This paper describes the results of the initial analysis of the pointing accuracy of a nanosatellite of the 6U (34×22×10cm) CubeSat class. The Application for RSO Proximity Analysis and IMAnaging (ARAPAIMA) mission is designed to perform in-orbit demonstrations of autonomous proximity operations for visible, infrared, and three-dimensional imaging of resident space objects (RSOs). ARAPAIMA will fly a payload consisting of a commercial off-the-shelf (COTS) infrared (IR) camera and a reaction wheel. The star tracker (STR) from the attitude determination and control system (ADCS) is also used as a payload instrument for imaging in the visible spectrum. The satellite’s actuators include a reaction control system (RCS) comprised of a three-axis RCS thruster and reaction wheel array together with a single orbital maneuvering thruster. The star tracker is complemented by an Inertial Measurement Unit (IMU) and a GPS unit. The satellite’s ability to reject external and internal disturbance torques while maintaining a pointing position relative to a stationary object is tested utilizing only the reaction wheel triad.
Analysis of the Pointing Accuracy of a 6U CubeSat Mission for Proximity Operations and Resident Space Object Imaging

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This paper describes the results of the initial analysis of the pointing accuracy of a nanosatellite of the 6U (34×22×10cm) CubeSat class. The Application for RSO Proximity Analysis and IMAgeing (ARAPAIMA) mission is designed to perform in-orbit demonstrations of autonomous proximity operations for visible, infrared, and three-dimensional imaging of resident space objects (RSOs). ARAPAIMA will fly a payload consisting of a commercial off-the-shelf (COTS) infrared (IR) camera and miniature laser rangefinder. The star tracker (STR) from the attitude determination and control system (ADCS) is also used as a payload instrument for imaging in the visible spectrum. The satellite’s actuators include a reaction control system (RCS) comprised of a three-axis RCS thruster and reaction wheel array together with a single orbital maneuvering thruster. The star tracker is complemented by an Inertial Measurement Unit (IMU) and a GPS unit. The satellite’s ability to reject external and internal disturbance torques while maintaining a pointing position relative to a stationary object is tested utilizing only the reaction wheel triad.

1. Introduction

The goal of the ARAPAIMA mission is to perform the in-orbit demonstration of autonomous proximity operations for visible, infrared, and three dimensional imaging of resident space objects (RSOs). The payload consists of a commercially available IR camera and a miniature laser rangefinder with a range of a few km. The star tracker from the attitude determination and control system is used as a visible light camera. The three instruments are installed on the nanosat so that their optical axes are pointing in the same direction. The nanosat is equipped with a warm gas propulsion system which enables it to perform orbital maneuvers and reaction attitude control. Further details on the satellite subsystems are presented in [11]

The goal of the mission is achieved in five steps of increasing complexity. During the first two steps the nanosat uses pre-loaded and ground initiated commands to maneuver within rangefinder range of the RSO and acquire a relative circular orbit with respect to it. The third step consists of the nanosat using constraint-based algorithms

\dagger The views expressed in this article are those of the author in the capacity of a private person and do not necessarily express the views of and should not be attributed to ESA.
and visual navigation to maneuver autonomously to reduce the size of the relative orbit to a few hundred meters. The fourth step involves detailed visible and IR imaging of the RSO from multiple relative orbit profiles [7]. During the fifth step a combination of chaser attitude motion and relative motion between the nanosat and the RSO is employed to perform 3D imaging by combining rangefinder measurements and knowledge of the nanosat inertial attitude and position [6].

The approach to the design of the ARAPAIMA mission closely follows that of a fully fledged science mission. The basic concept has been developed into the design of the experiment and initial simulations of the mission have shown encouraging results. The subsequent steps are 1) implementing the experiment; 2) executing it; 3) extracting the data products; 4) and interpreting the results. However, unlike a fully fledged space mission only a few of the mission level requirements have been flown down to system and subsystem level. The reason for the partial flow down of the requirements comes from the practical aspects of developing a space mission as a student project and within the budget and physical size constraints of a university built and operated mission. As a consequence, parts of the mission are developed in a bottom-up approach incorporating simulation, design, integration and testing of components and subsystems, with checks of their performance against an ideal mission model.

