On-Orbit Center Of Mass Relocation System For A 3U Cubesat

Humberto Hernández-Arias, Jorge Prado-Molina

Abstract: This paper describes the design and simulation tests of a system that uses a set of three internal shifting masses (ISMs) that can be displaced along three orthogonal axes, with the purpose to re-adjust the center of mass (CoM) in a nanosat. This procedure will reduce the magnitude of the on-orbit external disturbing torques, re-locating the CoM as close as possible to the geometrical center of the spacecraft. This implementation decreases the power needed to keep the attitude stable along time. Angular rates and angular accelerations, provided by an IMU, are employed by the onboard computer that executes the algorithm for the estimation of the CoM and the magnitude of the displacements needed on every sliding element. The disturbance torques considered were: solar pressure, aerodynamic drag and gravitational gradient. In every control loop CoM deviation is estimated using a recursive least square (RLS) method, the magnitude of the displacements of each ISM is calculated and moved accordingly. In the next step on-board computer identifies the new position of the CoM and the magnitude of the displacements, until the CoM position is within the threshold range, from the geometric center of the spacecraft, established in ± 5 mm. Feasibility of the CoM estimation, and relocation system was performed through numerical simulations using Matlab Simulink considering a 3U Cubesat nanosatellite dynamic model.

Index Terms: Cubesat, center of mass estimation, moving masses system, center of mass relocation, attitude disturbances reduction, Recursive Least Square

1 Introduction

ATITUDE stability on a spacecraft is constantly affected by external disturbance torques, for instance, aerodynamic drag; resulting from the interaction of the satellite with the remnant particles of the atmosphere that exist in low Earth-orbit (i.e. 300 to 1200 km), where there is not atmosphere as such, however, there are gaseous particles like mono-atomic oxygen, this interaction is known as atmospheric drag [1]0. There are also other forces that interact with the satellite changing its attitude, as the pressure produced by the solar wind 0, and the gravity gradient, 0. For many applications and experiments in space missions, it is essential that payload continuously exhibits a pre-established attitude, accurate and stable, in order to properly carried-out the programmed mission. This task is performed by the attitude determination and control subsystem (ADCS), essentially composed by an on-board computer, attitude sensors, actuators, and algorithms. The ADCS in a Cubesat has severe restrictions imposed by the standard, in terms of power available, total mass, and dimensions (3U Cubesat = $3\ dm^3$, 4.2 Kg, and 3 W) 0. So, an ADCS of high performance in a nano satellite is a challenge. The best knowledge of the mass properties of the satellite is essential when applying the attitude determination and control algorithm, which uses these parameters when solving the equations of motion of the spacecraft.

- *Humberto Hernandez-Arias is currently pursuing doctor of engineering degree in electrical engineering at the National Autonomous University of Mexico, Mexico, PH- (52) 55 56224358 ext. 45479. E-mail: xmbeto@hotmail.com*
- *Jorge Prado-Molina is the Head of the Geospatial Analysis Laboratory, at the Geography Institute of the National Autonomous University of Mexico, mexico, (52) 55 56224358 ext. 45475. E-mail: jprado@igg.unam.mx*

