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## Design and Implementation of Attitude Control System for GNSSaS6U CubeSat Satellite

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### Abstract

This paper studies the attitude determination and control system (ADCS) of the GNSSaS CubeSat. It investigates two modes of operation: the Nadir Pointing sub-mode and the Sun Pointing sub-mode of the GNSSaS. The controller that is used in the GNSSaS CubeSat is a sliding mode controller (SMC), which uses a magnetorquer and four reaction wheels as actuators. The Nadir Pointing sub-mode points the antenna towards the Earth for payload operation, whereas the Sun Pointing sub-mode points the satellite's solar panels towards the Sun. According to the simulation results, the ADCS can successfully maintain the antennae pointed toward the Earth in the Nadir Pointing sub-mode and the solar panels pointing toward the Sun in the Sun Pointing sub-mode within 0.1-degree accuracy. The Nadir Pointing sub-mode's performance is significant since it maintains the antenna's orientation towards the Earth for payload operation. The performance of the Sun pointing sub-mode is essential since it ensures that the satellite's battery is recharging while the solar panels are facing the Sun. The study shows that the ADCS system can achieve the requirements for pointing accuracy for both sub-modes.

**Keywords:** Attitude control, CubeSat, Pointing Accuracy, SMC.

### Nomenclature

$q$	The unit quaternion of the CubeSat that represents its orientation with respect to the inertial frame
$J_S$	The moment of inertia matrix of the CubeSat
$\dot{\vec{\omega}}_S^I$	The first derivative of the angular velocity of the CubeSat expressed in the Earth Centered Inertial frame
$\dot{H}_w$	The first derivative of the reaction wheels angular momentum vector
$\vec{\omega}_S^I$	The angular velocity of the CubeSat expressed in the Earth Centered Inertial frame
$H_w$	The reaction wheels angular momentum vector
$\vec{\tau}_{grvi}$	The gravity disturbance torque
$\vec{\tau}_{rad}$	The radiation disturbance torque
$\vec{\tau}_{drag}$	The drag disturbance torque
$\vec{\tau}_{mag}$	The magnetic field disturbance torque
$\vec{\tau}_{MTR}$	The magnetorquer control torque
$\Omega(\vec{\omega})$	The matrix representation of the angular velocity vector
$F$	The force acting on the satellite
$m$	The mass of the satellite
$\vec{r}$	The acceleration of the satellite
$\vec{r}$	The position vector of the satellite with respect to the Earth
$\mu$	The gravitational parameter of the Earth
$a$	The semimajor axis
$e$	The eccentricity
$i$	The inclination
$\Omega$	The right ascension of the ascending node
$\omega$	The argument perigee
$M$	The mean anomaly

### Acronyms/Abbreviations

ADCS	Attitude Determination and Control Subsystem
COM	Center of Mass

ECEF	Earth Centered Earth Fixed
ECI	Earth Centered Inertial
GNSS	Global Navigation Satellite System
GNSSaS	Global Navigation Satellite System Augmentation System
LEO	Low Earth Orbit
LQR	Linear Quadratic Regulator
LVLH	Local Vertical Local Horizontal
MPC	Model Predictive Control
MW_SMC	Sliding Mode Controller Parameter
NSSTC	National Space Science and Technology Center
NORAD	North American Aerospace Defence Command
PID	Proportional-Integral-Derivative
RAAN	Right Ascension of the Ascending Node
RMS	Root Mean Square
SMC	Sliding Mode Controller
TEL	Two Line Element
UAE SA	United Arab Emirates Space Agency
UAEU	United Arab Emirates University
UTC	Coordinated Universal Time

## 1. Introduction

CubeSats are nanosatellites built to standard size dimensions called Units, or U. One U is  $10 \times 10 \times 10$  cm; CubeSats can be 1U, 2U, 3U or 6U in size [1]. The advantages of using CubeSats are: 1) they are simple in design for a short mission, 2) they can be built approximately within two years, and 3) they are less expensive than large satellites [2]. The most challenging part of the CubeSat is the pointing accuracy and how to control the pointing of the CubeSat in Space.

The attitude control system is the system that helps in analyzing and controlling the direction of the satellites in space. The term attitude in the space field refers to the satellite's orientation relative to an object in space, such as the Earth. The attitude control system is one of the essential systems in the Satellites because it is responsible for 3-axis attitude control with desired pointing accuracy, and to counteract different disturbance environmental torques such as aerodynamics, solar and gravity disturbance torques.

Attitude control consists of three main parts: sensors, actuators and algorithms. Various sensors are used in measuring the attitude; examples are a magnetometer and a Star Tracker. There are different types of actuators, like reaction wheels and magnetorquers. Actuators orient the satellite to the desired attitude by applying the control torques. While the controller (algorithm) will be implemented in the flight computer, it is responsible for processing the sensor measurements, computing the attitude and generating the actuator's command, which are the control torques to control the CubeSat. The controller's accuracy depends on the actuators, sensors and the selected control method.

The algorithm is the core element in the attitude control system; it is usually designed using control methods such as Model Predictive Control (MPC), Linear Quadratic Regulator (LQR), Proportional-Integral-Derivative (PID), Lyapunov theory and sliding mode control (SMC) methods. Junquan, Mark, Thomas, and Regina [3], investigated the nonlinear adaptive fuzzy sliding mode control method to control the attitude of 1U CubeSat, using reaction wheels and magnetorquers as actuators and magnetometer and gyroscope as sensors. The proposed controller was designed for two scenarios; the first scenario is the attitude stabilization mode Nadir pointing and limb pointing using three magnetorquer and one reaction wheel on Y-axis. This configuration was designed in case a failure happens in the remaining two reaction wheels. The second scenario is used in attitude stabilization mode Nadir pointing using only three reaction wheels. This configuration gives a high attitude control accuracy and a short settling time. This paper conclude that the attitude control accuracy will be more robust using the nonlinear adaptive fuzzy sliding mode control method than the PD magnetic control method. Hamed Yadegari [4] investigates the interval type-II fuzzy terminal sliding mode control using magnetorquer as actuator, which shows that the controller is able to track the desired attitude response with the present of disturbance. The sliding mode controller has been investigated in the literature in various articles [5],[6] and [7], and it is concluded that the SMC has a good performance for tracking trajectory.