A CubeSat class satellite with this level of expected complexity and maneuverability is unprecedented in the field. This mission’s success will expand the expected range of capabilities of small satellites and establish a standard of performance hitherto unexpected without compromising the initial idea of universal feasibility. As a pioneer on this front, ARAPAIMA’s ADCS utilizes COTS components combined with an experienced knowledge of mission management and design, and complex system dynamics which will result in capabilities limited only by the satellite’s lifespan.

2. Mission Design

The ARAPAIMA mission concept consists of the previously referenced five steps leading to the successful orbit and three-dimensional imaging of an RSO, using passive visual-only navigation and real-time near-optimal guidance. The mission design focuses on a 6U (34×22×10cm) CubeSat class nanosatellite of approximately 12kg mass operating in low earth orbit (LEO). ARAPAIMA’s payload consists of an MLR2K laser rangefinder from FLIR Systems Inc. (maximum range of approximately 2km) and a short wave GA640C IR camera from Goodrich Aerospace (640x512 pixel resolution and 15µ pixel depth) with a 40° full angle FoV. An S3R star tracker from the ADCS (discussed in Section 4) for visible spectrum imaging complements this setup. Due to the heavy computational loads associated with image processing, a dedicated ARM processor will form part of the payload unit.

To perform its maneuvers, ARAPAIMA is equipped with a warm gas propulsion system that is calculated to last for a mission length of approximately one year, including the amount required for a deorbit and re-entry maneuver. This propulsion system drives the main orbital maneuver thruster (OMT) as well as eight RCS thrusters installed such
as to produce control torques in both directions about each body axis (Figure 1). Yellow represents the geocentric inertial frame; blue represents the Earth-centered-Earth-fixed frame; the red represents the local-vertical local-horizontal frame; and the green is the body fixed frame.

ARAPAIMA will fly as secondary payload on a routine host mission that is yet to be determined. The RSO of choice is the spent upper stage of the host rocket; its last action will be to deploy ARAPAIMA from a canisterized satellite dispenser (CSD) specially designed to house 3U, 6U, 12U, and 27U CubeSats. After separation the satellite will enter a detumble mode, during which solar panels will be deployed and angular rates canceled. The satellite then assumes a nadir pointing position and begins a full systems, guidance, propulsion and proximity navigation checkout while in communication with the Mission Operations Center (MOC).

During this process, the attitude and position of the satellite is described using four different reference frames shown in Figure 2. The body fixed frame (BFF) describes the orientation of the RCS thrusters. The traditional geostationary frame (ECI) acts as the inertial frame of reference. The Earth-centered-Earth-fixed (ECEF) frame rotates with the Earth; the z-axis overlaps the Earth’s rotation axis and the x-axis crosses the 0° latitude and 0° longitude mark. The fourth and final reference frame is a local-vertical local-horizontal (LVLH) type frame in the north-east-down (NED) notation. This definition is dependent on its center’s location in the latitude-longitude grid, which changes over time as it tracks the body’s orbital path. By definition, the z-axis of the frame will always point radially down towards the Earth.

During the 3D imaging operations and maneuvers, the relative motion between the CubeSat and the RSO are modeled by choosing a series of waypoints around the RSO and letting the chaser fly freely between them, modeled by a combination of Clohessy-Wiltshire-Hill chasing maneuvers and relative orbit maintenance patterns. The modeling of disturbance torques and their effects on these relative trajectories are of extreme interest for proximity operations mission planning. The following section will describe the setup of environmental and internal disturbance torques, and the couples driving ef-
fects of reaction wheel torque. This will then be used to draw conclusions regarding the pointing accuracy and execution of ARAPAIMA mission objectives.

3. Disturbance Torques

The disturbance torques acting on the CubeSat have been assigned to two categories. The first is environmental disturbance torques; due to aerodynamic forces, gravity gradient, residual magnetic moment, and solar radiation pressure. The second category is that of internal disturbance torques; due to reaction wheel imbalance, propellant slosh, flexible modes of the deployable solar panels, and misalignment of the orbital maneuver thruster.