The CoM of a spacecraft can change once it is positioned in orbit, due to some operations of reconfiguration 0, for example, the deployment of some instruments or the payload itself, i.e. solar panels, gravity booms, magnetometers 0, antennas, payload camera 0, or solar sails 0. Other causes may be the expulsion of fuel mass 0, not very common in Cubesats, or by the deformation of the spacecraft caused by excessive forces during launch. Even if a very careful CoM positioning is performed on the Laboratory, there will be an error of a few millimeters; this small amount of shifting will cause a deviation of the expected pointing direction once the nanosat being in orbit, due to the above mentioned disturbing torques. This system was projected to compensate this shifting, re-adjusting the center of mass, locating it as close as possible to the geometric center of the nanosatellite. This will reduce on-orbit energy consumption while maintaining pointing accuracy. The proposed system requires accurately estimating the position of the center of mass of the satellite; also, attitude determination algorithm requires establishing this parameter as well as the inertia matrix, to improve its performance. On the laboratory, the mass identification properties of the Cubesat (mass ID), can be done with several methods and tools, for example through 3D drawings using software applications like SolidWorks®, which computes a value of the center of mass 0, although this analytical value can be considered as true, it is difficult to consider some elements as wires, harnesses, connectors, or elements with irregular geometries. In this work we employed an approach to estimate the center of mass, based on the Recursive Least Square Method (RLS). The mass ID has been done in some works reported in the literature [12]. The RLS algorithm approach is one of the most employed, for example: Wilson *et. al*., used it, to identify the center of mass and the inverse inertia matrix of three different thruster-controlled spacecraft's 0. Kim and Agrawal 0, proposed a batch LS method for the mass properties estimation of a ground-based spacecraft simulator. Another approach for estimating mass properties is through the usage of Kalman Filtering, as in the work reported by Zhao *et. al.* 0. Kim *et. al.* proposed a mass properties estimation method that combines a LS and a Kalman filter 0. In 0, Silva exposes an applied mass properties ID method for the CalPoly air-bearing satellite simulator, based on experimental data and the LS algorithm. On the other hand, the use of moving mass elements inside a spacecraft has been studied before with different purposes. Chesi *et. al*. in, 0, proposed a set of moving masses taking advantage of the aerodynamic force in orbit, to create an external torque with the purpose of carrying-out attitude control. In 0, Kumar presents a scheme of attitude control for a small satellite with a linear moving mass to generate on-orbit control torques. Edwards, proposed an approach to detumble a large manned spacecraft in case of failure 0, leading the uncontrolled three-axial angular movement of the spacecraft into a simple spinning rotation by the action of two movable masses. A moving mass system to generate and modify the angle of attack of aerospace vehicles is proposed by Robinett *et al.* 0. A similar analysis is carried out by Guo and Zhao, in 0 they proposed an attitude control system scheme where changing the positions of two masses generating torques to control the attitude. Scholz et al. 0 studied an attitude control scheme with moving masses for a solar sail spacecraft, to compensate the offset between the center of mass and the center of solar radiation pressure. In contrast with the above mentioned previous works, here, we propose the re-location of the center of mass to reduce the lever arm generated by the disturbance torques, this will maintain pointing stability along time, decreasing the duty cycle of the attitude control system, resulting in a reduction of the power employed to stabilize the satellite in the long term.

