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STUDY AND SIMULATION OF THE ATTITUDE DETERMINATION AND CONTROL SUBSYSTEM OF A CUBESAT

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This paper studies the attitude of a CubeSat to optimize its mission life. The paper also presents, discusses and develops a simulator for the Attitude Determination and Control Subsystem (ADCS) of a CubeSat using Simulink. The simulator presented describes the behaviour of a CubeSat during its planned mission, and optimizes the satellite mission lifetime, as well as selects those satellite elements that will allow the best mission results. The simulator includes seven subsystems, each one representing every element of the satellite's ADCS system. Different simulation environments are discussed to demonstrate the actual workability of the simulator.

INTRODUCTION

A preliminary study of the Attitude Determination and Control Subsystem (ADCS) is needed in order to optimize the mission lifetime of a CubeSat. The main objective of the ADCS system is to determine the current attitude and to keep the satellite orientated into the desired attitude. To achieve this goal the ADCS system is divided into two subsystems:

1. Attitude Determination System (ADS), which includes the different sensors and the algorithm to determine the current attitude.

2. Attitude Control System (ACS), which includes the controller and the actuators. This system must control the orientation of the satellite into the desired attitude.

During the satellite mission the ACS has six typical attitude control modes to reach its mission goals. This modes are:[1] 1) Orbit Insertion, 2) Acquisition, 3) On-Station, 4) Slew, 5) Safe and 6) Special. In this study the last three modes are ignored because:

1. Slew is the attitude control mode which reorient the vehicle when required.

2. Safe is the control mode used in emergencies.

3. Special is the control mode for special requirements.

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All of them are considered not to be relevant in our simulation for the ADCS of a CubeSat. Therefore, only two attitude control modes will be taking into account: 1) Detumbling which is the same to Orbit Insertion and 2) Acquisition and Stabilization where Acquisition and On-station are joining together into the same attitude control mode.

Nowadays, many CubeSat have been already launched. Hence, many similarities can be appreciated when doing a comparison between the ADCS subsystem of these CubeSats. The simulator presented in this paper is based on these similarities and tries to be a flexible tool for testing typical ADCS systems of CubeSats, as Figure 1 shows.

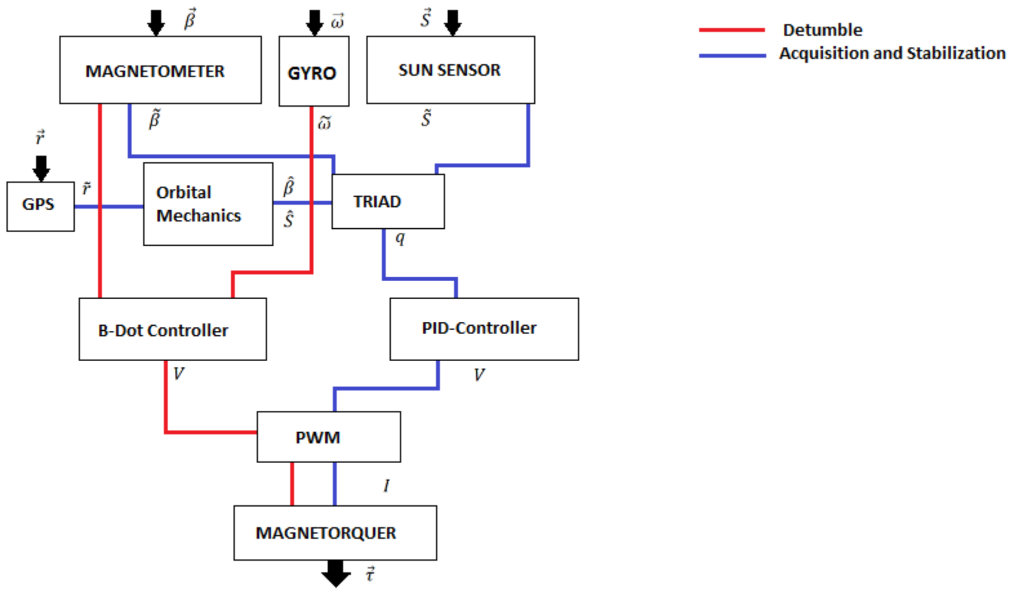

Figure 1. Typical ADCS architecture of CubeSats

The simulator presented here is a tunable simulator where the parameters of the different components of the ADCS can be modified to have a better approximation of the behaviour of the satellite into its orbit. Modelling is based on Low Earth Orbits (LEO) below 500 kilometres.

This paper is divided into three sections.[2] The first Section focuses on the control problem that will be shown together with the mathematical model of the kinematics and the dynamics of the satellite. In the following Section the simulator will be described. Finally, in the last Section the results obtained by the simulation will be discussed in order to validate our tool.

KINEMATICS AND DYNAMICS MODEL

In general CubeSat missions have pointing requirements. Therefore, the orientation of the spacecraft into a known reference frame is needed. Due to some external disturbances the attitude of the satellite may change. Only the aerodynamic drag is taken into account in this study because it is the biggest environmental disturbance acting on the satellite. In this simulator the attitude control can be realized through 3-axis magnetorquers or 3-axis reaction wheels. The following paragraphs describe the mathematical model embedded in the simulator.