Methods employed for the evaluation of each disturbance torque are described in this section. All disturbance torques are applied along the axes of the body-fixed frames of the CubeSat.

3.1. Aerodynamic Torques

The aerodynamic torques are produced by the upper atmospheric particles colliding with the CubeSat. The worst case scenario of solar max activity is modeled for the calculation of the aerodynamic torques in this study. The Naval Research Laboratory Mass Spectrometer and Incoherent Scatter Radar (NRLMSISE-00) model of the atmosphere at the nominal altitude of 500km gives a composition of 94% Oxygen and 6% Nitrogen. The number and mass densities are $n=3.769 \times 10^{14} \text{m}^{-3}$ and $\rho=1.02 \times 10^{-11} \text{kg/m}^{-3}$ respectively. The mean free path is $\lambda=27.33 \text{km}$; when compared with a reference length $l_{\text{ref}}=0.1 \text{m}$, this results in a Knudsen number, $Kn=\frac{\lambda}{l_{\text{ref}}}=273,300$, indicating that the CubeSat flies in the free molecular flow regime.

Table 1: Component properties used in the DSMC calculations.

<table>
<thead>
<tr>
<th>Comp</th>
<th>Mass (kg)</th>
<th>Rot. DoF</th>
<th>Ref. dia. (m)</th>
<th>Ref. temp (K)</th>
<th>Visc. temp pwr.</th>
</tr>
</thead>
<tbody>
<tr>
<td>O</td>
<td>$2.656 \times 10^{-26}$</td>
<td>0</td>
<td>$3.000 \times 10^{-10}$</td>
<td>273.0</td>
<td>0.80</td>
</tr>
<tr>
<td>N</td>
<td>$2.326 \times 10^{-26}$</td>
<td>0</td>
<td>$4.200 \times 10^{-10}$</td>
<td>273.0</td>
<td>0.79</td>
</tr>
</tbody>
</table>

The aerodynamic torques on the CubeSat have been calculated with a freely available direct simulation Monte Carlo (DSMC) program called DS3V. The program employs algorithms based on the methods described by Bird [2]. A variable hard sphere model (VHS) has been assumed for the simulation of the collisions between atomic oxygen, atomic nitrogen, and the CubeSat. Parameters of the model are shown in Table 1. The free stream speed $V_\infty$ assumed was $7.612 \text{ km/s}$, the orbital speed in a 500km altitude circular orbit.

A total of 91 runs of the DS3V code were performed for 13 angles of attack and 7 sideslip angles. The angle of attack $\alpha$ and the sideslip angle $\beta$ are illustrated in Figure 3. $\alpha$ was varied between $-90^\circ$ and $90^\circ$ in steps of $15^\circ$ and $\beta$ between 0 and $180^\circ$ in steps of $30^\circ$.  

The maximum cross-sectional surface area of the CubeSat, with all solar panels deployed, is used as the reference area, \( S_{aero} = 0.2276 \text{m}^2 \). The smallest size of the CubeSat is used as the reference length, \( l_{aero} = l_{ref} = 0.1 \text{m} \). The aerodynamic coefficients for the torques about each body axis, calculated about the geometric center of the CubeSat bus, are obtained by calculating \( T_{aero} \):

\[
T_{aero} = \frac{1}{2} \rho V_\infty^2 C_m S_{aero} l_{aero}.
\]

The aerodynamic torque coefficients are loaded in two-dimensional look-up tables called by a Simulink model of the ARAPAIMA attitude dynamics. These are plotted in Figure 4, bottom row. The maximum value of the aerodynamic disturbance torque about any of the body axes is calculated to be \( 10^{-6} \text{Nm} \).

Figure 3: Sketch of the ARAPAIMA geometric fixed frame and the wind axes.

The validity of the DSMC method is confirmed by the results obtained for the aerodynamic force coefficients. As seen in Figure 4, the maximum aerodynamic force coefficient is \( C_{y,max} = 2.11 \), within 5% of the drag coefficient \( C_d = 2.2 \) widely used to calculate drag for satellites in LEO.

