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Reorientation Maneuver of NEOSSat Using Only Two Reaction Wheels Under Magnetic Dipole Torque Disturbance

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Abstract

Since a reaction wheel (RW) failure in 2001, NEOSSat's Attitude Determination and Control System (ADCS) has been operating in a three-wheel configuration, down from its original four-wheel setup. However, to ensure continued functionality in the event of an additional wheel failure, the system should be capable of operating in a two-wheel configuration, maintaining at least partial, if not full, operational capability. This paper presents the design and analysis of a control algorithm enabling three-axis attitude stabilization and slew maneuvers for NEOSSat using only two reaction wheels under different failure scenarios, whether the functional RWs are orthogonal or not. The approach assumes that NEOSSat can come to rest through the transfer of angular momentum between the spacecraft body and the operational RWs and/or via interaction between the Earth's magnetic field and the spacecraft's magnetic dipole. Slew maneuvers from any initial orientation to a given target orientation are executed using a hierarchical control system with two control loops. The inner-loop feedback directly influences only the satellite's angular velocity components, which are determined by the outer-loop feedback based on the quaternion representing the attitude difference between the actual and desired orientations. To maintain pointing stability after the slew maneuver and prevent orientation drift due to external torque disturbances—primarily caused by the interaction between the spacecraft's magnetic dipole and the Earth's magnetic field—the desired orientation is specifically determined such that the disturbance torque vector lies within the plane defined by the axes of the two active reaction wheels. Simultaneously, the satellite's telescope remains pointed toward the target of interest and within its field of view. This strategy ensures that the disturbance torques can be absorbed by the two functional reaction wheels, thereby maintaining stable pointing. Simulations are conducted to validate the proposed concept.

Keywords: Attitude control system, two reaction wheels, attitude determination and control system, underactuated control of satellites, fault recovery in satellits

Acronyms/Abbreviations

ADCS	Attitude Determination and Control System.
NEOSSat	Near-Earth Object Surveillance Satellite.
MOST	Microvariability and Oscillations of Stars.
RW	Reaction wheel.
RTN	Radial, Transverse, Normal Frame.
ECI	Earth-Centered Inertial Frame.
GPS	Global Positioning System.
ECEF	Earth-Centered, Earth-Fixed

Nomenclature

$\mathbf{A}$	=	Rotation matrix representing satellite attitude.
α, δ	=	Right Ascension and Declination.
$\mathbf{b}_{\text{earth}}, \mathbf{b}'_{\text{earth}}$	=	Erath magnetic field vectors expressed in the satellite body frame and ECEF frame.
$\mathbf{e}_1, \mathbf{e}_2$	=	Unit vectors aligned with the axes of active wheels.
$\mathbf{h}_w, \mathbf{h}$	=	Angular momentums of the wheels and entire satellite.
$\boldsymbol{\xi}$	=	Boresight axis expressed in the ECEF frame.
$\mathbf{m}_{\text{dipole}}$	=	Residual dipole of the satellite.
$\mathbf{q}$	=	Quaternion representing orientation error.
$\boldsymbol{\omega}, \boldsymbol{\Omega}$	=	Angular velocities of the satellite and wheels.
$\mathbf{I}_c$	=	Inertia matrix of satellite.

$\boldsymbol{\zeta}$	=	Unit vector representing telescope boresight.
J	=	Wheel inertia.
$\boldsymbol{\tau}, \boldsymbol{\tau}_{\text{dis}}$	=	Wheel torque and disturbance torque.
$\boldsymbol{\eta}, \boldsymbol{\eta}^*$	=	Quaternions representing the actual and desired attitudes.
θ	=	Greenwich Sidereal Time (GST) or Earth Rotation Angle (ERA)
$[.\times]$	=	Matrix form of the cross-product.
$[\boldsymbol{\eta} \otimes]$	=	Quaternion product.

1. Introduction

NEOSSat is a Canadian microsatellite designed for the detection and tracking of asteroids and artificial satellites from orbit. As shown in Figure 1, the satellite is equipped with a 15 cm aperture f/5.88 Maksutov telescope, similar to the instrument used on the MOST spacecraft [1]. Operating in space, NEOSSat leverages the directional capabilities of this telescope to fulfill its mission objectives. To accurately detect and track space objects, the satellite relies on its Attitude Determination and Control System (ADCS), which provides three-axis stabilization and achieves a pointing stability of 2 arcseconds over a 100-second exposure period. The ADCS, based on a zero-momentum design, is a cornerstone subsystem responsible for the high-precision inertial pointing and tracking of the satellite and its optical boresight. NEOSSat operates in two primary modes: (i) Fine Point, or "Star Stare" mode, in which the telescope remains fixed on the stars-causing satellites and asteroids to appear as streaks in the imagery; and (ii) Fine Slew, or "Track Rate" mode, in which the telescope tracks moving targets by matching their angular velocity, resulting in streaked star backgrounds [13], as illustrated in Figure 2.