Attitude Dynamics

Using the Euler's Moment Equation the torque acting on a spacecraft body can be shown as[3,4]

$$\vec{\tau}^{\,b} = \dot{\vec{h}}^{i} = \dot{\vec{h}}^{b} + \vec{\omega}^{b}_{ib} \times \vec{h}^{b} \tag{1}$$

Using the angular momentum defined as $\vec{h}^{b} = \mathbf{J}\tilde{\omega}^{\mathbf{b}}_{\mathbf{ib}}$. Assuming that the moment of inertia ($\mathbf{J}$) remains constant when it is differentiated:

$$\dot{\vec{h}}^{b} = \mathbf{J}\dot{\tilde{\omega}}^{\mathbf{b}}_{\mathbf{ib}} \tag{2}$$

Combining above equations and using the rotation matrix between the inertial frame and the body frame ($\mathbf{R}^{\mathbf{b}}_{\mathbf{i}}$), the equation for the attitude dynamic is obtained as follows:

$$\dot{\vec{\omega}}^{b}_{ob} = \mathbf{S}(\mathbf{R}^{\mathbf{b}}_{\mathbf{i}}\tilde{\omega}^{\mathbf{i}}_{\mathbf{io}} + \tilde{\omega}^{\mathbf{b}}_{\mathbf{ob}})\mathbf{R}^{\mathbf{b}}_{\mathbf{i}}\tilde{\omega}^{\mathbf{i}}_{\mathbf{io}} - \mathbf{R}^{\mathbf{b}}_{\mathbf{i}}\dot{\tilde{\omega}}^{\mathbf{i}}_{\mathbf{io}} + \mathbf{J^{-1}}(-\mathbf{S}(\mathbf{R}^{\mathbf{b}}_{\mathbf{i}}\tilde{\omega}^{\mathbf{i}}_{\mathbf{io}} + \tilde{\omega}^{\mathbf{b}}_{\mathbf{ob}})\mathbf{J}(\mathbf{R}^{\mathbf{b}}_{\mathbf{i}}\tilde{\omega}^{\mathbf{i}}_{\mathbf{io}} + \tilde{\omega}^{\mathbf{b}}_{\mathbf{ob}}) + \tilde{\tau}^{\,\mathbf{b}}) \tag{3}$$

Where:

$\mathbf{S}$ is the skew matrix.

$\vec{\tau}^{\,b} = \vec{\tau}_a^{\,b} + \vec{\tau}_d^{\,b}$ is the total torque acting on the satellite with $\vec{\tau}_a^{\,b}$ as actuator torque and $\vec{\tau}_d^{\,b}$ as disturbance torque.

Attitude Kinematics

The attitude kinematics represents the orientation of the satellite in a known reference frame. The attitude is usually represented by quaternions. The differentiation quaternion gives the attitude kinematics which is the change in the satellite attitude:[3,5]

$$\dot{q} = \begin{bmatrix} \dot{\eta} \\ \dot{\vec{\epsilon}} \end{bmatrix} = \frac{1}{2}\begin{bmatrix} -\vec{\epsilon}^{\,T}\vec{\omega}^{b}_{ob} \\ (\eta\mathbf{I} + \mathbf{S}(\tilde{\epsilon})\tilde{\omega}^{\mathbf{b}}_{\mathbf{ob}}) \end{bmatrix} \tag{4}$$

Where:

$\mathbf{I}$ is the 3-by-3 Identity Matrix.

$\mathbf{S}(\tilde{\epsilon})$ is the skew symmetric matrix.

Magnetorquers

The magnetic torquers are used to generate a local magnetic dipole moment for attitude and angular momentum control. The torque produced by a magnetorquer is the cross product between the magnetic moment of the magnetic dipole and the Earth's magnetic field:[4]

$$\vec{\tau}_a^{\,b} = \vec{m}^{b} \times \vec{B}^{b} \tag{5}$$

Reaction Wheels

Reaction wheels work on the principle of Newton's third law. Therefore, the angular momentum produced by the spin rate of the wheel must be transferred to the spacecraft with opposite sign:

$$\dot{\vec{h}}^i = -\dot{\vec{h}}^i_w \tag{6}$$

In this study a 3-axis reaction wheels system is considered. This means that the reaction wheels will have their spin axes not coplanar in order to have a 3-axis attitude control.

SIMULATOR

The simulator is composed by two different Simulink models. These models represent the detumbling control mode and the acquisition and stabilization control mode. In this Section, both models are described.

Acquisition and Stabilization Model

In this model the simulator tries both to acquire a predefined desired attitude and to stabilize the CubeSat into this attitude during the lifetime mission.

The model is composed by different subsystems, which are drawn in Figure 2:

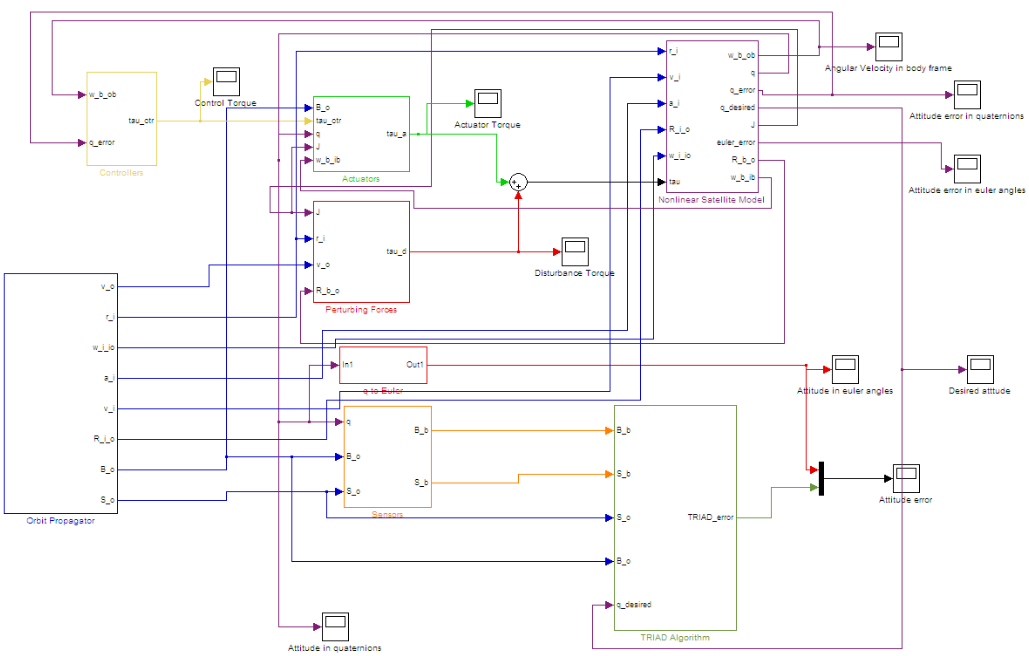

Figure 2. Simulink Model for Acquisition and Stabilization Control Mode

1. **Orbital Propagator:** This subsystem provides the theoretical reference vectors for the attitude determination algorithm. It also provides the orbital mechanics vectors needed to determine and control the spacecraft orientation.

The inputs used in this subsystem, and therefore in the whole simulator, are the Kepler's elements provided by an onffline orbit propagator. This propagator is the SGP4 offered by Vallado.[6] The SGP4 is run in "verification mode" where the start, stop and delta time can be chosen. In this simulator, the orbit is propagated during ninety days with a delta time of one minute.

The theoretical reference vectors calculated are the Earth's Magnetic Field Vector and the Sun Vector. The first one is calculated by a simple dipole model. The Sun Vector is calculated using the Julian Date model.

2. **Sensors:** The sensors block provides the measured vectors which will be used in the algorithm in order to determine the attitude. In this simulator only two sensors are modelled: the magnetometer and the sun sensor.

   The sensors provide a measured vector from a reference. Therefore, if the sensor works properly, it will provide the theoretical vector in the body frame, adding an error because of the misalignment. The sensors have an attitude error delimited by the datasheet of the commercial element. This means that each axis of the vector measured by the sensors have a misalignment from the real direction of the vector. These measurement errors can be modified in the subsystem mask.

3. **TRIAD Algorithm:** This block contains the TRIAD Algorithm.[7] This subsystem also contains a comparison between the estimate attitude offered by the algorithm and the desired attitude. Therefore, it gives an estimate error of the attitude.

4. **Controllers:** This subsystem is made up by three different controllers. The controllers shall control the angular velocity of the satellite and the attitude in quaternions. The controller can be chosen by a pop-up selector in the subsystem mask.

   Two different types of PD-Controllers and a PID-Controller have been implemented in this Simulink block: 1) The first PD-Controller implemented was the simplest controller.[8] This controller is based on the attitude error of the satellite and on the angular velocity of the satellite in the body frame. 2) The second PD-Controller proposed is based on the controller presented by Yuichi Ikeda, Takashi Kida and Tomoyuki Nagashio.[9] The difference between both PD-Controllers is that one of them uses matrices as constants instead of real numbers. 3) Finally, the PID-Controller was proposed to be a more flexible simulator. The PID-Controller is based also on the work done by Ikeda's group.[9]

5. **Actuators:** The Actuators subsystem contains two different 3-axis actuators: magnetorquers and reaction wheels. The actuator receives the control signal from the controller to produce a torque which changes the satellite attitude. The user can chose through a pop-up the actuator he wants to use in their simulation. The values of the maximum dipole moment and saturation of the reaction wheels can be changed according to the datasheet of the selected component.

6. **Perturbing Forces:** This subsystem only contains the aerodynamic drag disturbance because below 500 km of altitude it is the higher one,[10] like Figure 3 shows. Therefore, the other ones can be ignored.

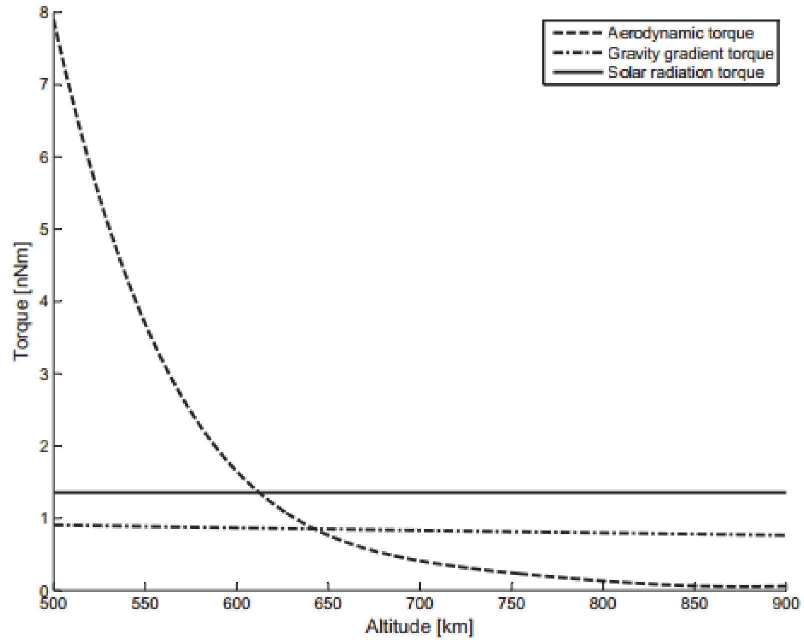

Figure 3. Environmental Disturbances Acting on Attitude[10]

The aerodynamic drag is modelled through the Exponential Model of the atmosphere.[6] The parameters of the satellite which affect this disturbance, like the cross sectional area, the mass of the satellite and the displacement between the centre of pressure and the centre of mass, can be modified.