### 3.2. Gravity Gradient Torque

The \( 1/r^2 \) dependency of the gravitational force produces a torque that acts in such a way that, in the absence of any other torques, one of the principal axes of the CubeSat aligns itself with the local gravity field vector. The gravity gradient torque is calculated with:

\[
T_g = 3 \mu \hat{r} \times (I\hat{r}) / r^3
\]

where \( \mu = 398,600 \text{km}^3/\text{s}^2 \) is the gravitational parameter of the Earth, \( I \) is the moment of inertia tensor about the body axes, \( \hat{r} \) is the unit vector along the position vector of
the CubeSat, and \( r \) is the magnitude of the position vector. The mass of \textit{ARAPAIMA} with a full propellant tank is 7 kg. To calculate the moment of inertia, it is assumed that the principal axes coincide with the geometric fixed frame (GFF) and the body-fixed frame (BFF) axes. This is calculated to be \( I = \text{diag}(0.0541, 0.0914, 0.1054) \text{kgm}^2 \). The maximum value of the gravity gradient torque is calculated to be \( 10^{-7} \text{Nm} \).

### 3.3. Magnetic Torque

The magnetic disturbance torque has two major sources: 1) the force produced on a charge \( q \) by the magnetic component of the Lorentz force, \( qv \times \mathbf{b} \) and 2) the torque experienced by an aspheric paramagnetic body which, in the absence of other torques, aligns its long axis with the local magnetic field \( \mathbf{B} \). All the electrically conductive parts of the CubeSat contribute charges (electrons) to the magnetic Lorentz force and, as a consequence, the electric currents in all the energized CubeSat subsystems produce a Lorentz-type torque. In addition, Eddy (Foucault) currents induced by time-varying magnetic fields, both external and internal, produce disturbance torques. Direct evaluation of the magnetic disturbance torques on a low Earth orbit satellite is an extremely complex endeavor and researchers in the field often resort to empirical data to estimate them. For example, Inamori [5] discusses magnetic disturbance torques for small satellite systems and, based on empirical data, lumps the effects described above into a disturbance residual magnetic moment (RMM). Using Inamori’s data it has been determined that a RMM of \( 0.1 \text{Am}^2 \) is typical for the uncompensated 6U CubeSat analyzed here.

For implementation of in the Simulink model, the worst case is assumed, i.e., the RMM of \textit{ARAPAIMA} is parallel to the \( O_y \) body-fixed axis. The maximum magnetic disturbance torque can be determined assuming that the magnetic field and the RMM in \( T_{\text{rmn}} = m_{\text{rmn}} \times \mathbf{B} \) are perpendicular. With an average value of the magnetic field of \( 2.5 \times 10^{-5} \text{T} \) the expected magnetic disturbance torque is \( 2.5 \times 10^{-6} \text{Nm} \).

### 3.4. Solar Radiation Pressure Torque

Solar radiation pressure (SRP) is created by solar photons exchanging momentum with the CubeSat. If the point of application of the SRP force is shifted from the center of mass of the CubeSat, an SRP disturbance torque is created. The magnitudes of the SRP force and moment are complex functions of the shape of the CubeSat, the shading, and the optical properties of the materials exposed to the sunlight. The worst case solar radiation torque [12] can be calculated with \( T_{\text{srp}} = F_{\text{srp}} l_{\text{srp}} \), with \( F_{\text{srp}} = \Phi A_{\text{srp,max}}(1 + q_r) \cos(\phi)/c \), where \( \Phi = 1.366 \text{W/m}^2 \) is the solar constant, \( c = 3 \times 10^8 \text{m/s} \) is the speed of light in vacuum, \( A_{\text{srp,max}} = 0.2276 \text{m}^2 \) is the maximum exposed surface area, the moment arm is \( l_{\text{srp}} = 0.1 \text{m} \), and \( q_r \) is the reflectance factor. Since about 85% of the CubeSat surface exposed to the Sun is covered by solar cells, a reflectance factor of \( q = 0.6 \) typical of solar panels [12] is used. For worst-case modeling, the incidence angle of the Sun is assumed as \( \phi = 0^\circ \) which gives the maximum SRP disturbance torque. The maximum solar radiation torque is calculated to be \( 1.7 \times 10^{-7} \text{Nm} \).
For Simulink modeling, the SRP force is assumed to point in the direction of the Sun vector. An average surface area of $A_{\text{srp,avg}}=0.11\,\text{m}^2$ is used instead of the maximum value, while the SRP moment arm remains at 0.1m. The normalized (unit) Sun vector is calculated with Satellite Tool Kit (STK) in the Earth-centered Earth-fixed (ECF) coordinate system, loaded to Simulink and transformed to the BFF for calculation of the SRP disturbance torque.