Other methods used to control the attitude of the satellites were found in the literature, such as the Model Predictive Control approach [8], linear quadratic regulator controller [9] and PID controller [10]. The PD controller,  $H_2$  controller,  $H_\infty$  controller, and a mixed  $H_2/H_\infty$  controllers were investigated in the literature as well. The literature concluded that the  $H_2$  controller performs well compared to the other controllers, while the mixed  $H_2/H_\infty$  has the most significant

stability margin. The  $H_2$  controller requires the most effective control effort, while the  $H_\infty$  requires the smallest control effort. In contrast, the mixed  $H_2/H_\infty$  has an intermediate response time value compared to the  $H_2$  and  $H_\infty$  [11].

In the research literature, several sensors, actuators, and control algorithms have been developed and implemented. The reaction wheels are one of the most important actuators in the ADCS of CubeSats. Several studies in the literature have suggested using reaction wheels in the ADCS of CubeSats. The reaction wheel is used as the actuator in the proposed controller which can achieve high tracking precision, fast convergence time and can reject the disturbance [12].

The primary purpose of this study is to develop an efficient and stable attitude control system that can fulfill the requirements for the GNSSaS CubeSat's attitude accuracy. This will be accomplished by putting into practice two modes of operation, each of which uses a separate set of algorithms to achieve the level of pointing accuracy required for the Nadir pointing and Sun pointing sub-modes, respectively. This paper is divided into four main sections. The first section represents the GNSSaS's mission and requirements; the second section describes the mathematical modelling; the third is the simulation results; and finally, the discussion and conclusion sections.

## 2. GNSSaS Mission and Requirements

### 2.1 GNSSaS Mission

GNSSaS is a 6U CubeSat designed and built at the National Space Science and Technology Center (NSSTC) at United Arab Emirates University (UAEU). The project is funded by the United Arab Emirates Space Agency (UAE SA). The material properties of the GNSSaS CubeSat are listed in Table 1 below. GNSSaS Attitude Control System are divided into six primary modes: OFF, Idle, Detumbling, Coarse Pointing, Medium Pointing, and Fine Pointing. Each pointing mode has specific sub-modes, such as the Nadir Pointing sub-mode, the Sun Pointing sub-mode, the Ground Target Tracking sub-mode, and the Momentum Dumping sub-mode. The coarse pointing mode has a lower pointing accuracy of approximately 15 degrees compared to other pointing modes. The Medium Pointing mode has a pointing accuracy of around 1 degree. However, the Fine Pointing mode is capable of achieving the highest pointing accuracy of around 0.5 degrees, which is the highest potential accuracy that can be achieved by the system. In this paper, the controller will be designed for the fine pointing mode, specifically for the nadir and sun pointing sub-modes, with a pointing accuracy of less than 0.5 degrees.

The National Space Science and Technology Center (NSSTC) of UAE University is responsible for the design and implementation of the GNSSaS CubeSat, shown in Figure 1 as a 6U communications CubeSat. Attitude Control system of the GNSSaS CubeSat utilizes the sliding mode controller. Six fine sun sensors, a magnetometer, a gyroscope, a Star Tracker, and a GNSS Receiver is used as sensors, while a magnetorquer and four reaction wheels is used as actuators. The hardware for the GNSSaS CubeSat is provided by GOMSpace©. To meet the mission requirements of the GNSSaS CubeSat, the Attitude Control System's pointing accuracy must be less than 0.5 degrees.

Table 1. GNSSaS Material Properties

Parameters	SolidWorks	
	Stowed	Deployed
<b>Configuration</b>	Stowed	Deployed
<b>Mass of the Satellite (kg)</b>	9.94948	9.94948
<b>Moment of Inertia (<math>kg \cdot m^2</math>)</b>	$I_{XX} = 0.055464747$	$I_{XX} = 0.084857475$
	$I_{YY} = 0.102831796$	$I_{YY} = 0.103231746$
	$I_{ZZ} = 0.136691035$	$I_{ZZ} = 0.165587804$
<b>Center of Mass (COM)(m)</b>	$\Delta X = 0.1703$	$\Delta X = 0.1702$ m
	$\Delta Y = 0.0064$	$\Delta Y = 0.0064$ m
	$\Delta Z = 0.0013$	$\Delta Z = -0.0012$ m

### 2.2 GNSSaS Operational Requirements

The mission requirements for the ADCS of the GNSSaS CubeSat depends on the payload requirements and other conditions such as the following:

- 1) Functional Requirements:
  - The Satellite shall be able to determine its own pointing vector in the 3D space
  - The ADCS shall be able to determine the position of the Sun
  - The Satellite shall be able to point to a specific target on execution of a command from the ground
- 2) Performance Requirements:
  - The Satellite shall be three axes stabilized
  - The Satellite shall have a pointing accuracy of  $< 1$  degree
  - The Satellite shall have a pointing stability of  $< 1$  RMS degree/s
  - The ADCS shall be able to stabilize the Satellite from the initial spin rate of 5 deg/sec within 1 day
- 3) Operational Requirements:
  - FCS shall be implemented in the ADCS computer
  - The ADCS shall communicate with I2C interface
  - The input voltage for the ADCS shall be 3.3 or 5V
  - The ADCS should follow the PC104 standard pin configurations
  - The ADCS optical component shall survive in the temperature range of  $-30\text{ C}^{\circ}$  to  $+70\text{ C}^{\circ}$

### 2.3 GNSSaS Reference Frame

The Attitude control environment uses the frame notation as described in this section. The coordinate systems used are right hand coordinate [13].

- Earth Centered Inertial (ECI)

The Earth Centered Inertial (ECI) frame has its origin in the center of mass of the Earth. It is oriented with its X-axis aligned with the mean equinox, its Z-axis pointing through the North Pole, and the Y-axis completes the right-hand orientated Euclidean basis.

- Earth Centered Earth Fixed (ECEF)

The Earth Centered Earth Fixed (ECEF) frame is defined in the same way as the ECI, however, the X-axis is pointing in the direction where the prime meridian meets equator, 0-degree Latitude, 0-degree Longitude, thus rotating around the Z-axis. The Y-axis completes the right-hand orientated Euclidean basis.