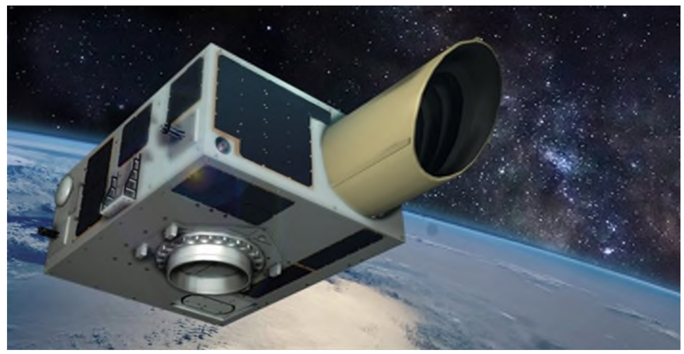

Figure 1: NEOSSAT artistic rendering.

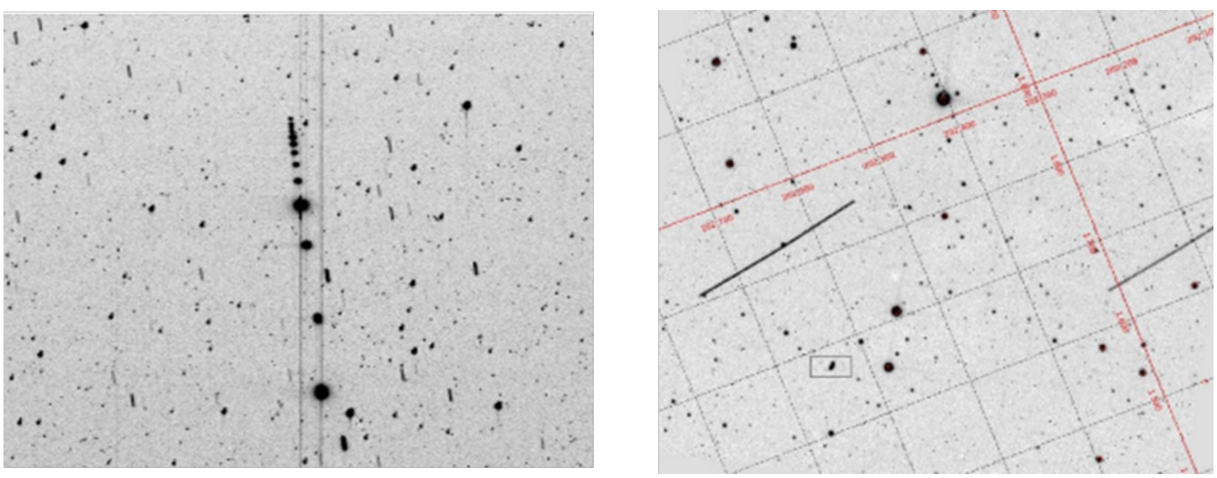

Figure 2: The Iridium 17 stack shows the object fading out as it retreats just after the TCA (left), while the Near-Earth asteroid 2017 VR12 is highlighted in the box (right).

The original Attitude Determination and Control System (ADCS) includes: One three-axis magnetometer; Coarse sun sensing, derived from solar panel output on each face; Four reaction wheels with integrated range sensors (X,Y,Z,and Skew); Three torque rods; A custom narrow-field star tracker using the telescope boresight; two GPS receivers [1]. NEOSSat's x-wheel experienced a significant failure in the year 2021 [12]. The failure is suspected to be electrical in nature, as telemetry data showed a gradual decrease in wheel speed rather than a sudden seizure. Subsequently, new ADCS parameters were uploaded, incorporating revised matrices for a three-wheel configuration. Nominal operations resumed, with science activities continuing at the same frequency as before the wheel failure. However, given the possibility of another wheel failure within the current three-wheel ADCS, the system should be prepared to operate in a two-wheel configuration to maintain functionality.