Aerodynamic, gravity gradient, magnetic and Solar Radiation Pressure torques comprise the environmental disturbance torque model. The initial attitude of the CubeSat is chosen such that the body axes are parallel with the Earth-centered inertial (ECI) axes, making the initial body angular rates are null. Figure 5 shows the variation of the environmental torques with respect to time across one orbit.

![Graph showing environmental disturbance torques](image)

**Figure 5:** Evolution of the environmental disturbance torques, expressed in the BFF, during the first orbit of the CubeSat.

Next, we address internal torques, produced as a result of control inputs. A robust controller will be able to minimize the external torques on the system while coping with the internal torques discussed in the following subsections.

### 3.5. Reaction Wheel Imbalance Torque

Since reaction wheels do not have completely uniform mass distributions, mass lumps can be considered at the ends of the disk. These lumps cause the reaction wheels to present with two primary imbalances: Static and Dynamic. Static imbalance is also known as the “hopping” effect, observed when a mass lump exists on the edge of the
rotating disk, causing a radially outward force. The force produced is sinusoidal and is heavily related to the spin rate of the reaction wheel. Static imbalance is calculated using the following vector expression\[1\]

$$\vec{F}_{xyz} = \{0, U_S \omega^2 \sin(\omega t), U_S \omega^2 \cos(\omega t)\}$$

$U_S$ is the coefficient of static imbalance of the wheel, $\omega$ is the wheel’s angular velocity, and the wheel spins along the x-axis. This imbalance becomes a disturbance torque after converting $F_{xyz}$ into the BFF with $\vec{r}_{RW} \times \vec{vecF}_{xyz}$, where $\vec{r}_{RW}$ is the position of the reaction wheel from the center of mass with respect to the BFF.

Similarly, the dynamic imbalance results from mass lumps changing the principal moment of inertia axis from the spin axis. A torque is produced from the slight precession of the reaction wheel, which is calculated using:

$$\vec{\tau}_{xyz} = \{0, U_D \omega^2 \sin(\omega t), U_D \omega^2 \cos(\omega t)\}$$

$U_D$ is the coefficient of dynamic imbalance of the wheel, $\omega$ is the wheel’s angular velocity and the wheel spins along the x-axis. $\vec{\tau}_{xyz}$ is similarly transformed into into the BFF.

$U_S$ and $U_D$ are coefficients that are inherited from the manufacturer and cannot be controlled. Our analysis uses $U_S$ and $U_D$ values of $5 \times 10^{-10}$ Ns$^2$ and $10^{-10}$ Nm$s^2$ respectively.

### 3.6. Propellant Slosh Torques

While complex propellant slosh is an active research area, simple linear lateral sloshing models give good approximations to the slosh of propellant in a zero-g environment[8]. The model used consists of a solid mass which is “attached” to the tank. A second mass is used to represent the free surface motion. This mass is attached to the wall by a spring-damper system with the following equation\[10\]

$$m \ddot{x} + b \dot{x} + kx = F_{\text{Tank}},$$

where $m$ is the portion of propellant mass attached to the spring, $b$ is the damping coefficient of the system emulating the propellant’s viscous tendencies (or other damping elements like the tank’s baffle design), and $k$ is the spring coefficient representing the springiness of the dominating force (capillary forces dominate in the simulation’s regime).

The disturbance force is represented by the spring force. Tank shape, fill fraction, and the tank’s acceleration drive the parameters of the mass-spring-damper model. It is
assumed that a sloshing frequency of $\nu_{\text{slosh}}=0.03 \text{Hz}$ and a fill fraction of 0.032, and the following parameters are used: $m=0.9534 \text{kg}$, $b=0.3577 \text{Ns/m}$, and $k=0.0126 \text{N/m}$, with the $F_{\text{Tank}}$ forced by the attitude motion of the satellite.