7. **Nonlinear Satellite Model:** In order to simulate a complete system, a model of the real satellite is needed. The spacecraft is modelled using the theoretical attitude dynamics and attitude kinematics presented before. In addition to the tunable values of the whole simulator, the inertia matrix of the satellite can be modified in this subsystem.

   The most relevant outputs of the simulator are the attitude measured in Euler Angles and the angular velocity in the body frame. These are the outputs of this subsystem.

Detumbling Model

The Detumbling model shall reduce the angular velocity of the spacecraft. Therefore, the final angular velocity after the Detumbling control mode must be close to zero in each axis. The Simulink model can be found in Figure 4.

The model is divided into six different subsystems.

1. Orbital Mechanics
2. Sensors
3. B-Dot Controller
4. Actuators

5. Perturbing Forces
6. Nonlinear Satellite Model

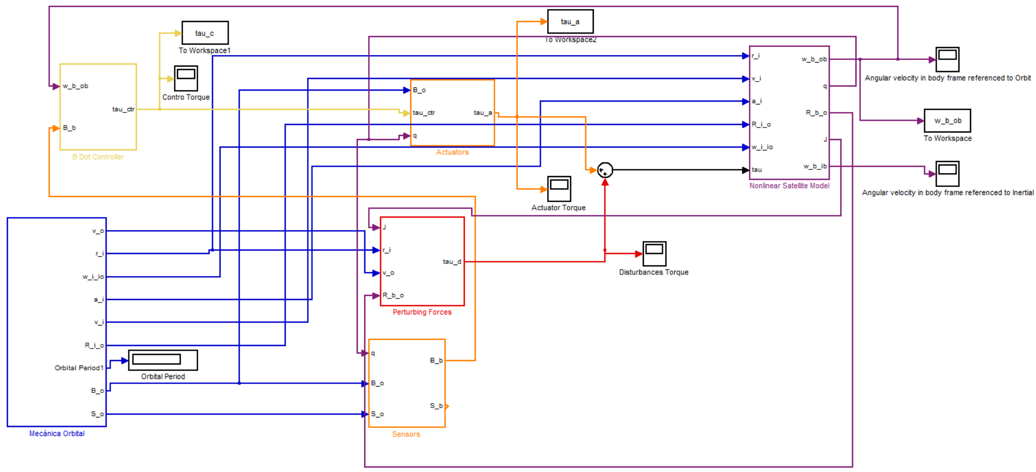

Figure 4. Simulink model for Detumbling Control Mode

This model has some differences from the Acquisition and Stabilization model:

Firstly, the Orbital Mechanics does not use the orbital parameters provided by the off-line SGP4 propagator. Instead, it uses the orbital parameters which define the orbit. These parameters can be modified in the subsystem mask.

Secondly, only one controller can be chosen, because the B-Dot control law is suited for the detumbling purpose.[11] In consequence with many detumbling studies, 3-axis magnetorquers were selected as actuator for this control mode.

Finally, the attitude determination algorithm is not needed in this model because the purpose of it is to reduce the spin rate of the satellite.

SIMULATION RESULTS

In order to illustrate the functionality of this simulator three different simulations have been carried out. It is important to note that the three simulations are based on the same orbit. This orbit corresponds to the FITSAT-1 orbit, propagated by Vallado's orbit propagator. The TLE used to be propagated can be found in Figure 5.

```
#FITSAT-1 (NIWAKA)
1 38853U 98067CP   13118.78074884  .00125286  00000-0  81581-3 0  2103
2 38853  98.0000 339.4804 0000000 200.5119 159.5389 15.74921680 32179     0.0      129600.0       1.000
```

Figure 5. FITSAT-1 TLE

Detumbling

As we said before, the goal for this control mode is to decrease the spin rate of the satellite to zero. Using the B-Dot Controller in addition to the 3-axis magnetorquer, with a limitation of its magnetic moment and the 3-axis magnetometer with errors by misalignment, the angular velocity in the body frame should be close to zero in the 3 body axes.

Taking into account the displacement between the centre of pressure and the centre of mass as 1 centimetre in the 3 axes, we have the following characteristics shown in Table 1.

Table 1. Values for Detumbling Simulation

Parameter	Value
CubeSat Type	2U
Inertia Matrix (**J**)	$\begin{bmatrix} \frac{m(b^2+c^2)}{12} & 0 & 0 \\ 0 & \frac{m(a^2+c^2)}{12} & 0 \\ 0 & 0 & \frac{m(a^2+b^2)}{12} \end{bmatrix}$
Dimensions	$a = b = 10cm$; $c = 20cm$
Satellite Mass	2 Kg
Coefficient Drag	2.2
Cross Sectional Area	0.015 m^2
Initial $\vec{\omega}^b_{ob}$ (rad/sec)	$[1 \quad 2 \quad 3]^T$
B-Dot Constant	10900
Magnetic Moment	0.3 Am^2
Misalignment	± 0.37 Deg
Simulation Step	$\Delta t = 1$ sec

With these values, the simulation provides the following output of the angular velocity in the body frame (Figure 6):