Spacecraft typically control their attitude using onboard multi-axis reaction wheels to manage rotation across three degrees of freedom. However, reaction wheel failures are a well-documented issue that can arise during a mission. This study focuses on attitude control in scenarios where two of the four reaction wheels fail, leaving the spacecraft to operate with only the remaining two. We analyze not only the underactuated nature of the system—where the number of control inputs is fewer than the generalized coordinates—but also the compensation of external torque disturbances under these failure conditions. Since the attitude control of a symmetric affine system is inherently non-holonomic, angular velocity is constrained by the conservation of angular momentum. Consequently, time-invariant continuous feedback cannot stabilize the system to arbitrary states. Therefore, spacecraft attitude control studies often assume zero initial angular momentum, demonstrating that control is feasible using time-varying [2] or switching feedback [3]. The control of spacecraft under actuator failure, based on the concept of a singular controller, was investigated in [11]. It is well established that a spacecraft can achieve full 3D attitude control with only two reaction wheels if its initial angular momentum is zero [2,5]. However, if the initial angular momentum is nonzero, the system exhibits a drift term, making control more challenging [6,7]. In such cases, attitude control to an arbitrary stationary state is only possible if the initial angular momentum lies within the plane spanned by the remaining control inputs [8]. Otherwise, attitude control requires considering non-stationary trajectories [9, 10], a constraint that we account for in our control approach. Notably, momentum transfer between a single reaction wheel and a rigid spacecraft is possible using feedback linearization [4]. This approach allows the wheel to absorb the spacecraft's initial angular momentum, bringing it to rest in steady state.

This work aims to restore full control of spacecraft slew maneuvers and attitude regulation using only two reaction wheels. A two-stage nonlinear state feedback controller, consisting of inner and outer feedback loops, enables the transition from an initial attitude to a desired one. The desired attitude is selected to: (i) ensure the target remains within the telescope's field of view, and (ii) align disturbance torques—primarily induced by the magnetic dipole—within the plane defined by the axes of the two active reaction wheels.

2. Feedback Control Design Strategy

2.1 Attitude Dynamics Model

Figure 3 depicts the body-fixed coordinate frame of the NEOSSat, where +X axis is aligned with its telescope. Suppose the quaternion $\boldsymbol{\eta} = [\eta_1 \ \eta_2 \ \eta_3 \ \eta_4]^T$ represents the spacecraft's orientation, where $\boldsymbol{\eta}_v = [\eta_1 \ \eta_2 \ \eta_3]^T$ is the vector part and η_4 is the scalar part. Also, let the quaternion $\boldsymbol{\eta}^*$ represent the desired orientation to be achieved at the end of the slew maneuver. The difference between the desired and actual orientations can be expressed as a quaternion, $\mathbf{q}$, which is derived from the following quaternion product

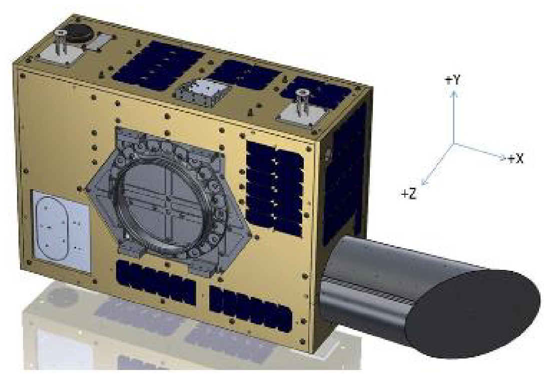

Figure 3: NEOSSat's body frame definition.

$$\mathbf{q} = \boldsymbol{\eta}^{*-1} \otimes \boldsymbol{\eta} \tag{1}$$

Here, $\boldsymbol{\eta}^{*-1}$ is the inverse of the quaternion $\boldsymbol{\eta}^*$, obtained by negating its vector part, and $[\boldsymbol{\eta} \otimes]$ represents the quaternion product, defined as follows:

$$[\boldsymbol{\eta} \otimes] = \boldsymbol{\Lambda}(\boldsymbol{\eta}_v) + \eta_4 \mathbf{I} \quad \text{where} \quad \boldsymbol{\Lambda}(\boldsymbol{\eta}_v) = \begin{bmatrix} -[\boldsymbol{\eta}_v \times] & \boldsymbol{\eta}_v \\ -\boldsymbol{\eta}_v^T & 0 \end{bmatrix}, \tag{2}$$

with $[\cdot\times]$ denoting the matrix form of the cross-product. Note that $\mathbf{q} = [0 \ \ 0 \ \ 0 \ \ 1]^T$ implies the actual quaternion converges to the desired one, i.e., $\boldsymbol{\eta} = \boldsymbol{\eta}^*$.

