dominated mainly by the SRP in general. Hayabusa2 is no exception. We start from the general formulation of SRP force is described as follow:

$$\mathbf{f} = -\frac{S_0}{c}\left(\frac{R_{S/C}}{R_E}\right)^2 [|\mathbf{s}\cdot\mathbf{n}|(C_{abs}+C_{dif})\mathbf{s} + (\mathbf{s}\cdot\mathbf{n})\{B_f C_{dif} + C_{abs}\kappa + 2C_{spe}|\mathbf{s}\cdot\mathbf{n}|\}\mathbf{n}]dA \tag{1}$$

where S_0 is the solar constant, c is the light speed, $R_{S/C}$ is the solar distance of the spacecraft, R_E is the representative solar distance of the Earth, C_{spe}, C_{dif} and C_{abs} are the specular, diffusion and absorption coefficient, respectively, B_f is the Lambertian coefficient, κ is the thermal emissivity, s is the Sun direction vector and n is the normal vector of the effective area of the SRP. The Sun direction vector is described as follow:

$$\mathbf{s} = \begin{bmatrix} \sin\alpha_s\cos\delta_s(\cos\alpha\cos\psi - \sin\alpha\sin\delta\sin\psi) - \cos\alpha_s\cos\delta_s(\sin\alpha\cos\psi + \cos\alpha\sin\delta\sin\psi) + \sin\delta_s\cos\delta\sin\psi \\ -\sin\alpha_s\cos\delta_s(\cos\alpha\sin\psi + \sin\alpha\sin\delta\cos\psi) + \cos\alpha_s\cos\delta_s(\sin\alpha\sin\psi - \cos\alpha\sin\delta\cos\psi) + \sin\delta_s\cos\delta\cos\psi \\ \sin\alpha_s\cos\delta_s\sin\alpha\cos\delta + \cos\alpha_s\cos\delta_s\cos\alpha\cos\delta + \sin\delta_s\sin\delta \end{bmatrix} \tag{2}$$

where α is the Right Ascension (RA), δ is the declination in the inertial frame, ψ is the rotation angle around Z-axis of the body and subscript s means the RA and declination of the Sun direction in the inertial frame. Assuming the Z-axis of the body-fixed frame is pointing close to the Sun direction and the equatorial plane, Eq. (2) can be reduced:

$$\mathbf{s} \approx \begin{bmatrix} -(\alpha-\alpha_s) \\ -(\delta-\delta_s) \\ 1 \end{bmatrix} \tag{3}$$

By using Eq. (3) and the normal vector **n**, the SRP torque formulation can be obtained. Although the normal vector **n** is derived from the integration of the local body shape, we skip the detail explanation here because the formulation is very complex in the strict expression (Ref. [3]). As the result of Eqs. (1) and (3) and the appropriate form of the normal vector **n**, the SRP torque can be formulated as follow:

$$\begin{bmatrix} T_x \\ T_y \\ T_z \end{bmatrix} = \begin{bmatrix} D & E & F \\ G & H & I \\ J & K & L \end{bmatrix} \begin{bmatrix} \cos\psi & \sin\psi & \\ -\sin\psi & \cos\psi & \\ & & 1 \end{bmatrix} \begin{bmatrix} \alpha - \alpha_s \\ \delta - \delta_s \\ 1 \end{bmatrix} \tag{4}$$

where *D~L* are the original SRP parameters which are invented in Ref. [3] and ψ is the phase angle around Z-axis.