### 3.7. Solar Panel Flexible Mode Torque

The joints of the solar panel are modeled after a torsional spring-damper system. The panel itself is considered to be a rigid body attached to the CubeSat by a flexible joint. The spring and damping coefficients are approximated from by a video model of the deployment of a 3U solar panel provided by Clyde-Space, similar to the hardware proposed for ARAPAIMA\[11\].

$$J\ddot{\theta} + b\dot{\theta} + k\theta = \tau_{\text{Panel}}$$

$J$ is the moment of inertia of the panel with respect to the joint axis, $b$ is the damping coefficient of the joint, $k$ is the torsional spring coefficient of the joint, and $\tau_{\text{Panel}}$ is the torque acting on the hinge, modeled as being in the opposite direction of the reaction wheel torque response. $J$ is calculated using measurements made available by Clyde-Space. The top and bottom (3U Panel) are calculated to have a moment of inertia 0.0350kgm$^2$ and the side panels (6U Panel) have a moment of inertia of 0.0193kgm$^2$. The spring and damper coefficients are approximated to be 0.45Nm/rad and 0.09Nms/rad respectively. The initial deployment response is shown in Figure 7.

### 3.8. Disturbance Torque Results

Simulink results of the of each internal disturbance torques are shown. Because the reaction wheel torque drives some of the disturbances, the reaction wheel torque response is also shown. The total internal disturbance plot is shown in Figure 8 together with individual plots for each internal disturbance.
Figure 8: Sum all internal imbalances throughout one orbit with no control torques applied (top right) and individual contribution of body torques due to solar panel flexing, propellant slosh, and reaction wheels respectively.

Figure 9: Body torques due to the static imbalance of the reaction wheels
4. Attitude Determination and Control System

During normal operations (scientific mode) the CubeSat uses an STC-CLC202A monochrome camera from SenTech Inc. and a S3S star tracker (STR) from Sinclair Interplanetary for attitude determination. [11] The S3S star tracker has a rated accuracy of 7as about axes normal to its optical axis and 70as about the optical axis. The monochrome camera has a resolution of 1628x1236 with an element size of 4.4/μm. Three RW-0.03-4 reaction wheels are used to reject internal and external perturbations. The RW’s are mounted so that their axes are closely aligned with those of the satellite; each can produce a nominal torque of 2mNm and store an angular momentum of 30mNms at 5600RPM. They are also used to perform minor changes in the satellite’s attitude such as slewing on different axes. The RCS thrusters are used to periodically off-load the angular momentum accumulated in the RW’s.

During detumbling operations, information gathered form the STR and the monochrome camera is unusable due to the large angular rates of the satellite. In this situation a nanoIMU from MEMSense [11] is used to measure these rates which are then counteracted by the RCS thrusters and RW’s for detumble. These two sensors, together with the Novatel OEMV-1 GPS receiver and patch antenna, complete the guidance navigation and control suite.

During proximity operations, the main OMT is used to get within range of the RSO and supply the small ΔV needed to transfer to and maintain the relative orbit. RCS thrusters are used to spin stabilize the satellite during these maneuvers. During RSO
rendezvous, the thrusters are also used to set the satellite in its relative circular orbit.

While the ADCS of the ARAPAIMA mission is very well defined in terms of requirements for successful execution, the state of the system is limited by unknown budgets and constraints arising from other aspects of the mission still in development. The design team has not advanced to hardware-in-the-loop testing; at present time, the only hardware tested with the ADCS is the laser rangefinder, during testing of selected 3D imaging algorithms. [4] However, significant progress has been made in the software design of a control system. A plant faithful to the encountered disturbances discussed previously has been designed from scratch and implemented with the attitude dynamics of a satellite in orbit by applying the torques directly to the body axis. The celestial mechanics portion of the model has been incorporated by means of an oblate Earth model in which the orbital altitude fluctuates with time. It also has a great impact in determining the attitude and position of the NED frame; this ensures that the disturbances modeled in this paper have a solid foundation on position with respect to the Earth’s surface.