Suppose $\mathbf{I}_c = \operatorname{diag}(I_{xx}, I_{yy}, I_{zz})$ represents the satellite's total moment of inertia matrix, $\boldsymbol{\omega} = [\omega_x \ \omega_y \ \omega_z]^T$ is the angular velocity vector of the satellite body expressed in body coordinates, and $\mathbf{h}_w$ is the angular momentum vector of the wheels. Then, the total angular moment is given by:

$$\mathbf{h} = \mathbf{I}_c\boldsymbol{\omega} + \mathbf{h}_w. \tag{3}$$

The rate of change of the total angular momentum is

$$\dot{\mathbf{h}} + \boldsymbol{\omega} \times \mathbf{h} = \boldsymbol{\tau}_{\text{dis}}, \tag{4}$$

where $\boldsymbol{\tau}_{\text{dis}}$ represents the external disturbance torque applied to the satellite. The angular moment of the wheels can be written as

$$\mathbf{h}_w = J\mathbf{E}\boldsymbol{\Omega}, \tag{5}$$

where J is the wheel inertia, $\boldsymbol{\Omega} = [\Omega_x \;\; \Omega_y \;\; \Omega_z]^T$ represents the two-wheel speeds, and the column vectors of matrix $\mathbf{E}$ define the active spin axes. For example, if the actuator along the x-axis fails, the matrix assumes the following form:

$$\mathbf{E} = \begin{bmatrix} 0 & 0 & 0 \\ 0 & 1 & 0 \\ 0 & 0 & 1 \end{bmatrix}. \tag{6}$$

The equation of the wheel torque is

$$\dot{\mathbf{h}}_w = -\boldsymbol{\tau} \tag{7}$$

Using equations (5) and (5) in (4) gives the dynamics model of the attitude of the satellite.

$$\mathbf{I}_c\dot{\boldsymbol{\omega}} + \boldsymbol{\omega} \times (\mathbf{I}_c\boldsymbol{\omega} + \mathbf{h}_w) = \boldsymbol{\tau} + \boldsymbol{\tau}_{\text{dis}} \tag{8}$$

2.2 Hierarchical Feedback Control

Assume the satellite's orientation is such that the disturbance torque vector lies within the span of the active wheel axes matrix **E**, allowing it to be compensated by the remaining active wheels in a feedforward manner. Also assume the overall torque applied to the satellite in expressed by $\boldsymbol{\tau} = \mathbf{u} - \boldsymbol{\tau}_{\text{dis}}$, where $\mathbf{u}$ is an auxiliary control input. We further assume zero total momentum before the slew maneuver and a negligible change in the magnitude of angular momentum during the maneuver, i.e.,$\| \mathbf{h} \| \approx 0$. Then under these circumstances, the (8) is reduced to

$$\mathbf{I}_c\dot{\boldsymbol{\omega}} = \mathbf{u} \tag{9}$$

Based on the quaternion parameterization of attitude kinematics $\mathbf{q}$, the time derivative of the quaternion is related to the angular velocity by

$$\dot{\mathbf{q}} = \frac{1}{2}\boldsymbol{\Lambda}(\boldsymbol{\omega})\mathbf{q}, \tag{10}$$

where matrix function $\boldsymbol{\Lambda}(\cdot)$ was previously defined in (2). Without loss of generality, assume that the x-axis and skew-axis wheels fail in the following analysis, leaving the y-axis and z-axis wheels operational. Then, we consider the following nonlinear controller to obtain the instantaneous values of the angular rates based on the quaternion feedback as follows

$$\omega_y^* = -k_q q_2 + \mu\frac{q_1 q_2}{q_2^2 + q_3^2} \tag{11}$$

$$\omega_z^* = -k_q q_3 + \mu\frac{q_1 q_3}{q_2^2 + q_3^2} \tag{12}$$

where $k_q > 0$ and $\mu > 0$ are the positive feedback gains. By susbstituting the expressions of the virtual angular velocity inputs from (2.2) into (10), the closed-loop dynamics on the unactuated axis is given by

$$\dot{q}_1 = -\frac{1}{2}\mu q_1, \tag{13}$$

which means that the attitude component $q_1(t) = q_1(0)e^{-\mu t/2}$ about the unactuated axis converges to zero. The time derivative of the scaler quaternion q_4 is greater than zero for any nonsingular attitude:

$$\dot{q}_4 = -\frac{1}{2}(q_2\omega_y + q_3\omega_z) = \frac{1}{2}k_q(q_2^2 + q_3^2) \geq 0 \tag{14}$$