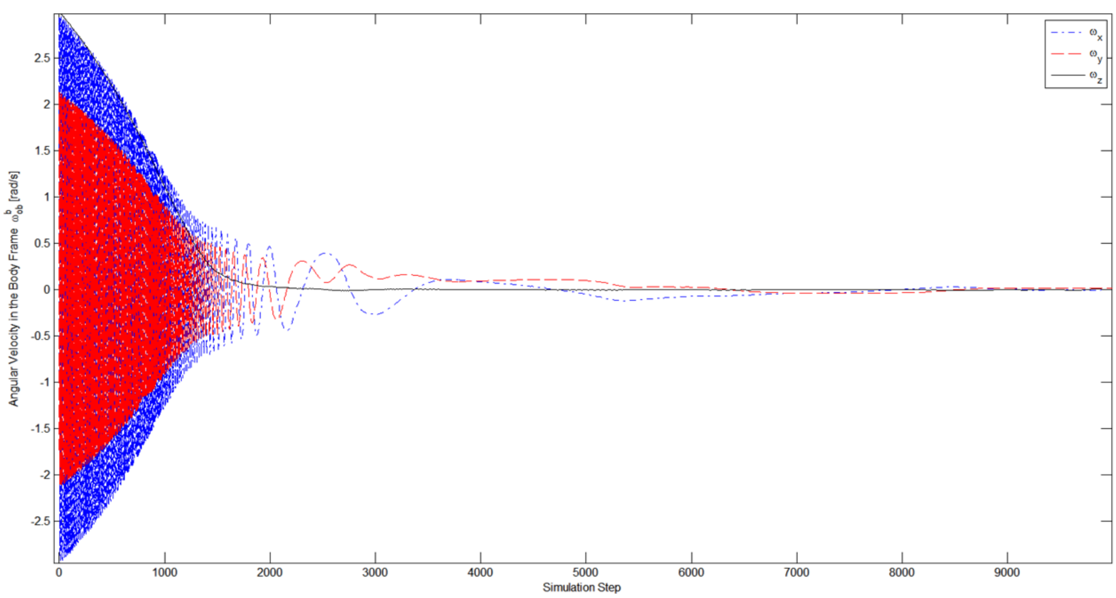

Figure 6. Detumbling Result

Analysing Figure 6, it can be appreciated that after 7,500 seconds the angular velocity is practically equal to zero in the three axes.

Case 1

The cases are separated by the different ACS system chosen in the simulation. In this first case the architecture of the ADCS system is as follows:

1. Controller: PID-Controller the following control constants presented in Table 2.

Table 2. Values for PID-Controller Constants Simulation Case 1

Control Constant	Value
$\mathbf{K_{p2}}$	$10\mathbf{I}$
k_{p3}	12
$\mathbf{K_{d2}}$	$40\mathbf{I}$
b_2	0.1
k_{i2}	0.1

2. Actuator: 3-axis magnetorquer with maximum dipole moment as $0.3\ Am^2$.

3. Sensors: 3-axis magnetometer (± 0.37 [Deg] as misalignment error) plus 3-axis sun sensor (± 0.01 [Deg] as misalignment error).

4. TRIAD Algorithm.

Knowing the ADCS system used, the simulation parameters are presented in Table 3:

Table 3. Values for Acquisition and Stabilization Simulation Case 1

Parameter	Value
CubeSat Type	2U
Inertia Matrix ($\mathbf{J}$)	$\begin{bmatrix} \frac{m(b^2+c^2)}{12} & 0 & 0 \\ 0 & \frac{m(a^2+c^2)}{12} & 0 \\ 0 & 0 & \frac{m(a^2+b^2)}{12} \end{bmatrix}$
Dimensions	$a = b = 10cm$; $c = 20cm$
Satellite Mass	2 Kg
Coefficient Drag	2.2
Cross Sectional Area	0.015 m^2
Initial $\vec{\omega}_{ob}^b$ (rad/sec)	$[0 \quad 0 \quad 0]^T$
Initial Attitude in Euler Angles (Deg)	$[0 \quad 0 \quad 0]^T$
Desired Attitude in Euler Angles (Deg)	$[0 \quad 60 \quad 0]^T$
Simulation Step	$\Delta t = 1$ (min)
Simulation Time	129600 (min)

The simulation time and the simulation step come from the delta time and stop time given in the verification TLE file, to be consequent with the results obtained with the propagator.

Running the simulation with these values, results are shown in Figure 7:

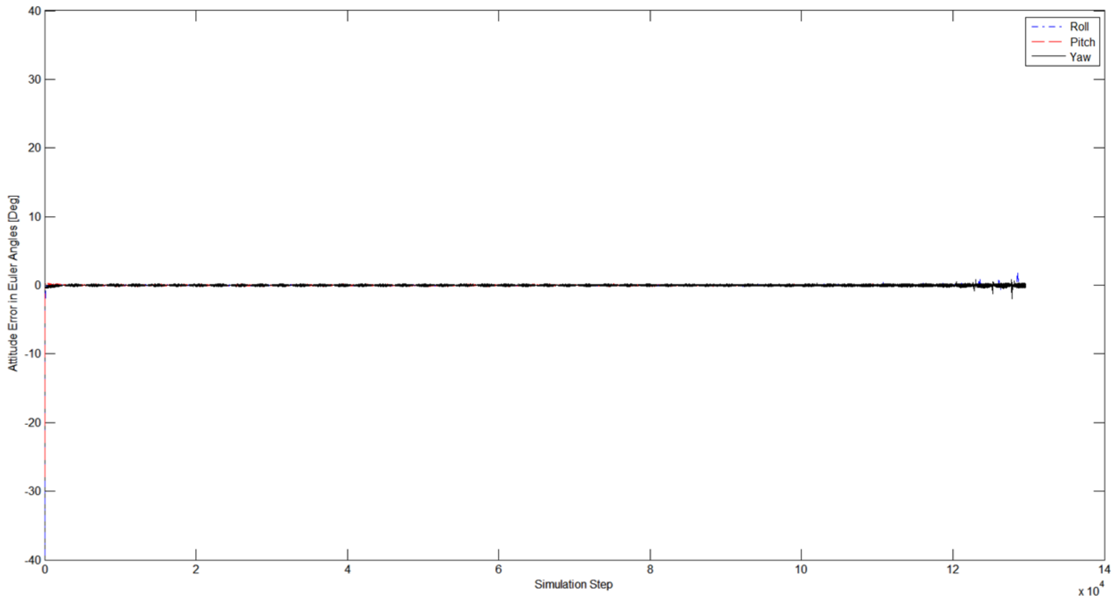

Figure 7. Attitude Error in Euler Angles Case 1

Zooming to view the time needed to acquire the desired attitude is shown in Figure 8:

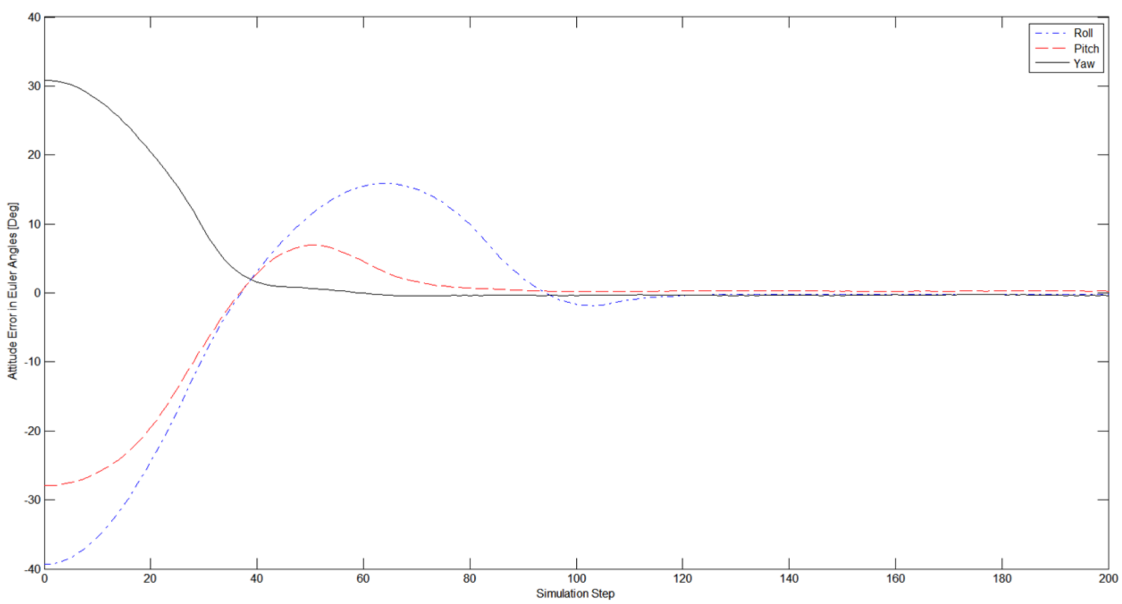

Figure 8. Acquisition Time to Get the Desired Attitude (Attitude Error in Euler Angles) Case 1

Case 2

This second case considers a different architecture of the ADCS system to illustrate the differences between this one and the previous one.

1. Controller: PD-Controller with the following control constants shown in Table 4.

Table 4. Values for PD-Controller Constants Simulation Case 2

Control Constant	Value
k_p	0.6
k_d	2

2. Actuator: 2-axis magnetorquer with maximum dipole moment as 0.3 Am^2.

3. Sensors: 3-axis magnetometer (± 0.37 [Deg] as misalignment error) plus 3-axis sun sensor (± 0.01 [Deg] as misalignment error).

4. TRIAD Algorithm.

With these characteristics values are shown in Table 5:

Table 5. Values for Acquisition and Stabilization Simulation Case 2

Parameter	Value
CubeSat Type	1U
Inertia Matrix ($\mathbf{J}$)	$\begin{bmatrix} \frac{m(b^2+c^2)}{12} & 0 & 0 \\ 0 & \frac{m(a^2+c^2)}{12} & 0 \\ 0 & 0 & \frac{m(a^2+b^2)}{12} \end{bmatrix}$
Dimensions	$a = b = c = 10cm$
Satellite Mass	1 Kg
Coefficient Drag	2.2
Cross Sectional Area	0.01 m^2
Initial $\vec{\omega}_{ob}^b$ (rad/sec)	$[0 \quad 0 \quad 0]^T$
Initial Attitude in Euler Angles (Deg)	$[15 \quad -25 \quad 30]^T$
Desired Attitude in Euler Angles (Deg)	$[0 \quad 0 \quad 0]^T$
Simulation Step	$\Delta t = 1$ (min)
Simulation Time	129600 (min)

The following Figure 9 represents the attitude error measured in Euler Angles:

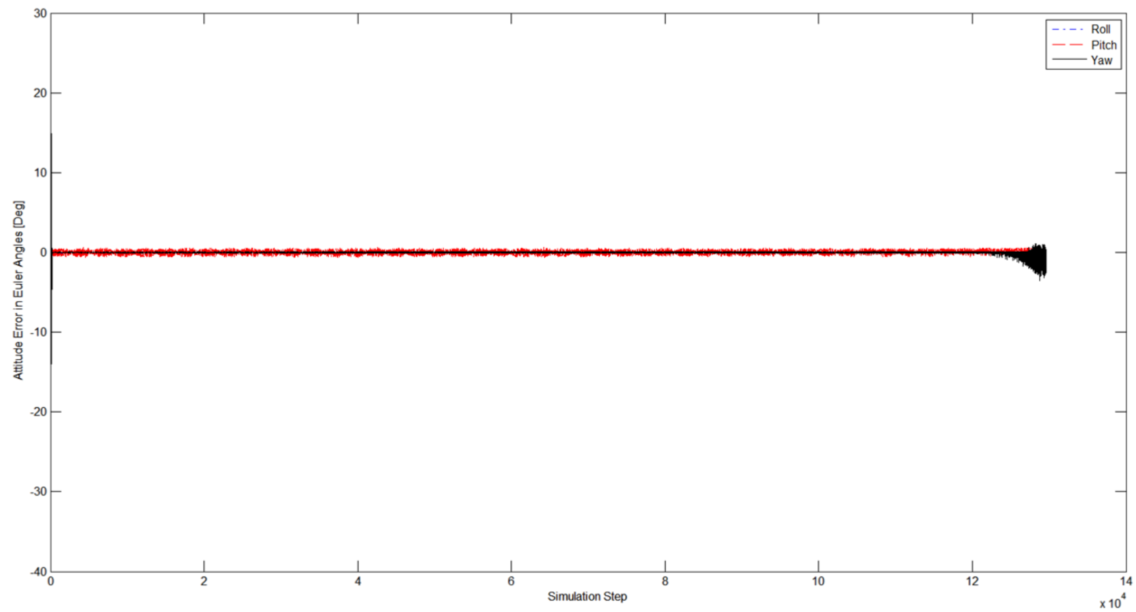

Figure 9. Attitude Error in Euler Angles Case 2

Zooming to illustrate the time needed to acquire the desired attitude is as follows (Figure 10):

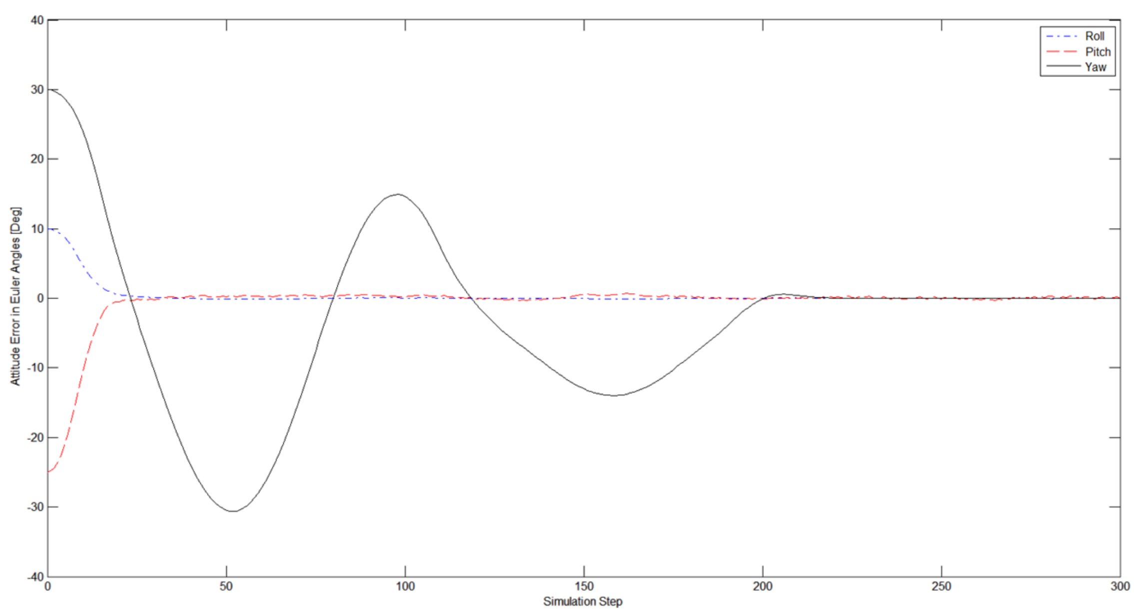

Figure 10. Acquisition Time to Get the Desired Attitude (Attitude Error in Euler Angles) Case 2

Case 3

Finally a third case is considered to illustrate a different actuator acting on this simulator:

1. Controller: PID-Controller the following control constants presented in Table 6.

Table 6. Values for PID-Controller Constants Simulation Case 3

Control Constant	Value
$\mathbf{K_{p2}}$	$10\mathbf{I}$
k_{p3}	12
$\mathbf{K_{d2}}$	$40\mathbf{I}$
b_2	0.1
k_{i2}	0.1

2. Actuator: 3-axis reaction wheels with maximum wheel speed as 5000 rpm and moment of inertia as 0.0008 Kgm^2.

3. Sensors: 3-axis magnetometer (± 0.37 [Deg] as misalignment error) plus 3-axis sun sensor (± 0.01 [Deg] as misalignment error).

4. TRIAD Algorithm.

Knowing the ADCS system used, the simulation parameters are shown in Table 7:

Table 7. Values for Acquisition and Stabilization Simulation Case 3

Parameter	Value
CubeSat Type	3U
Inertia Matrix ($\mathbf{J}$)	$\begin{bmatrix} \frac{m(b^2+c^2)}{12} & 0 & 0 \\ 0 & \frac{m(a^2+c^2)}{12} & 0 \\ 0 & 0 & \frac{m(a^2+b^2)}{12} \end{bmatrix}$
Dimensions	$a = b = 10cm$; $c = 30cm$
Satellite Mass	3 Kg
Coefficient Drag	2.2
Cross Sectional Area	0.02 m^2
Initial $\vec{\omega}^b_{ob}$ (rad/sec)	$[0 \quad 0 \quad 0]^T$
Initial Attitude in Euler Angles (Deg)	$[10 \quad -25 \quad 30]^T$
Desired Attitude in Euler Angles (Deg)	$[25 \quad 10 \quad 120]^T$
Simulation Step	$\Delta t = 1$ (min)
Simulation Time	129600 (min)

The simulation time and simulation step come from the delta time and stop time given in the verification TLE file, to be consequent with the results obtained with the propagator.