Euler equation is described as follow in the general form:

$$(\mathbf{I\omega} + \mathbf{h}) + \boldsymbol{\omega} \times (\mathbf{I\omega} + \mathbf{h}) = \mathbf{T} \tag{5}$$

where **I** is the moment of inertia tensor, **ω** is the angular rate of the body-fixed frame with respect to the inertial frame, and **h** is the inertial angular momentum of the reaction wheels. In the OWC mode, each vector becomes:

$$\mathbf{I} = \begin{bmatrix} I_{XX} & & \\ & I_{YY} & \\ & & I_{ZZ} \end{bmatrix}, \qquad \boldsymbol{\omega} = \begin{bmatrix} \omega_x \\ \omega_y \\ \omega_z \end{bmatrix}, \qquad \mathbf{h} = \begin{bmatrix} \\ \\ h_z \end{bmatrix} \tag{6}$$

Substituting Eq. (6) into (5), we neglect $\dot{\omega}_z$ and the second order terms of ω_x and ω_y. In addition, if we assume that the time dependency of the precession is enough small compared to the nutation, we can also neglect $\dot{\omega}_x$ and $\dot{\omega}_y$:

$$\omega_x = -\frac{T_y}{h_z} \qquad \omega_y = -\frac{T_x}{h_z} \tag{7}$$

The kinematics equations [6] for X and Y components can be derived as follow in the case when the δ is small:

$$\frac{d}{dt}\begin{bmatrix} \alpha \\ \delta \end{bmatrix} = \begin{bmatrix} \dfrac{\sin\psi}{\cos\delta}\omega_x + \dfrac{\cos\psi}{\cos\delta}\omega_y \\ -\cos\psi\omega_x + \sin\psi\omega_y \end{bmatrix} = \begin{bmatrix} \sin\psi\omega_x + \cos\psi\omega_y \\ -\cos\psi\omega_x + \sin\psi\omega_y \end{bmatrix} \tag{8}$$

Substituting Eq. (8) into (7), the differential equation for the α and δ becomes:

$$\frac{d}{dt}\begin{bmatrix} \alpha \\ \delta \end{bmatrix} = \frac{1}{h_z}\begin{bmatrix} \cos\psi & -\sin\psi \\ \sin\psi & \cos\psi \end{bmatrix}\begin{bmatrix} T_x \\ T_y \end{bmatrix} \tag{9}$$

Substituting Eq. (4) into (9), and solve about α and δ, the analytical solution can be derived as follow:

$$\alpha = \alpha_{eq} + C_1 e^{\frac{D+H}{2h_z}t} \sqrt{\frac{\{M^3 + (D+H)^2 M\}^2 + N^2 P^2}{N^2}} \cos\left\{ \frac{M}{2h_z} t + C_2 - \tan^{-1}\left(\frac{NP}{M^3 + (D+H)^2 M} \right) \right\} \tag{10}$$

$$\delta = \delta_{eq} + C_1 e^{\frac{D+H}{2h_z}t} \sin\left(\frac{M}{2h_z} t + C_2 \right) \tag{11}$$

where C_1 and C_2 are integration constants determined from the initial state, and M, N and P are defined as follows:

$$M = \sqrt{\frac{-3D^2 + E^2 + G^2 - 3H^2 - 2DH - 6EG + 2(H-D)(E+G)\sin 4\psi + \{(D-H)^2 - (E+G)^2\}\cos 4\psi}{2}} \tag{12}$$

$$N = 4(DH - EG)\{2G + (D-H)\sin 2\psi - 2(E+G)\sin^2 \psi\} \tag{13}$$

$$P = \frac{(D-H)\cos 2\psi - (E+G)\sin 2\psi}{G - E + (D-H)\sin 2\psi + (E+G)\cos 2\psi} \tag{14}$$

Also the solution for equilibrium point is obtained:

$$\alpha_{eq} = \alpha_s + \frac{h_z}{2(DH-EG)}\left[\{D + H + (E+G)\sin 2\psi + (H-D)\cos 2\psi\}\dot{\alpha}_s + \{G - E + (H-D)\sin 2\psi - (E+G)\cos 2\psi\}\dot{\delta}_s\right] + \frac{(DI-FG)\sin\psi + (EI-FH)\cos\psi}{DH-EG} \tag{15}$$

$$\delta_{eq} = \delta_s + \frac{h_z}{2(DH-EG)}\left[\{E - G + (H-D)\sin 2\psi - (E+G)\cos 2\psi\}\dot{\alpha}_s + \{D + H - (E+G)\sin 2\psi + (D-H)\cos 2\psi\}\dot{\delta}_s\right] + \frac{(EI-FH)\sin\psi - (DI-FG)\cos\psi}{DH-EG} \tag{16}$$