Under assumption that y-axis and z-axis wheels are operational, equation (9) leads to $\dot{\omega}_x = 0$, $I_{yy}\dot{\omega}_y = u_y$ and $I_{zz}\dot{\omega}_z = u_z$. Now consider the following control law for tracking the angular velocity dictated by (2.2)

$$\mathbf{u} = \mathbf{E}\mathbf{I}_c(k_\omega(\boldsymbol{\omega}^* - \boldsymbol{\omega}) + \dot{\boldsymbol{\omega}}^*), \tag{15}$$

and $\omega_x = \omega_x^* = 0$ and $k_\omega > 0$ is a positive feedback gain. The control inputs u_y and u_z in (15) serve as the inner-loop feedback, directly influencing only the angular velocity components ω_y and ω_z. These velocity components are determined by the outer-loop feedback in (11) and (12) to regulate the attitude accordingly. Thus, by defining the velocity error as $\tilde{\boldsymbol{\omega}} = \mathbf{E}(\boldsymbol{\omega}^* - \boldsymbol{\omega})$, the closed-loop system dynamics can be described by: $\dot{\tilde{\boldsymbol{\omega}}} = -k_\omega \tilde{\boldsymbol{\omega}}$.

Consider the following Lyapunov function

$$\begin{aligned} V &= k_q(\| \mathbf{q}_v \|^2 + (1 - q_4)^2) + \frac{1}{2}\| \tilde{\boldsymbol{\omega}} \|^2 \\ &= 2k_q q_4 + \frac{1}{2}\| \tilde{\boldsymbol{\omega}} \|^2 \end{aligned} \tag{16}$$

where (16) is derived from the identity $\| \mathbf{q}_v \|^2 + (1 - q_4)^2 = 2(1 - q_4)$, which follows from the definition of a unit quaternion. Using identities (14) and $\dot{\tilde{\boldsymbol{\omega}}} = -k_\omega \tilde{\boldsymbol{\omega}}$ in the time derivative of (16) leads to

$$\begin{aligned} \dot{V} &= -k_q\dot{q}_4 + \tilde{\boldsymbol{\omega}}^T\dot{\tilde{\boldsymbol{\omega}}} \\ &= -k_q^2(q_2^2 + q_3^2) - k_\omega \| \tilde{\boldsymbol{\omega}} \|^2 < 0 \end{aligned} \tag{17}$$

Therefore, as time progresses, $\mathbf{q} \to [0\ \ 0\ \ 0\ \ 1]^T$ and $\tilde{\boldsymbol{\omega}} \to \mathbf{0}$.

2.3 Optimal Alignment for Disturbance Rejection

The objective of this section is to derive the desired attitude for NEOSSat, ensuring that the observation satellite can point to any given target while simultaneously constraining the disturbance torque-primarily induced by the magnetic dipole-to lie within the plane formed by the axes of two active reaction wheels; see Figure 4. These conditions enable the disturbance torque to be effectively absorbed by the functional reaction wheels while keeping the target within the telescope's field of view.

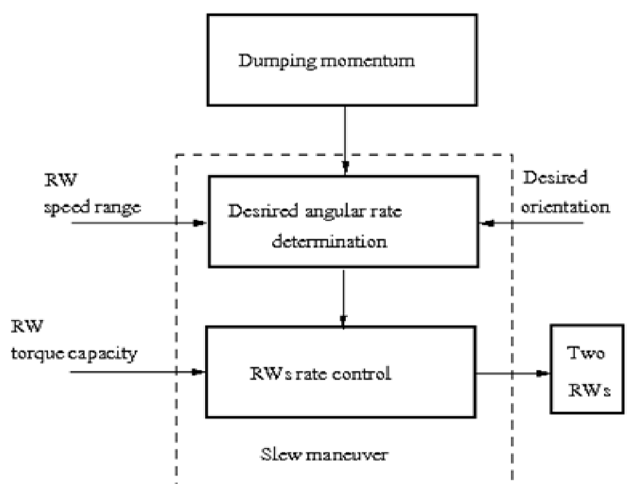

Figure 4: Attitude control system with two wheels

In satellite observations, such as those performed by NEOSSat, the attitude used to track a space object is typically defined by two parameters: Right Ascension, α, and Declination, δ. The roll angle around the telescope boresight is considered redundant, as it does not affect the position of the target within the telescope's field of view (see Figure 5). The concept behind optimal alignment for disturbance rejection using two reaction wheels leverages this redundancy by selecting an appropriate roll angle that aligns the external torque disturbance vector within the plane defined by the active reaction wheel axes.