Running the simulation with these values, results are shown in Figure 11:

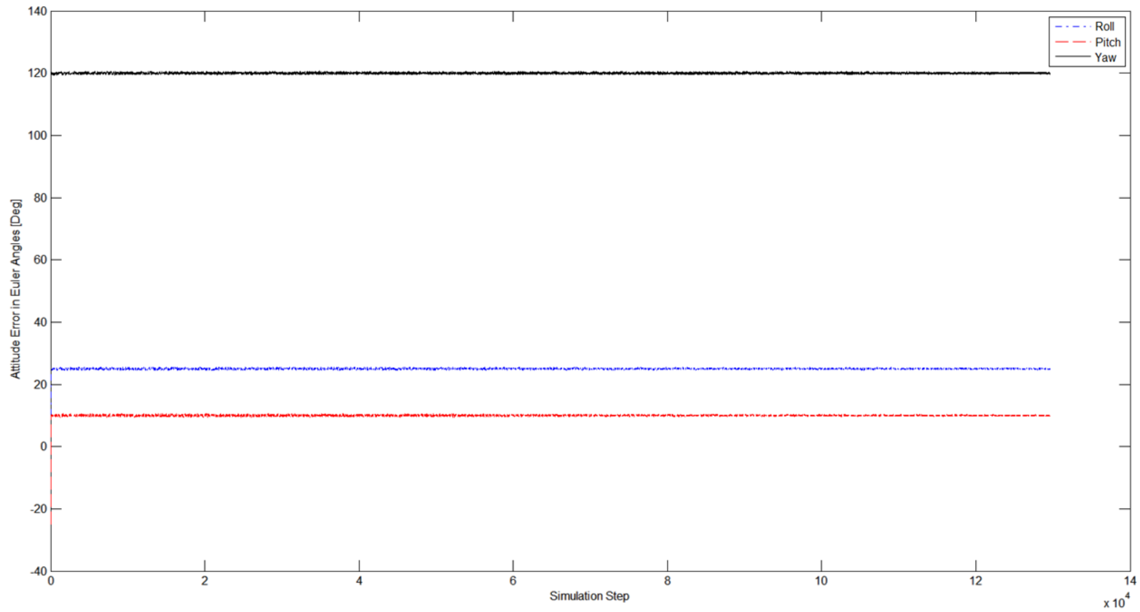

Figure 11. Attitude in Euler Angles Case 3

Zooming to view the time needed to acquire the desired attitude is shown in Figure 12:

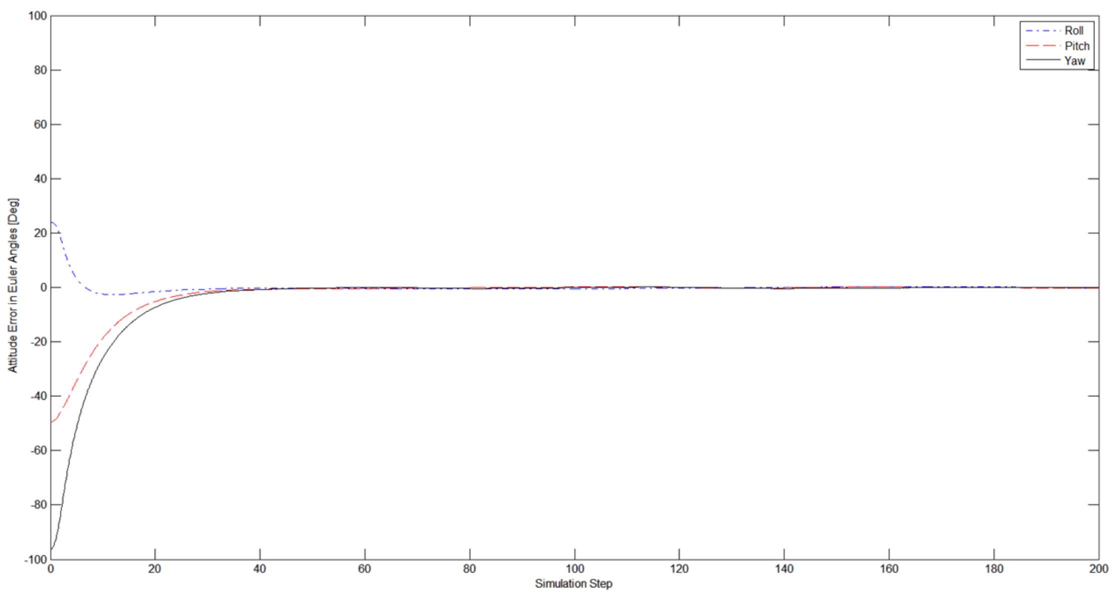

Figure 12. Acquisition Time to Get the Desired Attitude (Attitude Error in Euler Angles) Case 3

CONCLUSION

The simulator presented in this paper is mainly oriented to CubeSats orbiting in LEO orbits below 500 kilometres. Different simulations have been carried out to help study the behaviour of the satellite during its lifetime mission. To carry out this study it is important to know the TLE file of the satellite or to know the TLE file of a CubeSat orbiting similar orbits. The importance of this knowledge is to propagate the orbit with the SGP4 propagator offered by Vallado.[6] The simulations also show that this tool is a good first approximation to study the possibilities for ACS systems with a defined ADS system, which is based on a 3-axis sun sensor, a 3-axis magnetometer and a TRIAD algorithm. A potential future line of work could be to increase the features of the ADS system of the simulator in order to provide more options for an integrated ADCS system.

NOTATION

$\vec{\tau}^{\,b}$ a vector

$\dot{\vec{h}}^{i}$ a time differentiated vector

$\mathbf{J}$ a matrix

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