TABLE 1
UNITS FOR MAGENTIC PROPERTIES

$\mathbf{I}_{sat}$	Cubesat inertia matrix
$\boldsymbol{\omega}_{sat}$	Angular velocity of the satellite body-fixed coordinate system with respect to the inertial coordinate frame
$\dot{\boldsymbol{\omega}}_{sat}$	Angular acceleration of the satellite body-fixed coordinate system with respect to the inertial coordinate frame
CoM	Center of Mass of the satellite structure
C_{geom}	Geometric center of the satellite structure
CPs	Center of solar pressure
CPa	Center of aerodynamic pressure
$\boldsymbol{\tau}_{total}$	Total torque around the CoM of the satellite
τ_k	Sum of disturbance torques $=\boldsymbol{\tau}_{w_i}+\boldsymbol{\tau}_{dis}$
$\boldsymbol{\tau}_{w_i}$	Torque produced by the reaction wheels
$\boldsymbol{\tau}_{dis}$	Disturbance torques
$\mathbf{I}_w$	Inertia of the reaction wheel
$\boldsymbol{\omega}_{rw_i}$	Angular rate of the reaction wheel
$\dot{\boldsymbol{\omega}}_{rw_i}$	Angular acceleration of the reaction wheel
τ_g	Gravity gradient torque
ϕ	Yaw attitude angle
θ	Theta attitude angle
ϕ	Pitch attitude angle
R	Distance of the spacecraft from Earth's center
$\mathbf{r}_{Cps}$	Vector from the CoM to the CPs
$\mathbf{r}_{CoM}$	Vector from the Geometric center of the spacecraft to its Center of mass
$\mathbf{r}_{solar}$	Lever arm for the solar torque compute, equals to: $\mathbf{r}_{Cps}-\mathbf{r}_{CoM}$
$\boldsymbol{\tau}_{solar}$	Solar radiation pressure torque
$\mathbf{F}_s$	Solar radiation force
ρ_s	Solar constant
$\mathbf{c}$	Speed of light
A_s	Area of the satellite surface exposed to the solar radiation
k	Reflectance coefficient of satellite's surface
$\mathbf{r}_{aero}$	Lever arm for the aerodynamic torque, equals to: $\mathbf{r}_{Cpa}-\mathbf{r}_{CoM}$
$\boldsymbol{\tau}_{aero}$	Aerodynamic drag torque
$\mathbf{F}_a$	Aerodynamic force
ρ_a	Atmospheric density
V	Satellite velocity
S	Area of the face exposed to the aerodynamic drag
C_D	Drag coefficient
$\mathbf{F}_{aero}^{\times}$	Skew symmetric representation of the aerodynamic force
$\mathbf{F}_{solar}^{\times}$	Skew symmetric representation of the solar radiation force
$\mathbf{CM}_{off}$	Center of mass deviated
$\mathbf{CM}_{obj}$	Center of mass as correction objective
$\mathbf{r}_{off}$	Vector from C_{geom} to $\mathbf{CM}_{off}$
$\mathbf{r}_{obj}$	Vector from C_{geom} to $\mathbf{CM}_{obj}$
$\mathbf{r}_{off}$	Vector error from the $\mathbf{CM}_{obj}$ to the $\mathbf{CM}_{off}$
$\boldsymbol{P}_i$	Position vector of each moving mass
$\boldsymbol{l}_i$	Initial position vector of each moving mass
$\boldsymbol{v}_i$	Unit vector in the direction of displacement of each mass
d_i	Displacements of every adjustment mass referred to their initial positions
m	Total mass of the satellite
m_B	Sum of the mass of the moving mass elements m_1, m_2 and m_3
$\mathbf{P_0}$	Position of the center of mass when the moving mass elements are not present
$\mathbf{V}$	Velocity vector of the satellite, tangent to the orbit path
μ	Earth's gravitational constant equals to: $0.3986\times 10^{15}\mathrm{m}^3/s^2$
EoM	Equations of Motion
RW	Reaction Wheels

2 SATELLITE MATHEMATICAL MODEL

The 3U Cubesat satellite was modeled as a rigid body, and the equations of motion (EoM) were derived from the Newton-Euler formulation, where angular momentum changes directly with the applied torque 000, as showed in (1):

$$\mathbf{I}_{sat}\dot{\boldsymbol{\omega}}_{sat}+\boldsymbol{\omega}_{sat}\times(\mathbf{I}_{sat}\boldsymbol{\omega}_{sat})=\boldsymbol{\tau}_{total} = \sum_k \tau_k \quad (1)$$

$\boldsymbol{\tau}_{total}$, is composed by the sum of all the torques actuating over the spacecraft, *i.e.* those generated by the actuators, and the external disturbance torques. In this case a set of three reaction wheels (RW) was selected as actuators, giving a torque of $\boldsymbol{\tau}_{w_i}$, meanwhile the disturbance torques considered in the space environment were $\boldsymbol{\tau}_{dis}$, those are due to solar pressure, aerodynamic drag, and gravitational gradient. Total torque around the center of mass of the satellite is described by equation 2:

$$\boldsymbol{\tau}_{total}=\sum_k \boldsymbol{\tau}_k=\boldsymbol{\tau}_{w_i}+\boldsymbol{\tau}_{dis} \quad (2)$$

The RW torque can be expressed as 0:

$$\boldsymbol{\tau}_{w_i}=\mathbf{I}_w\dot{\boldsymbol{\omega}}_{rw_i}-\boldsymbol{\omega}_{sat}\times\mathbf{I}_w\boldsymbol{\omega}_{rw_i} \quad (3)$$

In this paper aerodynamic drag, solar pressure, and gravity gradient are considered as the main disturbance torques actuating into the nanosat.