- Spacecraft (S)

The Spacecraft (S) frame is fixed onto the spacecraft body. The default orientation has been chosen as +X: The positive X-axis is aligned with the orbit velocity vector which is aligned also with the tuna cans and it is perpendicular to the 2U side as shown in Figure 1 +Z: The positive Z-axis is out of the side from which the camera/payload points which means that it is perpendicular to the 6U side. +Y: is given as  $Z \text{ cross } X$  thus completing a right handed Euclidean coordinate system. The origin of the frame is situated in the center of the geometry.

- Orbit (O)

The Orbit (O) frame has its origin at the satellite center of mass, with its X-axis aligned with the satellite's velocity vector, its Y-axis pointing in the direction of the orbital angular momentum and the Z-axis completes the right hand orientated Euclidean basis.

- Local Vertical Local Horizontal (LVLH)

The Local Vertical Local Horizontal (LVLH) frame has its origin at the satellite. The +Z-axis points towards the Earth's center of mass (COM), the Y-axis points in the direction of the orbital angular momentum and the X-axis completes the right-hand orientated Euclidean basis. Only in the case of a perfect circular orbit, the Orbit reference frame and the LVLH reference frame will be align.

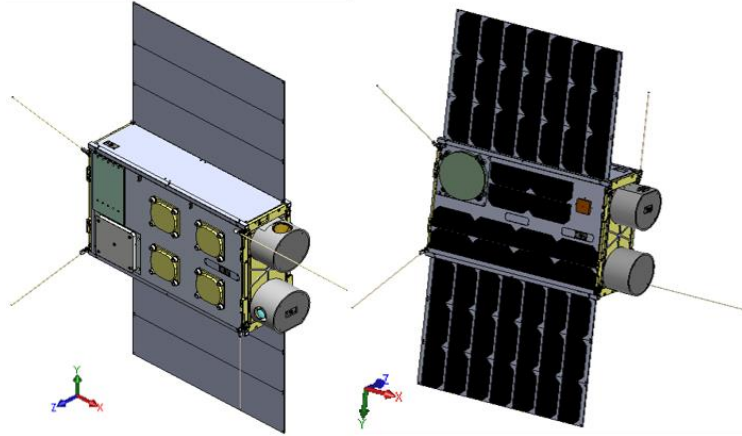


Fig. 1. GNSSaS Spacecraft Reference Frame (S)

### 3. Mathematical modelling

#### 3.1 Kinematic Equations of Motion

Quaternions will be used to represent the attitude of the satellite. The equation that describes the relationship between angular velocity and the derivative of a quaternion can be generated by using the description that is provided in Equation (1) below [14].

$$\dot{q} = \frac{1}{2} \vec{\omega} \otimes q = \frac{1}{2} \Omega(\vec{\omega}) q \quad (1)$$

Where:

$$\Omega(\vec{\omega}) = \begin{bmatrix} 0 & \omega_z & -\omega_y & \omega_x \\ -\omega_z & 0 & \omega_x & \omega_y \\ \omega_y & -\omega_x & 0 & \omega_z \\ -\omega_x & -\omega_y & -\omega_z & 0 \end{bmatrix} \quad (2)$$

$$\vec{\omega} = \begin{bmatrix} \omega_{xo} \\ \omega_{yo} \\ \omega_{zo} \\ 0 \end{bmatrix} \quad (3)$$

$$q = \begin{bmatrix} q_1 \\ q_2 \\ q_3 \\ q_4 \end{bmatrix} \quad (4)$$

Equation (2) represents a matrix representation of the angular velocity vector  $\omega$ , Equation (3) shows the angular velocity vector and Equation (4) represents the quaternion vector. Substituting Equation (2) and (4) into Equation (1), will result in producing the quaternion rate which is important for the kinematic modelling of the CubeSat. It is used to track the orientation of the object over time, the quaternion rate is shown in Equation (5) below [14].

$$\dot{q} = \frac{1}{2} \Omega(\vec{\omega}) q = \frac{1}{2} \begin{bmatrix} 0 & \omega_z & -\omega_y & \omega_x \\ -\omega_z & 0 & \omega_x & \omega_y \\ \omega_y & -\omega_x & 0 & \omega_z \\ -\omega_x & -\omega_y & -\omega_z & 0 \end{bmatrix} \begin{bmatrix} q_1 \\ q_2 \\ q_3 \\ q_4 \end{bmatrix} \quad (5)$$

#### 2.1. Dynamic Equations of Motion

CubeSat's rotating motion can be described using Euler's equations of motion. They are based on the CubeSat properties such as the center of mass (COM), moment of inertia and the mass of the CubeSat. The following equation presents the equation of motion of the CubeSat, Equation (6):

$$J_S \dot{\vec{\omega}}_S^l = -\dot{H}_w - \vec{\omega}_S^l \times (J_S \vec{\omega}_S^l + H_w) + (\vec{\tau}_{grvi} + \vec{\tau}_{rad} + \vec{\tau}_{drag} + \vec{\tau}_{mag}) + \vec{\tau}_{MTR} \quad (6)$$

Where:

$J_S$  is the moment of inertia matrix of the CubeSat,

$\dot{\vec{\omega}}_S^l$  is the first derivative of the angular velocity of the CubeSat expressed in the Earth Centered Inertial frame,

$\dot{H}_w$  is the first derivative of the reaction wheels angular momentum vector,

$\vec{\omega}_S^l$  is the angular velocity of the CubeSat expressed in the Earth Centered Inertial frame,

$H_w$  is the reaction wheels angular momentum vector,

$\vec{\tau}_{grvi}$  is the gravity disturbance torque,

$\vec{\tau}_{rad}$  is the radiation disturbance torque,

$\vec{\tau}_{drag}$  is the drag disturbance torque,

$\vec{\tau}_{mag}$  is the magnetic field disturbance torque,

$\vec{\tau}_{MTR}$  is the magnetorquer control torque.