The current model utilizes RW’s alone as actuators and neglects inherent sensor inaccuracies. These inaccuracies are of future interest, particularly those of the STR and monochrome camera. Due to visual-based navigation and image processing, these could result in significant orbital path error if not handled appropriately.

The model of the system takes advantage of the small time step nature of numerical integrators such as Simulink to perform single-input single-output (SISO) control on the system. Three Proportional-Derivative (PD) controllers control the angular position of one body axis relative to the NED frame. The angular positions being monitored are the x, y, and z components of the rotation vector formed by the product of the unit Euler axis and the Euler angle. The PD controllers send a control signal consisting of a torque command into the RW actuator model which is then further filtered by a second layer of Proportional-Integral (PI) controllers that manage the RW DC motors using voltage signals.

5. Simulation and Results

As mentioned previously, the simulation runs on MATLAB/SIMULINK. The solver used is ODE14x, an extrapolation method that is very accurate when using fixed step simulation. The fixed step size is 0.01 s. The simulation is run for one orbit period (5676s for a 500km circular orbit). Figure 11 show the output of ECEF position and velocity as the orbit time lapses.

With only RWs as actuators, keeping good pointing accuracy in the presence of all modeled disturbances requires a well-tuned controller. The initial direction was to use Ziegler-Nichols tuning methods, but after in-depth simulation it was determined that the system was too sensitive for integral action. This, along with no classical method for PD control left manual tuning. The controller is structured as a three-axis controller with a zero radian reference signal. The controller response is interpreted as a torque
command and sent directly to the plant (disturbances) and into the reaction wheel block input. The RW torque command and the 6DOF outputs are fed into all the other internal and external disturbance blocks. Resulting disturbances are fed into the 6DOF block to produce a quaternion from the BFF to the NED frame.

With good tuning, the perfect quaternion response produced by the model of the attitude dynamics should result in a constant [1 0 0 0] quaternion as time lapses.

Without control, due to all the disturbance torques present, the satellite cannot maintain alignment between the BFF and NED frames. Figure 12 shows the quaternion and the 3-2-1 Euler angles from NED to BFF without any control. Clearly, the quaternion is far from identity, i.e., the two frames are not aligned. The angles between BFF and NED also shown in Figure 12 further supports this observation.

Next, the PD controller is implemented. Figure 13 shows results of the satellite’s Euler rotation vector[3] components from the BFF to NED frame after manual tuning. Controller gains pertaining to the classic PID error equation used to produce the results
are: Proportional \( K_p = 0.001 \) and derivative \( K_d = 0.022 \). As it is evident in Figure 13 the angles are in the range of 0.002 radians or approximately 7 arcmin.

![Figure 13: Euler angles from BFF to NED (target frame) with active PD control.](image)

This falls short of our ideal 1 arcmin mark. However, as a preliminary analysis benchmark, the results are promising. As the manual tuning process accuracy improves more accurate results are expected. Of note in Figure 13 is a pseudo-steady state error. Properly introduced integral action should reduce this error significantly. Further analysis is ongoing to find a proper integral coefficient that does not excite the system dynamics excessively.

6. Conclusions

The main purpose of the current work was to develop and simulate the pointing accuracy of a nadir-pointing CubeSat in the presence of external and internal disturbances. Aerodynamic drag, magnetic dipole moment, gravity gradient, solar radiation pressure, propellant slosh, reaction wheel imbalance, and solar panel array flex were studied. Although a preliminary analysis, results were found to be very encouraging. A very high accuracy goal has been set for an ideal control scenario; positive results at a pre-PDR design point indicate the likelihood of a favorable final response, even when sensor inaccuracies are implemented. Further, the high frequency oscillatory motion in Figure 15 is expected to decrease significantly in magnitude with the introduction of the RCS thrusters into the simulation. Finally, modeling will be streamlined by the acquisition of a D-Space satellite test bed which will allow the ARAPAIMA team to test flight hardware alongside software simulations. Future implementations planned include complex celestial mechanics analysis involving orbital maneuvers towards and relative to the RSO.
References