2.1 Disturbance torque models

The interaction of the satellite with the space environment generates forces which are applied in a point called center of pressure, measured from the CoM, which produces disturbance torques that affect its attitude, causing deviations of the established reference pointing values. As the disturbance forces has components that affect all the three-axis of the spacecraft, it is desirable that the CoM being located close to the geometric center, reducing the disturbance force lever arm and thus the produced torque, (see figure 1). For drag torque calculation, the center of aerodynamic pressure (CPa) is considered located at the geometric center of the surface of the satellite exposed to the atmospheric drag. In the case of the solar pressure, the center of pressure (CPs) is also considered located at the geometrical center of the side of the satellite that faces the Sun.

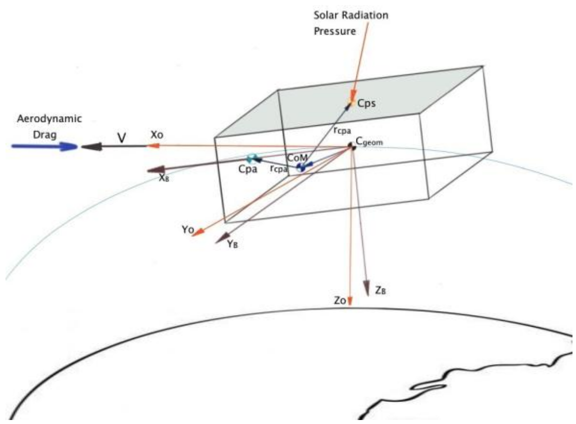

Fig. 1. Coordinate systems for the CoM estimation: Orbital Frame $X_O Y_O Z_O$, with its origin in the CoM of the satellite, X_O is tangent to velocity vector V, Z_O points towards Earth's center, and Y_O completed the orthogonal system. The Body frame $X_B Y_B Z_B$ is a frame aligned with the principal axis of the satellite and moves with it, and it is expected that these two frames be aligned when the desired attitude is reached. The Cpa is the point where it is considered the aerodynamic force is applied. The Cps corresponds to the point where the solar radiation force is concentrated.

Gravity gradient: In orbit, gravitational field is not uniform, spacecraft experiences these variations materialized as an external torque, because Earth doesn't have a perfect distribution of its mass. Equation 4 can be used to compute the net gravity gradient torque 0:

$$\begin{aligned}\tau_{g_x} &= \left(\frac{3\mu}{2\mathrm{R}^3}\right)\left(I_{ZZ}-I_{yy}\right)\sin(2\phi)\cos^2(\theta)\\ \tau_{g_y} &= \left(\frac{3\mu}{2\mathrm{R}^3}\right)\left(I_{ZZ}-I_{xx}\right)\sin(2\theta)\cos(\phi)\\ \tau_{g_z} &= \left(\frac{3\mu}{2\mathrm{R}^3}\right)\left(I_{xx}-I_{yy}\right)\sin(2\theta)\sin(\phi)\end{aligned} \quad (4)$$

Where ϕ and θ, refer to the roll and pitch attitude angles.