### 3.2 Orbital Model

The orbital model used in the simulator is based on the Simplified General Perturbations (SGP4) algorithm. This model uses the Two-Line Element (TLE), usually generated by the North American Aerospace Defense Command (NORAD) which contains the orbital parameters. The orbital elements are used in implementing the SGP4 model, described in detail in the following literature [15]. The six orbital parameters are presented below:

$a$  : is the semimajor axis

$e$  : is the eccentricity

$i$  : is the inclination

$\Omega$  : is the right ascension of the ascending node RAAN

$\omega$  : is the argument perigee

$M$  : is the mean anomaly

### 4. Simulation Results

To obtain the intended performance of the ADCS, the multiple modes of operation of the ADCS must be statistically verified. The numerical simulation results of two ADCS modes of operation, nadir pointing and sun pointing, are presented in this section. These numerical simulations are carried out using GOMSpace© ADCS Software which is created with use of the MATLAB/Simulink packet, a figure of the simulator Simulink part is presented in the Figure 2 below.

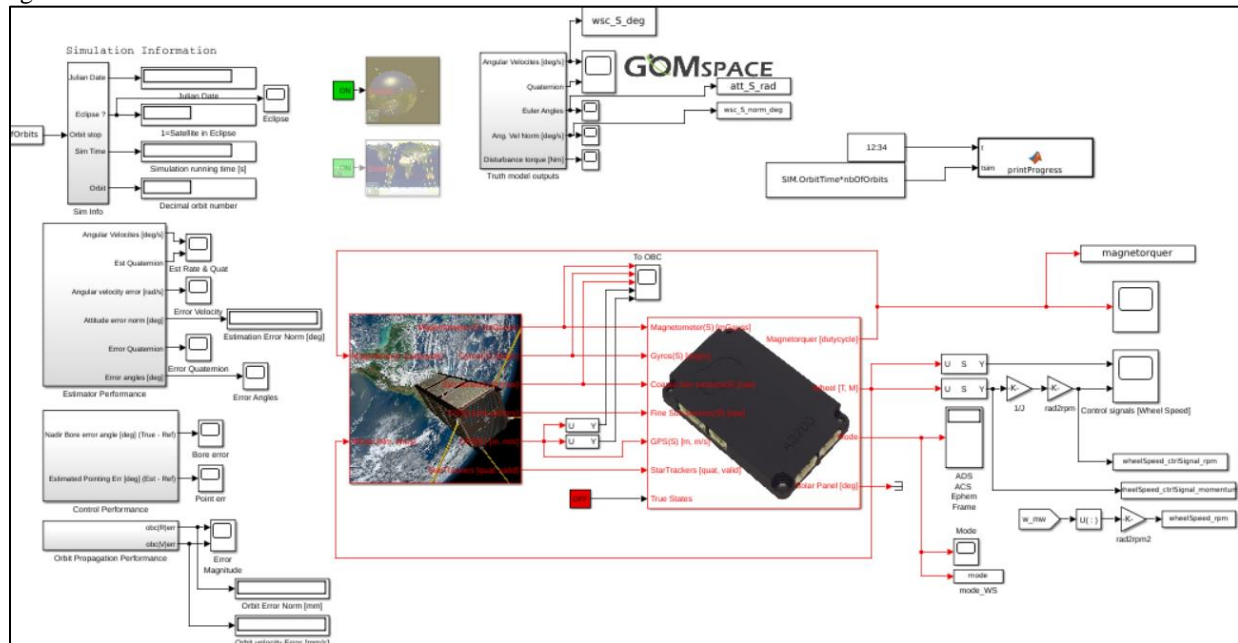


Fig. 2. ADCS GOMSpace© simulator (Simulink) [13]

#### 4.1 Sliding Mode Controller SMC

The sliding mode controller is a robust controller design developed to avoid the mismatch between the actual and desired control model, which can result from unmodelled dynamics or the variation in system parameters. In the GOMSpace© ADCS Simulator library, MW\_SMC is the sliding mode controller parameter, basically the wheel control mode designed to control the momentum of wheels during the pointing modes [13].

#### 4.2 Simulation Results for Nadir Pointing Sub-mode

This simulation test aims to check the pointing performance in Fine Pointing Mode using Nadir pointing sub-mode. All available sensors and actuators were used to get accurate results. Table 2 presents the orbital and simulation parameters used in the simulator. The initial condition of the satellite's angular velocity is 0.1 deg/sec in each direction.

##### 4.2.1 Simulation Parameters

Table 2. Orbital and Simulation Parameters

Parameter	Description
Orbit type	Circular, Sun-Synchronous
Altitude	550 km
Inclination	97.5976 °
RAAN	266.193 °
Number of runs	1
Simulation Step	1 sec
Simulation Time	1 full orbit
Start of the simulation	1 Apr 2021 12:00:00 UTC
Satellite configuration	Deployed

##### 4.2.2 Hardware Model Used

The following sensors and actuators are switched ON for this simulation:

Sensors:

- GNSS Receiver
- Magnetometer
- Gyro
- Star tracker
- Six Fine Sun Sensors
- Coarse Sun Sensor

Actuators:

- 3- axis Magnetorquer
- Four Reaction Wheels

##### 4.2.3 Simulation Results

In this scenario, after detumbling, the satellite constantly changes its attitude to align the +Z axis with the nadir and the +X axis with the velocity vector. During this analysis, each sensor that was available was utilized. The simulation was conducted outside of the eclipse's path.