According to Eqs. (10) - (14), the precession trajectory can be ellipsoid around the attitude equilibrium point and there is also divergent or convergent feature due to the exponential term. From Eqs. (15) and (16), the equilibrium point is determined by 6 SRP parameters (D~I), and the phase angle around Z-axis of the body with respect to the Sun direction. In Fig. 7, the converged ellipsoidal case of precession trajectory is plotted based on the analytical solution introduced as Eqs. (10) ~ (16). The trajectory is dependent on the SRP 9 parameters. The SRP parameters are determined from the relative position between center of mass and pressure, the optical properties of the exposed area, local shape of the spacecraft, and shadow. It means that the SRP parameters can be different by spacecraft and the precession trajectory is also different as that result. The 9 parameters for Fig. 7 are determined by the on-ground FEM model of Hayabusa2 based on the parameters of design as the most probable values.

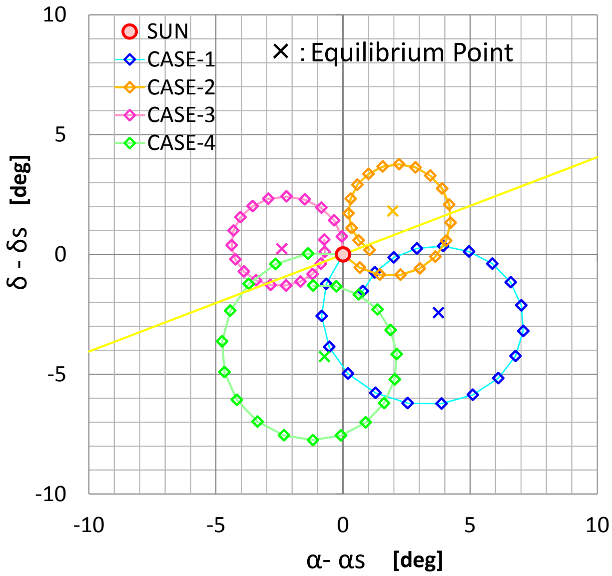

Figure 7. Example of precession trajectory under SRP in OWC mode

On February in 2015, we transit to the OWC mode at first time. Because of the residual angular rate and attitude offset introduced in the section 3, the initial point of the angular momentum vector (called just attitude below) cannot be chosen on the arbitrarily point. As the result, the initial attitude offsets -3 degree to $\alpha - \alpha_s$ and 2 degree to $\delta - \delta_s$ from the Sun direction as shown in left figure of Fig. 8. From initial attitude, the precession trajectory becomes ellipsoidal motion in the Sun-pointing frame. The equilibrium point only depends on the phase angle ψ around Z-axis as described in Eqs. (15) and (16), and it appears almost in the first quadrant in the Sun-pointing frame.

The residual between the estimated value and the observed value of the Right Ascension (RA) and Declination (Dec) is shown in the right figure of Fig. 8. As shown in this figure, the attitude recalculated within 0.4 degree accuracy based on the newly investigated model [3] and this means that the model is adequately accurate evaluating by the actual flight results.