Solar radiation pressure: Solar radiation produces a force, which depends on the exposed area and the material of the surface of the satellite. Solar radiation pressure means a force per unit area that induces a torque, which can be calculated with equation 5, 0:

$$\begin{aligned}\boldsymbol{\tau}_{solar} &= \mathbf{r}_{solar}\times\mathbf{F}_s\\ &= \left(\mathbf{r}_{Cps}-\mathbf{r}_{CoM}\right)\times\frac{\rho_s}{\mathbf{c}}A_s(1+\mathrm{k})\end{aligned} \quad (5)$$

Aerodynamic drag: In Low Earth-Orbit (LEO), the remnants of the atmosphere cause an aerodynamic drag force that induces a torque onto the satellite this can be described by eq. (6), 0:

$$\boldsymbol{\tau}_{aero} = \mathbf{r}_{aero}\times\mathbf{F}_a \quad (6)$$

Where $\mathbf{r}_{aero} = \left(\mathbf{r}_{Cpa}-\mathbf{r}_{CoM}\right)$, is the vector from the CoM to the CPa of the aerodynamic drag, considering that the atmospheric flow is tangential to the orbit. The aerodynamic force can be expressed with equation 7:

$$\mathbf{F}_a = \frac{1}{2}\rho_a V^2 S C_D \quad (7)$$

3 Center of Mass Estimation

To apply the RLS method with the purpose of estimating satellite's center of mass, an equation of the system in the form of: $Ax = b$, is required, where $\boldsymbol{x}$ contains the parameters to be estimated. In the method proposed here, above mentioned on-orbit external disturbance torques, as well as the torques from the actuators, *i.e.* reaction wheels, were employed. From equation (1), and considering the atmospheric, solar, and gravity torques we have:

$$\begin{aligned}\mathbf{I}_{sat}\dot{\boldsymbol{\omega}}_{sat}+\boldsymbol{\omega}_{sat}\times(\mathbf{I}_{sat}\boldsymbol{\omega}_{sat}) &= \mathbf{r}_{Cpa}\times\mathbf{F}_{aero}+\mathbf{r}_{Cps}\times\mathbf{F}_{solar}+\boldsymbol{\tau}_{ctrl}\\ &+\boldsymbol{\tau}_g\end{aligned} \quad (8)$$

In order to obtain the standard form $\boldsymbol{Ax}\approx\boldsymbol{b}$ required to use the least square method, we rewrite eq. (8). Considering that: $\mathbf{r}_{solar}=\mathbf{r}_{Cps}-\mathbf{r}_{CoM}$, and $\mathbf{r}_{aero}=\mathbf{r}_{Cpa}-\mathbf{r}_{CoM}$, this allow us to combine and rearrange equation (8) as:

$$\begin{aligned}(\mathbf{F}_{aero}^{\times}+\mathbf{F}_{solar}^{\times})\mathbf{r}_{CoM} &= \mathbf{I}_{sat}\dot{\boldsymbol{\omega}}_{sat}+\omega_{sat}\times(\mathbf{I}_{sat}\boldsymbol{\omega}_{sat})\\ &-\boldsymbol{\tau}_{ctrl}-\boldsymbol{\tau}_g+\mathbf{F}_{aero}^{\times}\mathbf{r}_{Cpa}+\mathbf{F}_{solar}^{\times}\mathbf{r}_{Cps}\end{aligned} \quad (9)$$

Then the elements required in the RLS algorithm for the CoM estimation can be defined as:

$$\begin{aligned}&\mathrm{A}\mathbf{x}=\mathrm{b}\\ &\mathrm{A}=(\mathbf{F}_{aero}^{\times}+\mathbf{F}_{solar}^{\times}),\qquad \mathbf{x}=\mathbf{r}_{CoM},\\ &\mathrm{b}=\mathbf{I}_{sat}\dot{\boldsymbol{\omega}}_{sat}+\omega_{sat}\times(\mathbf{I}_{sat}\boldsymbol{\omega}_{sat})-\boldsymbol{\tau}_{ctrl}-\boldsymbol{\tau}_g+\mathbf{F}_{aero}^{\times}\mathbf{r}_{Cpa}\\ &\quad+\mathbf{F}_{solar}^{\times}\mathbf{r}_{Cps}\end{aligned} \quad (10)$$