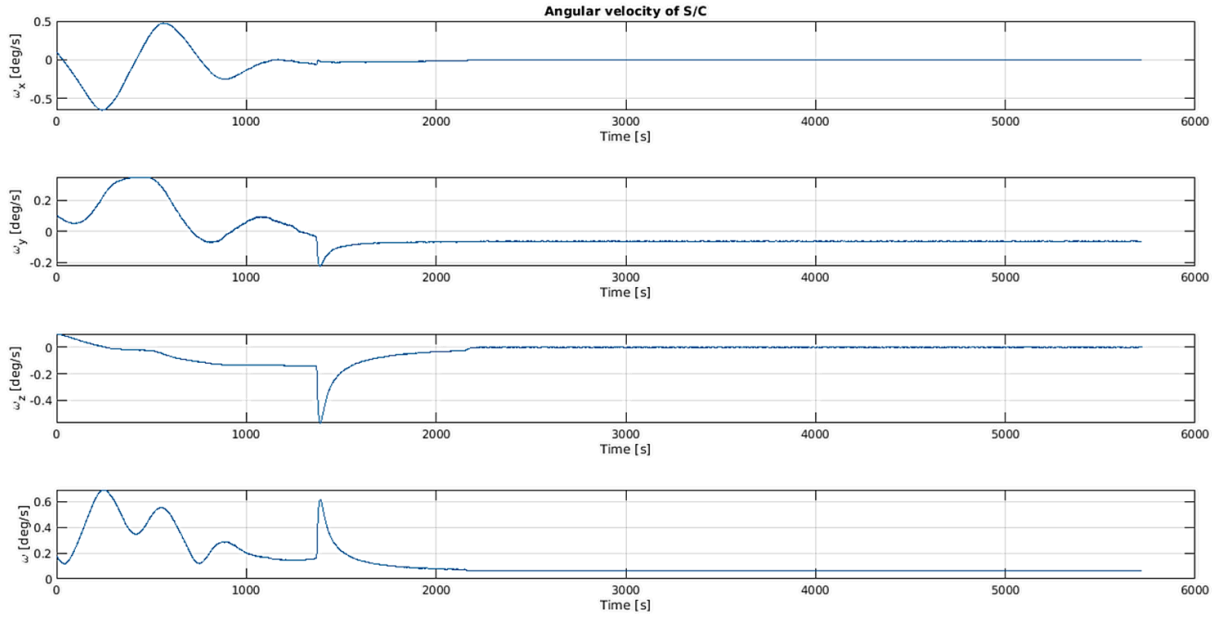


Fig. 3. Angular rate of the spacecraft during Nadir pointing sub-mode (Fine Pointing mode)

Figure 3 presents the angular velocity of the spacecraft throughout the simulation. The initial phase is detumbling. The peak of the angular velocity after detumbling shows the satellite's attitude control in order to orient the +Z axis toward the nadir.

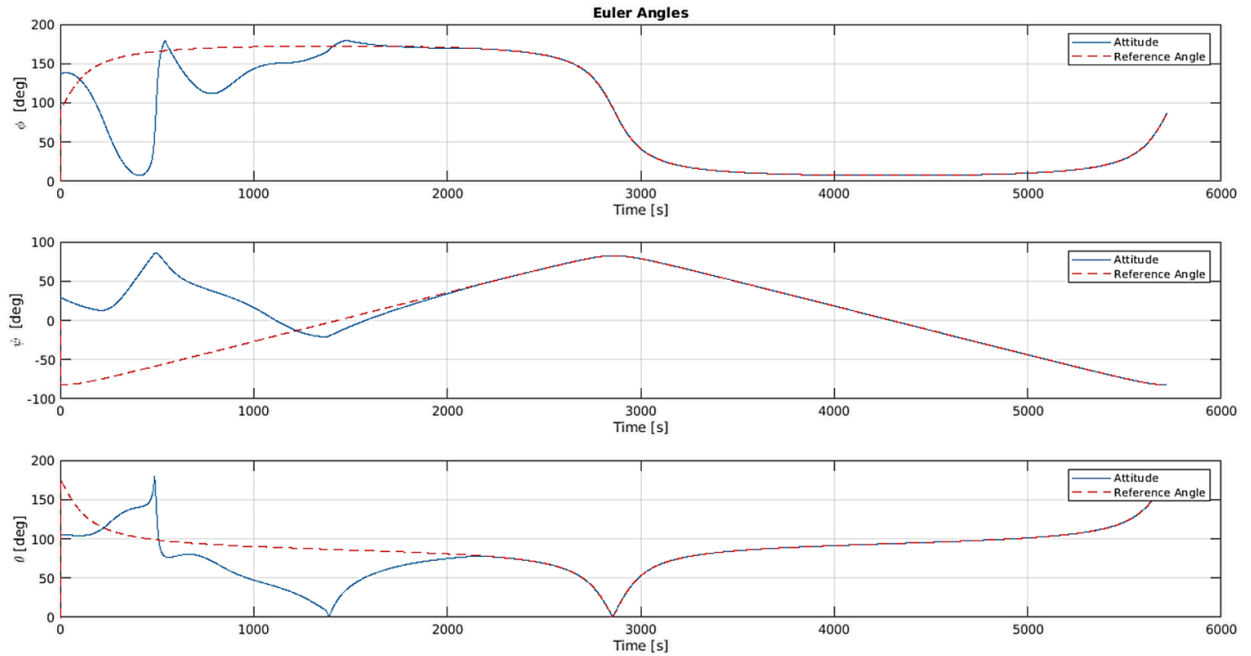


Fig. 4. Euler angles during Nadir pointing sub-mode with respect to ECI frame (Fine Pointing mode)