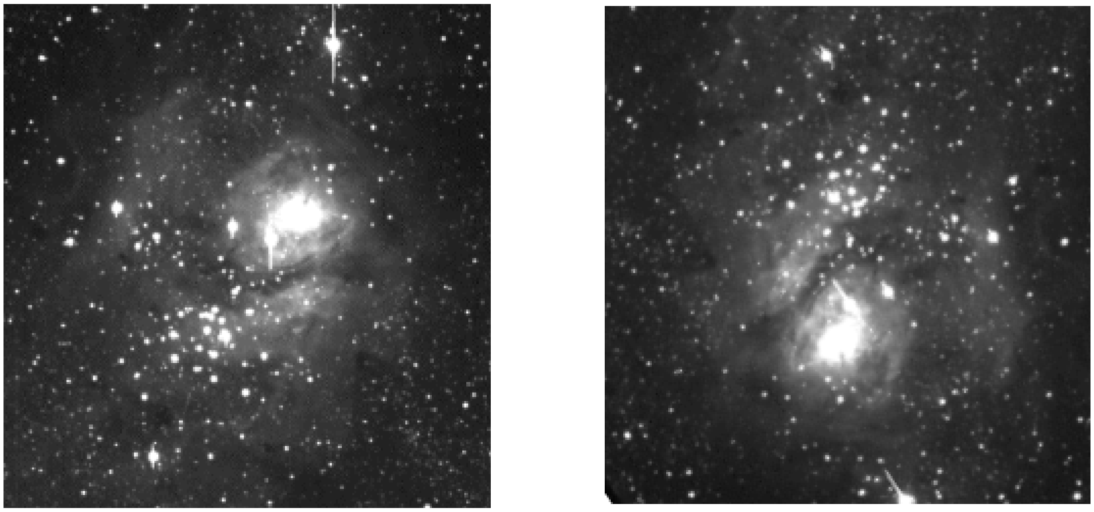

Figure 5: Original NEOSSat image of the M8 Lagoon Nebula (left) [13] and its roll-rotated version around the telescope boresight (right).

Figure 6 schematically shows how adjusting the roll angle about the boresight can align the magnetic dipole disturbance torque with the plane of two reaction wheel axes. Let the unit vector $\boldsymbol{\xi}$ represent the direction of the space object expressed in ECEF (Earth-Centered, Earth-Fixed) coordinate frame. This unit vector is related to Declination and Right Ascension by:

$$\boldsymbol{\xi} = \mathbf{R}\begin{bmatrix}\cos\delta\cos\alpha \\ \cos\delta\sin\alpha \\ \sin\alpha\end{bmatrix}.$$

Here, **R** is the rotation matrix from ECI (Earth-Centered Inertial) frame to ECEF frame constructed as follows:

$$\mathbf{R} = \begin{bmatrix}\cos\theta & \sin\theta & 0 \\ -\sin\theta & \cos\theta & 0 \\ 0 & 0 & 1\end{bmatrix},$$

where θ is the Greenwich Sidereal Time (GST) or Earth Rotation Angle (ERA) in radians. Suppose $\boldsymbol{\zeta} = [1\ \ 0\ \ 0]^T$ is a fixed unit vector expressed in the satellite body frame, aligned with the telescope's boresight axis. Let $\boldsymbol{A}$ be the rotation matrix representing the satellite's orientation with respect to ECEF. Then, the unit vector $\boldsymbol{\xi}$ aligns with a line extending from the center of the NEOSSat telescope toward the space object if the two vectors are related through the satellite's rotation matrix as:

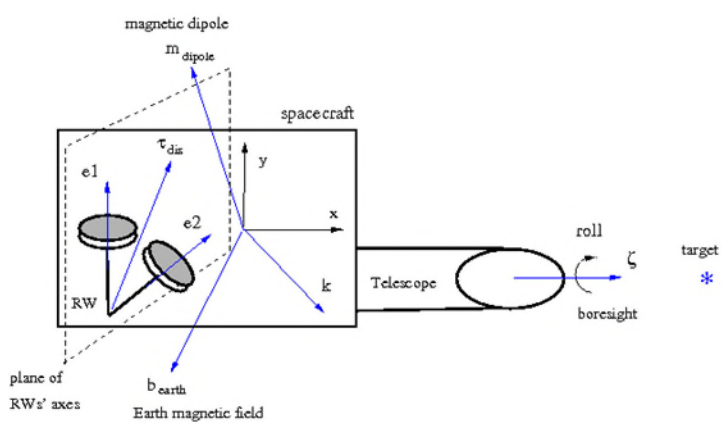

Figure 6: Disturbance torque vector being compensated by two RWs.

$$\mathbf{A}\boldsymbol{\zeta} = \boldsymbol{\xi} \tag{18}$$

The disturbance torque due to magnetic dipole can be expressed by

$$\boldsymbol{\tau}_{dis} = \mathbf{m}_{\text{dipole}} \times \mathbf{b}_{\text{earth}} \tag{19}$$

where $\mathbf{m}_{\text{dipole}}$ represents the residual magnetic dipole of the satellite, and the vectors $\mathbf{b}_{\text{earth}}$ represent the Earth's magnetic field in the body frame. Now, suppose the unit vectors $\mathbf{e}_1$ and $\mathbf{e}_2$ represent the axes of the active reaction wheels, i.e., the nonzero columns of matrix (6). Then, the condition for the alignment of the disturbance torque vector in the plane of $\mathbf{e}_1$ and $\mathbf{e}_2$ is given by:

$$\boldsymbol{\tau}_{\text{dis}} \cdot (\mathbf{e}_1 \times \mathbf{e}_2) = 0 \tag{20}$$

Upon substitution of (19) in (20), we arrive at

$$\mathbf{b}_{\text{earth}} \cdot \mathbf{k} = 0\,, \quad \text{where} \quad \mathbf{k} = \mathbf{m}_{\text{dipole}} \times (\mathbf{e}_1 \times \mathbf{e}_2) \tag{21}$$

is a constant vector independent of the satellite orientation. Now, denoting $\mathbf{b'}_{\text{earth}} = \mathbf{A}\mathbf{b}_{\text{earth}}$ as the Earth's magnetic field vector expressed in the ECEF frame and applying this identity in (21) gives

$$\mathbf{k}^T \boldsymbol{A}^T \mathbf{b'}_{\text{earth}} = 0 \tag{22}$$

One can solve equation (18) together with (22) to obtain the rotation matrix $\mathbf{A}$, which can then be equivalnely converted into the desired quaternion $\boldsymbol{\eta}^*$.

3. Case Study

NEOSSat is a compact, suitcase-sized microsatellite with dimensions of $137 \times 78 \times 38$ cm, including the telescope baffle, and a mass of 74 kg; see Figure 3. Assuming a uniform box model, the inertia matrix is given by $\mathbf{I}_c = \text{diag}\,(4.6 \;\; 12.5 \;\; 15.3)\ \text{kgm}^2$, while the inertia of all reaction wheels is assumed to be $J = 1.3 \times 10^{-3}\ \text{kgm}^2$. The initial orientation error for the slew maneuver is assumed to be $\boldsymbol{q} = [-0.0458 \;\; -0.1638 \;\; -0.4915 \;\; 0.8192]^T$, while the desired orientation error is $\boldsymbol{q} = [1 \;\; 0 \;\; 0 \;\; 0]^T$.

Two simulation scenarios are considered as follows:

- Failure mode 1: Only the y-axis and z-axis reaction wheels (RWs) are operational.
- Failure mode 2: Only the skew-axis and y-axis reaction wheels (RWs) are operational.

Figure 7 presents the trajectories of the satellite's angular velocity and the quaternion representing the orientation error during the slew maneuver with two reaction wheels under the hierarchical control system. The plots demonstrate that the controller successfully completed the satellite's reorientation within 20 minutes.