3.1 Numeric Simulations for the RLS Mass ID

Matlab–Simulink simulations were carried-out to evaluate the performance of this estimation method in the identification process of the center of mass. Numeric simulations included solar radiation pressure, aerodynamic drag, and gravity gradient as disturbances torques, also with white noise added. The model included the EoM of the dynamics and kinematics of the satellite considering a 3U Cubesat with the characteristics mentioned on table 2. The tests consist in evaluating the attitude dynamics with previously defined parameters and with a known CoM, the values of the angular rate and the disturbance forces obtained at each time step are used in the RLS formulation (see eq. 10) to obtain an estimated CoM value. In table 2, the most important parameters involved in the simulation are presented.

TABLE 2

Parameters used onto simulink numerical simulations

arameter	Value
Satellite total mass	4.2 Kg
Moments of inertia of each reaction wheel	$I_{w_i} = 3.092 \times 10^{-5}\ [Kg\, m^2]$
Reaction wheel angular rate	360 rpm
Satellite's moments of inertia matrix	$I_{sat} = \begin{bmatrix} 0.005 & 0 & 0 \\ 0 & 0.025 & 0 \\ 0 & 0 & 0.025 \end{bmatrix} [Kg\, m^2]$
CoM position vector, $\mathbf{r}_{\text{CoM}}$	$[0.02 \quad -0.03 \quad -0.04]\ m$
Cpa	$[0.15 \quad 0 \quad 0]\ m$
Cps	$[0 \quad -0.05 \quad 0]\ m$
Orbit	Polar LEO, Altitude: 370 Km, inclination 92°

In figure 2, the graphic depicts estimated values of the CoM for a 180 s simulation. The values of the center of mass used in the EoM were: $\text{CoM} = [0.02 \quad -0.03 \quad 0.04]\ \text{m}$, considering these distances as the true position of the CoM. These values were approximately obtained in simulation tests, as can be seen in the next graphs.

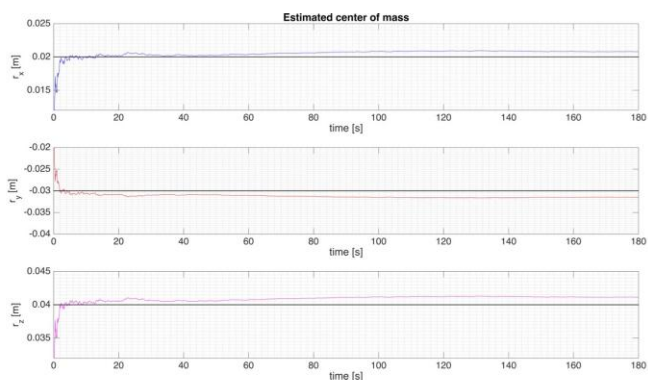

Fig. 2. *Estimated values of the Center of mass: the simulation shows that the solution converges in approximately 120 s, when these estimated values are stabilized.*

Figure 3 shows a graphic of the relative error obtained with the simulation of the RLS identification method.

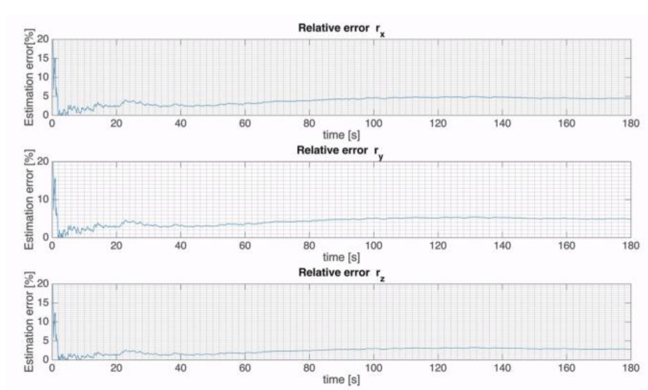

Fig. 3. *Relative error of the CoM estimation. These graphics shows a relative error with a maximum of 5%, these values can be translated in the range of millimeters.*

4 INTERNAL SHIFTING MASSES SYSTEM

The proposed system consists of a set of three masses positioned on three orthogonal axes aligned with the satellite body coordinate system. Each mass can be displaced along the unit vectors denominated as: v_1, v_2 and v_3; meanwhile l_1, l_2 and l_3 represent initial positions, see figures 4-5.