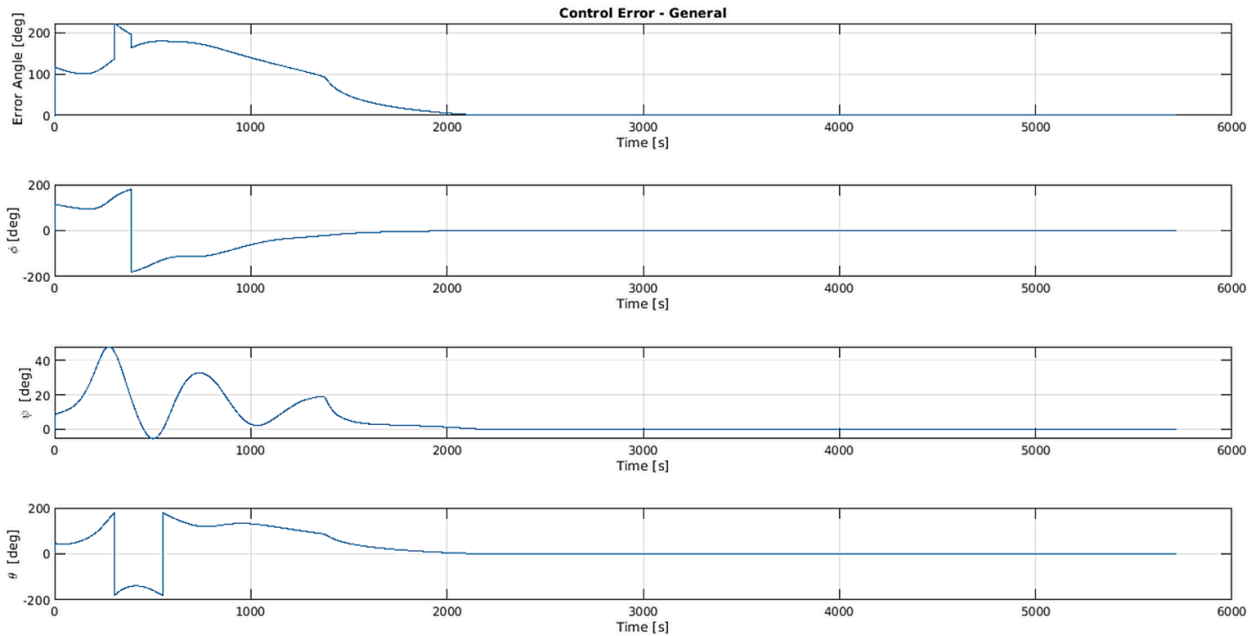


Fig. 5. Error angle during Nadir pointing sub-mode (Fine Pointing mode)

Figure 4 shows the spacecraft attitude angles with respect to the ECI frame. Figure 5 presents the control error during the simulation. Following transitioning from the Detumbling mode to the 3-axis mode, the value of the control error reduces, representing the fact that the satellite's attitude is not controlled in the Detumbling mode and that the satellite is in a random attitude after the detumbling phase.

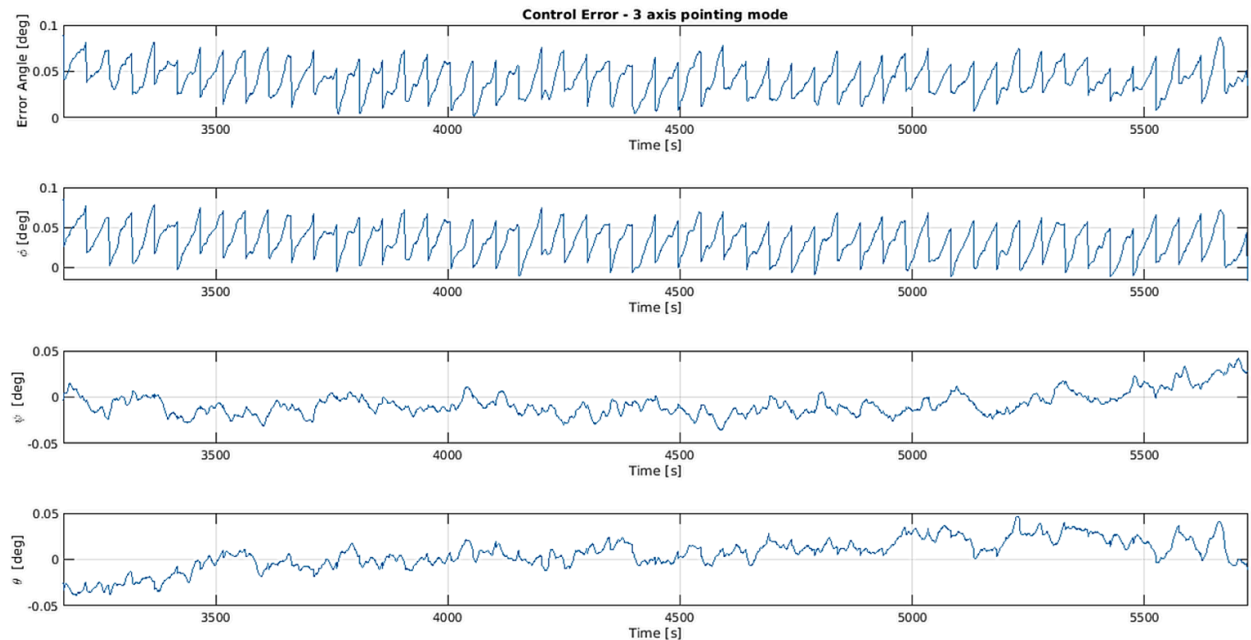


Fig. 6. Error angle after setting correct attitude during Nadir pointing (Fine Pointing mode)

The next figure can be used to evaluate the control performance of the GNSSaS CubeSat. Figure 6 shows the error angle during nadir pointing. The plot begins 30 minutes after the transition from Detumbling mode to 3-axis control mode to allow for the satellite's orientation towards the Nadir. The first plot shows the angle between the reference angle (where the payload must point) and the actual true pointing of the payload axis. The remaining three figures

represent the same parameter as three Euler angles. The error angle is maintained less than 0.1 degree as seen in the figure.

#### 4.3 Fine Pointing Performance - Sun Pointing sub-mode

This simulation test aims to check the pointing performance in Fine Pointing Mode using Sun Pointing sub-mode. All available sensors and actuators were used to get accurate results. Table 3 shows the orbital and simulation parameters used in the simulator. The initial condition of the satellite's angular velocity is 0.1 deg/sec in each direction.

##### 4.3.1 Simulation Parameters

Table 3. Orbital and Simulation Parameters

Parameter	Description
Orbit type	Circular, Sun-Synchronous
Altitude	550 km
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The following sensors and actuators are switched ON for this simulation:

Sensors:

- GNSS Receiver
- Magnetometer
- Gyro
- Star tracker
- Six Fine Sun Sensors
- Coarse Sun Sensor

Actuators:

- 3-axis Magnetorquer
- Four Reaction Wheels

##### 4.3.3 Simulation Results

In this scenario, the satellite, after detumbling, constantly adjusts its attitude in order to point the -Z axis towards the Sun. During this analysis, every available sensor was used. The simulation was done outside the eclipse zone.