The reaction wheel (RW) speed and torque profiles in Mode 1 failure are plotted in Figure 8. Maximum speeds of the RWs during the slew maneuver are:

$$\omega_x = 6{,}200\ \text{RPM} \quad \text{and} \quad \omega_y = 27{,}300\ \text{RPM}$$

Maximum torques of the RWs during the slew maneuver are:

$$\tau_x = 6.2\text{mNm} \quad and \quad \tau_y = 8.8\text{mNm}$$

The reaction wheel (RW) speed and torque profiles in Mode 2 failure are plotted in Figure 9. Maximum speeds of the RWs during the slew maneuver are:

$$\omega_{\text{skew}} = 8{,}700\ \text{RPM} \quad \text{and} \quad \omega_y = 25{,}800\ \text{RPM}$$

Maximum torques of the RWs during the slew maneuver are:

$$\tau_{\text{skew}} = 8.8\text{mNm} \quad \text{and} \quad \tau_y = 12.8\text{mNm}$$

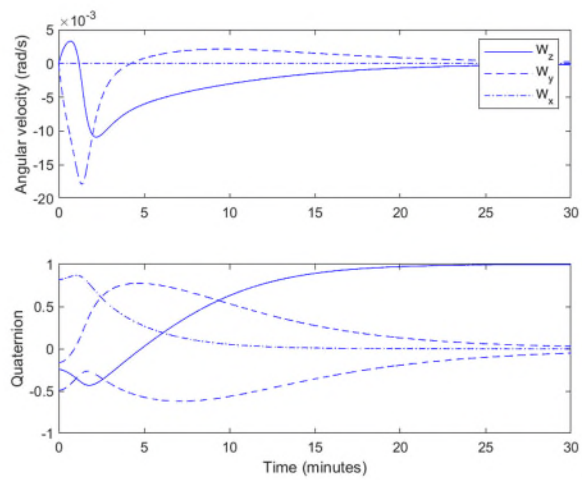

Figure 7: Slew maneuver using two RWs.

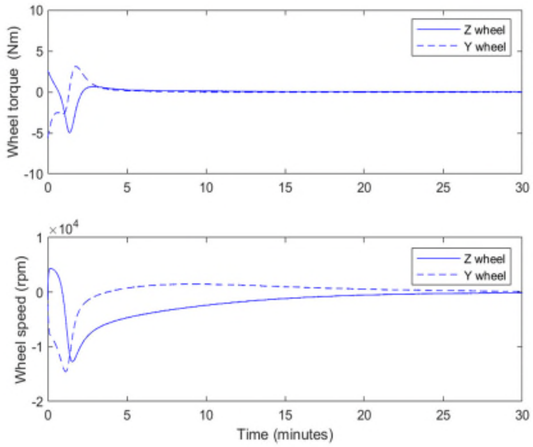

Figure 8: Torques and speeds of the wheels in Failure Mode 1.

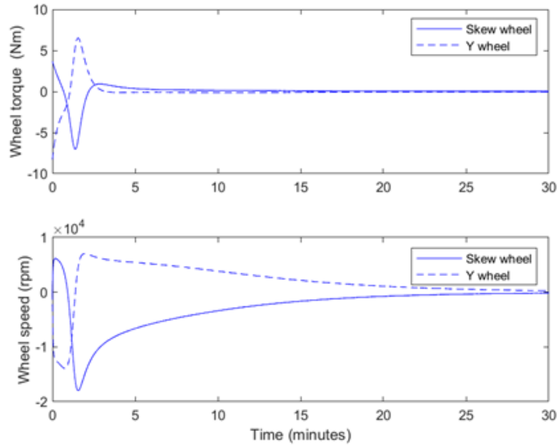

Figure 9: Torques and speeds of the wheels in Failure Mode 2.

4. Conclusion

This paper presented the design and analysis of a control algorithm for achieving three-axis attitude stabilization and slew maneuvers for NEOSSat using only two reaction wheels under various failure scenarios. The proposed approach accounted for both cases where the functional reaction wheels were either orthogonal or non-orthogonal. It was assumed that the spacecraft could achieve rest through the transfer of angular momentum between the satellite body and the operational reaction wheels and/or through interaction with the Earth's magnetic field. A hierarchical control strategy was implemented, with the inner-loop feedback directly influencing the angular velocity components, while the outer-loop feedback determined the necessary control inputs based on the quaternion representation of the attitude error. To ensure stable pointing after the slew maneuver and minimizing orientation drift caused by external torque disturbances, the desired orientation was selected such that the disturbance torque vector remained within the plane of the two active reaction wheels. This approach enabled the functional reaction wheels to absorb disturbances while keeping the telescope pointed at the target within its field of view. Simulation results demonstrated the feasibility of achieving three-axis attitude stability and executing slew maneuvers using only two reaction wheels, regardless of their relative orientation. The algorithm successfully reoriented NEOSSat from any initial orientation to a desired orientation within 20 minutes, given the reaction wheels' speed and torque constraints.

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