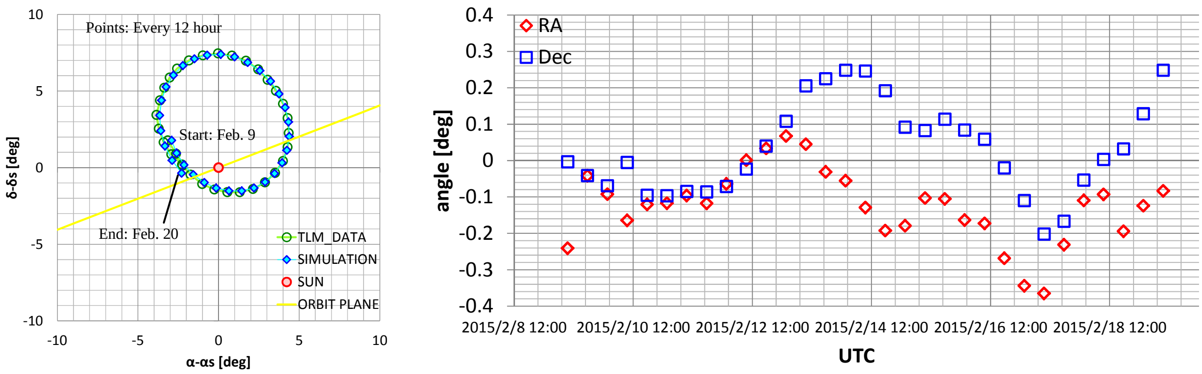

Figure 8. Model verification by estimated attitude trajectory and the flight data in Sun-pointing frame (left) and residual of the Right Ascension and Declination (right)

6. Control of Equilibrium Point

According to Eqs. (15) and (16), the equilibrium point of the precession trajectory is determined by the phase angle ψ around Z-axis. This is because the equilibrium point direction is almost uniquely determined by the relative position vector between the center of mass and pressure, and its direction in the Sun-pointing frame can be changed by the orientation of the rotation angle around Z-axis as shown in the left figure in Fig. 9. The plot of equilibrium points derived analytically is shown in the right figure in Fig. 9.

In Hayabusa2 case, both of the center of mass and the center of photon pressure shift from the geometric center of spacecraft to the other directions. The relative geometry of these two centers produces the bias of the solar radiation torque even in the case that the body Z-axis points to the Sun direction. As a result of the bias of the solar radiation torque, the equilibrium point is changed due to the phase angle around the body Z-axis illustrated in Fig. 9.

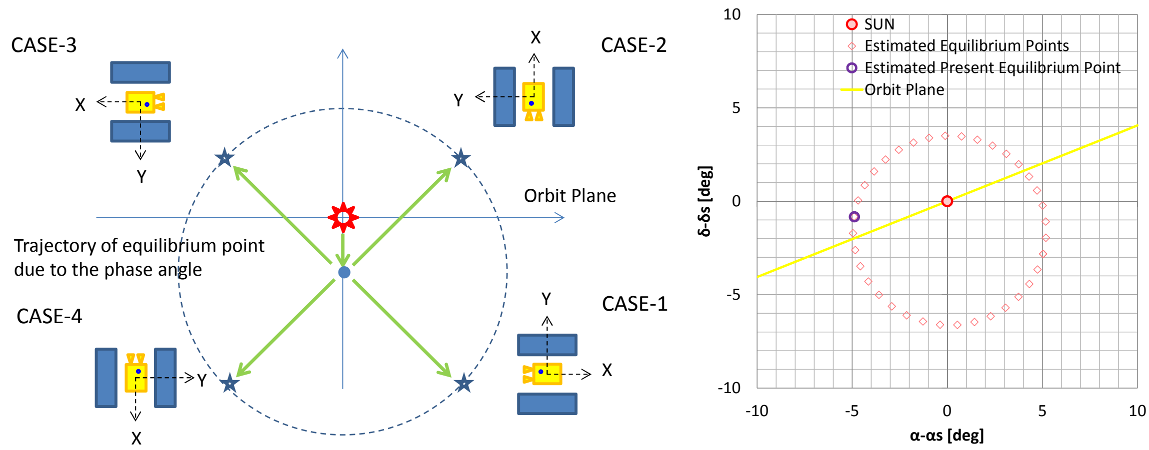

Figure 9. Relationship between relative position vector of center of pressure with respect to center of mass (left) and equilibrium points plotted based on analytical solution (right)