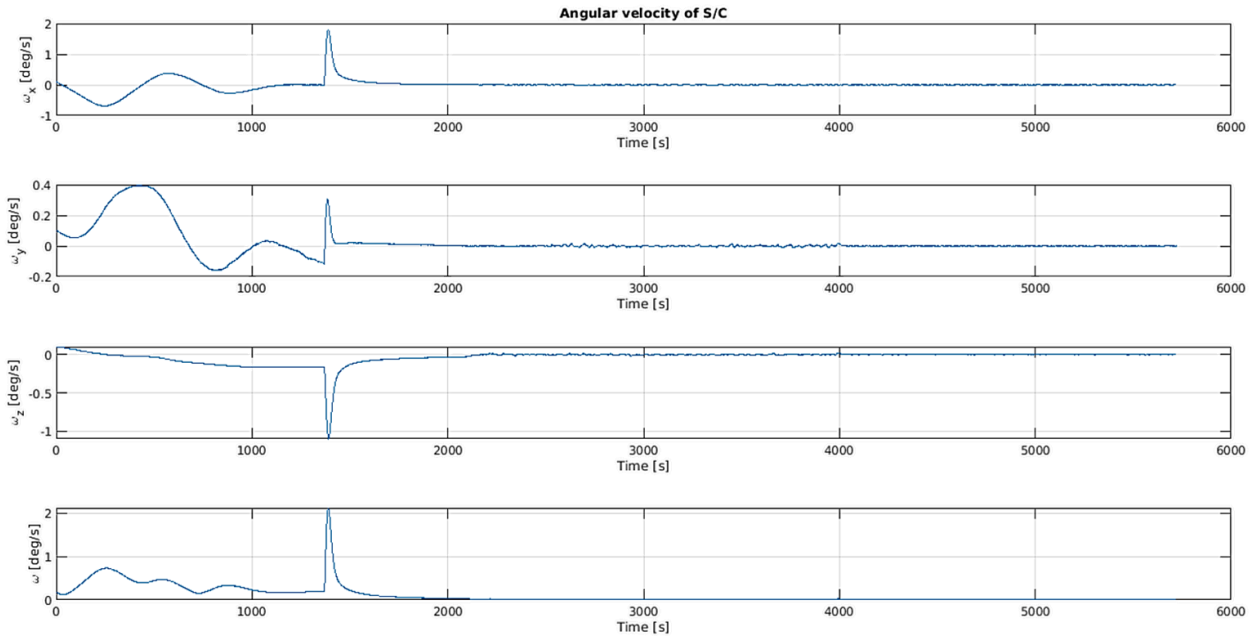


Fig. 7. Angular rate of the spacecraft during Sun pointing sub-mode (Fine Pointing mode)

The Figure 7 presents angular rate of the spacecraft throughout the whole simulation. After detumbling, the peak of the angular velocity represents controlling the satellite's attitude, in order to point +Z axis towards the Sun.

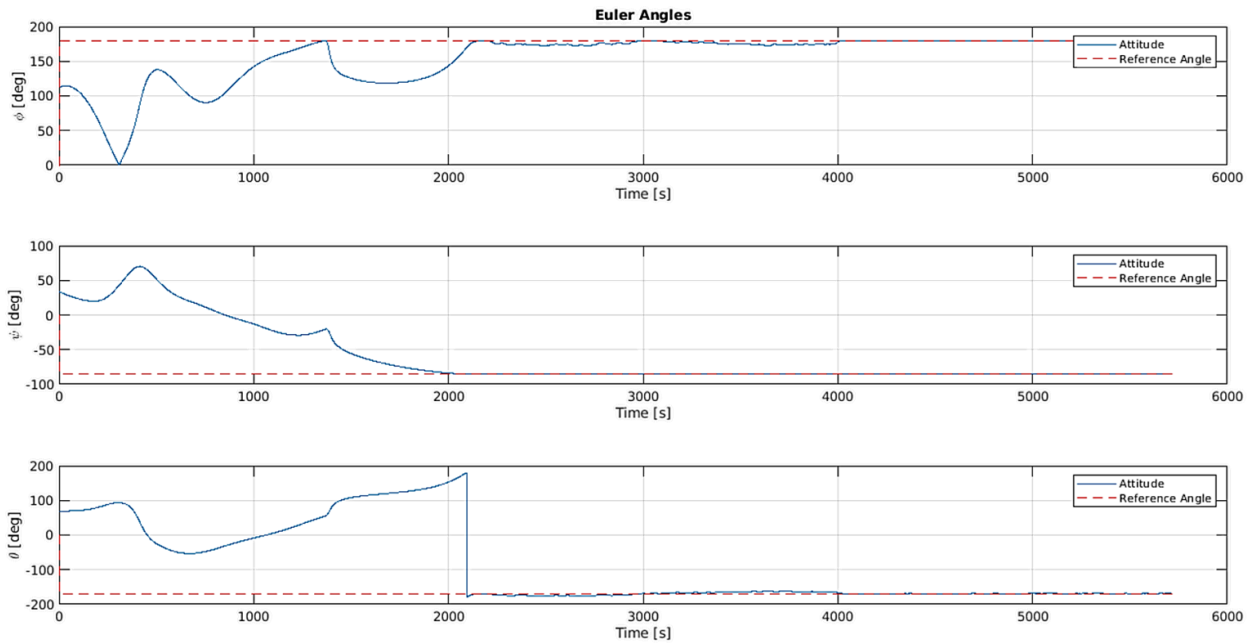


Fig. 8. Euler angles during Sun pointing sub-mode with respect to ECI frame (Fine Pointing mode)

Figure 8 shows the spacecraft attitude angles with respect to the ECI frame.