From this physical property, we noticed the fact that the precession trajectory can be changed, if we change the equilibrium point by changing the phase angle ψ around Z-axis at certain timing. If we go back to the flight result introduced in section 5 (also shown in right figure of Fig. 10), and simulate to change the equilibrium point at the cross point where the precession trajectory and line of equilibrium points due to the phase angle ψ, it is clearly shown that the precession trajectory is changed due to the change of the equilibrium point in the right figure in Fig. 10. This fact indicates that the precession trajectory can be controlled only by switching the equilibrium point direction by changing the phase angle.

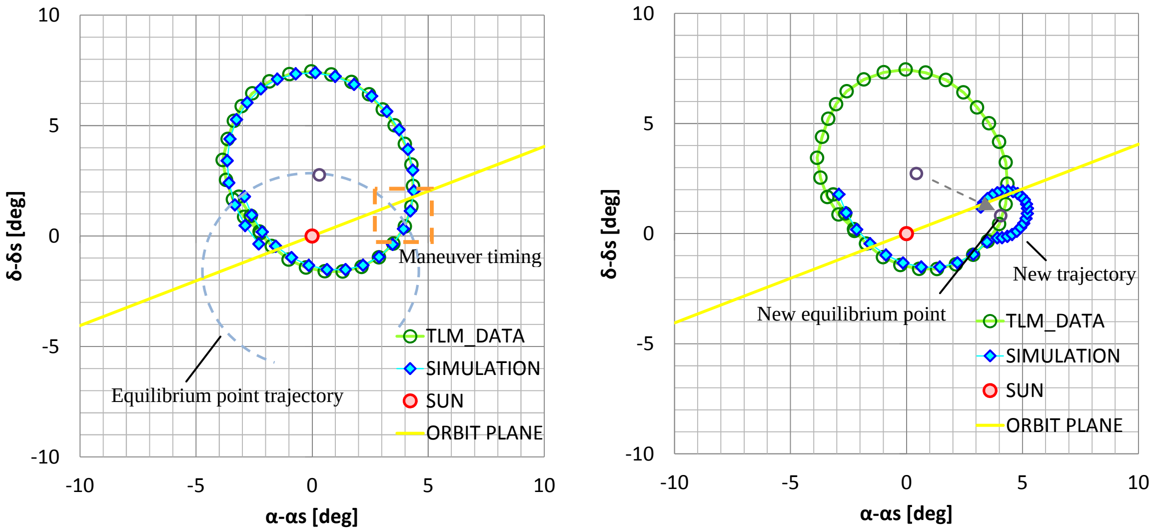

Figure 10. Flight result (left) and simulation of changing equilibrium point (right)

Applying this attitude dynamics to the actual operation, we keep the +Z-axis direction within the attitude restriction, i.e., 11 degree limited by the thermal condition. In the case of June (left figure in Fig. 11), the equilibrium point is controlled to move close to the attitude point. As the result, the precession radius can be gradually reduced. Although the attitude control is not executed during 11 days in the case of August (right figure), the precession radius is gradually reduced. This motion is induced only by the SRP torque. The attitude motion is very sensitive even to the small difference of the local shape of the spacecraft because the attitude motion itself is already very small phenomenon. In fact, the precession is gradually converged in August case. After the maneuver on August 11, the attitude goes into very small precession mode whose radius is less than 1 degree. This is the minimum record of the small precession of the Hayabusa2 operation so far. In both case, the attitude control by changing the equilibrium point succeed to reduce the precession radius.

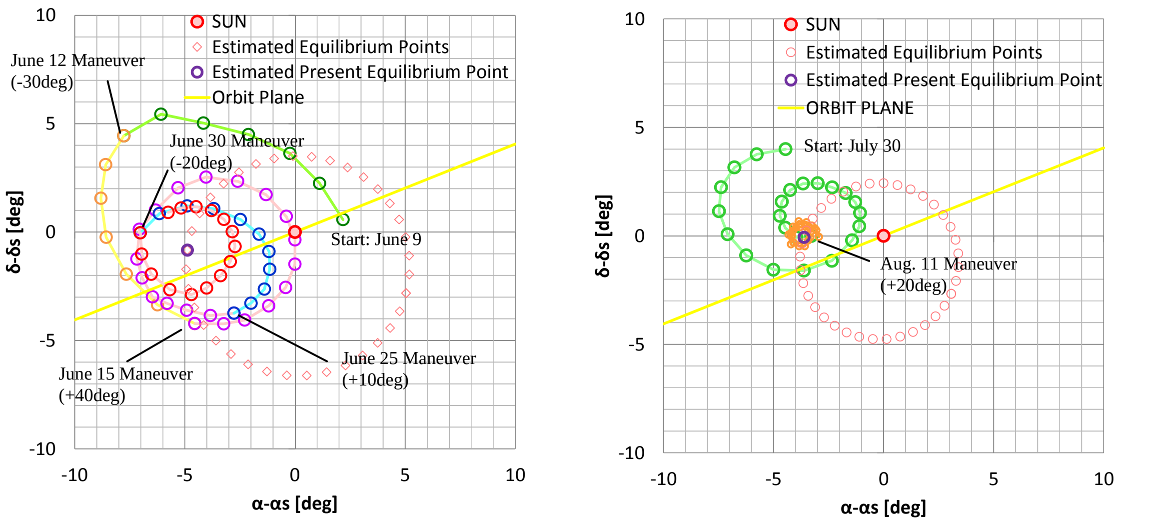

Figure 11. Flight results of the equilibrium point control in June (left) and August (right)

7. Conclusion

The novel attitude control scheme by controlling the equilibrium point of the precession trajectory is presented. This scheme is based on the SRP model which was invented before launch of Hayabusa2 [3]. According to the analytical solution of this SRP model, the equilibrium point is determined by the phase angle around the Z-axis of the body. By utilizing this attitude dynamics property, the precession trajectory can be controlled by controlling the phase angle ψ within the attitude restriction and succeed to reduce the precession radius. As the result, the utility of this control scheme is verified in the actual flight operation. This control scheme has possibility to be applied to the future spacecraft and potentially even to the operational spacecraft which has more than one RW under the SRP.

8. References

[1] Kuninaka H., "Hayabusa2 Project, Deep Space Exploration of Hayabusa-2 Spacecraft," ISTS-2015-k-61, 30th International Symposium on Space Technology and Science, Kobe, June 4-10, 2015.

[2] Tsuda Y., Yoshikawa M., Abe, M., Minamino H., Nakazawa S., System Design of The Hayabusa 2 – Asteroid Sample Return Mission To 1999 JU3, ActaAstronautica, Vol.90, pp.356-362, doi: 10.1016/j.actaastro.2013.06.028, 2013.

[3] Tsuda, Y., Ono, G., Akatsuka, K., Saiki, T., Mimasu, Y., Ogawa, N. and Terui, F. "Generalized Attitude Model for Momentum-biased Solar Sail Spacecraft," AAS Flight Mechanics Conference, AAS15-656, Vail, CO, USA, 2015.

[4] Kawaguchi, J. and Shirakawa, K., "A Fuel-Free Sun-Tracking Attitude Control Strategy and the Flight Results in Hayabusa (MUSES-C)," AAS Flight Mechanics Conference, AAS07-176, Sedona, 2007.

[5] Tsuda, Y., Saiki, T., Funase, R., Mimasu, Y., "Generalized Attitude Model for Spinning Solar Sail Spacecraft," AIAA Journal of Guidance, Control and Dynamics, Vol. 36, No. 4, pp. 967-974, 2013.

[6] Wertz, J.R. Ed., "Spacecraft Attitude Determination and Control," Kluwer Academic Publishers, Boston, pp.765-766, 1978.