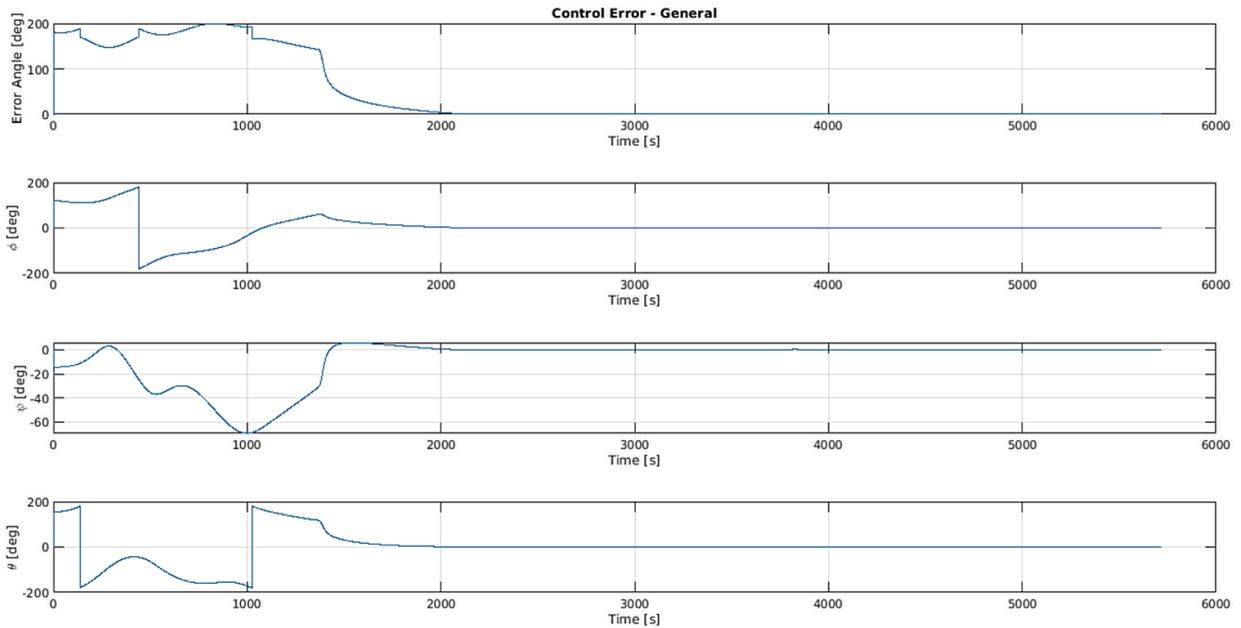


Fig. 9. Error angle during Sun pointing sub-mode (Fine Pointing mode)

Figure 9 presents the control error during the simulation. Value of the control error decreases after switching from the Detumbling mode to the 3-axis mode, which represents the fact that the satellite attitude is not controlled in the Detumbling mode and the satellite is in random attitude after the detumbling phase.

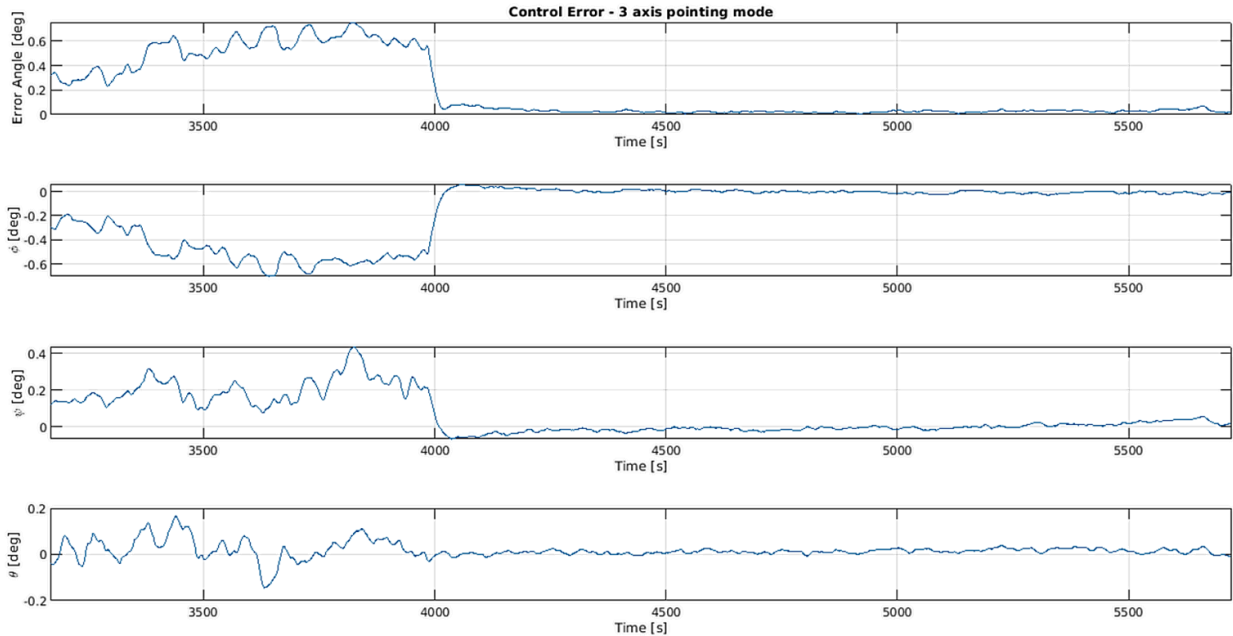


Fig. 10. Error angle after setting correct attitude during Sun pointing sub-mode (Fine Pointing mode)

Figure 10 shows the error angle during simulation, in this figure control performance of the spacecraft can be checked. The plot starts 30 minutes after the change from the Detumbling mode to the 3-axis control mode in order to give the time to for orienting the satellite towards the Sun. The first plot shows the angle between the reference angle (where the payload is required to point) and the actual true pointing of the payload axis. The other 3 plots show the same parameter, but divided into 3 Euler angles.

## 5. Discussion and Conclusions

In the preceding section, numerical simulation results show the performance of the ADCS of the GNSSaS CubeSat in two different modes of operation. The results demonstrate that the ADCS can successfully maintain the antennae pointing towards the Earth in Nadir Pointing sub-mode and the solar panels pointing towards the Sun in Sun Pointing sub-mode. The results indicate that the antennas can maintain a pointing accuracy to the Earth within 0.1 degrees. During this mode, the satellite's angular velocity approaches zero degrees in around 30 minutes.

The performance of the Sun-pointing mode is especially essential since it ensures that the satellite's battery is recharging while the solar panels are facing the Sun. During this mode, the satellite's angular velocity approaches zero degrees in around 25 minutes. The results of this mode indicate that the pointing accuracy is maintained within 0.1 degrees.

The simulation results provide confidence in the ADCS's capacity to complete the mission as planned. It should be emphasized, that these findings are based on simulation, and that additional testing and validation in real-world situations are necessary to prove the ADCS's performance. The numerical simulation results reported in this work show that the ADCS can successfully operate and achieve the intended performance in various modes of operation and provide important insight into its operation. Testing the suggested ADCS on actual hardware is planned for the next steps.

## Acknowledgements

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