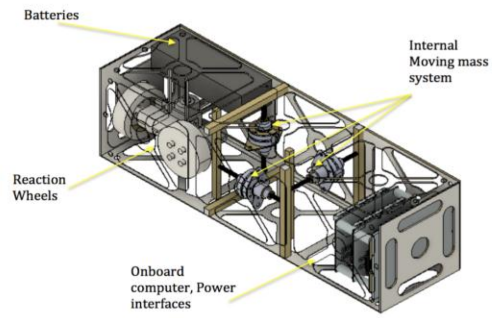

Fig. 4. *Proposed Internal moving mass system inside a 3U Cubesat structure, showing the three linear motors, the reaction wheels, the onboard computer, and the batteries.*

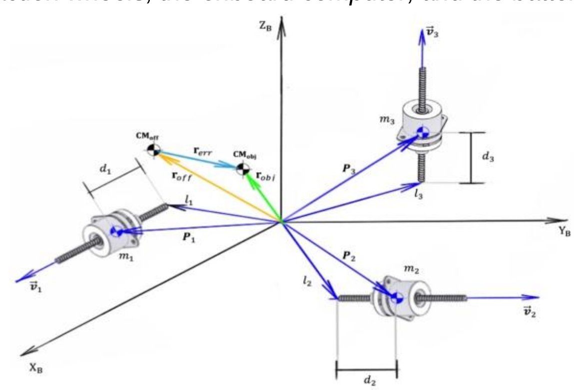

Fig. 5. *Internal shifting masses system scheme: the displacements of every mass can be denoted by* d_1, d_2 *and* d_3*, respectively, referred to their initial positions.*

The position vector, $\boldsymbol{P}_i$, of each moving mass can be described by equation 11:

$$\boldsymbol{P}_i = \boldsymbol{l}_i + d_i \boldsymbol{v}_i \quad for\ i = 1,2,3 \tag{11}$$

In figure 4, $\mathbf{CM}_{off}$ is the position of the center of mass previously estimated, $\mathbf{CM}_{obj}$ corresponds to the position of the center of mass that is expected to reach once the moving mass adjustment is applied, and that is getting closer to the center of geometry of the Cubesat structure. The position vectors of $\mathbf{CM}_{off}$ and $\mathbf{CM}_{obj}$ are denoted by $\mathbf{r}_{\text{off}}$ and $\mathbf{r}_{\text{obj}}$, respectively. The vector $\mathbf{r}_{\text{err}}$ is the difference between $\mathbf{r}_{\text{off}}$ and $\mathbf{r}_{\text{obj}}$, it can be seen on equation (12):

$$\mathbf{r}_{\text{off}} = \mathbf{r}_{\text{obj}} + \mathbf{r}_{\text{err}} \tag{12}$$

Thus, the value of $\mathbf{r}_{\text{err}}$ represents the offset that must be corrected with the mass system. The position vector of $\mathbf{CM}_{off}$ can be computed with eq. 13:

$$\mathbf{r}_{\text{off}} = \frac{1}{m}\left[(m - m_B)\mathbf{P_0} + \sum_{i=1}^{3} m_i(\boldsymbol{l}_i + d_i \boldsymbol{v}_i)\right] \tag{13}$$

Where m is the total mass of the cubesat including the moving masses, with $m_B = m_1 + m_2 + m_3$, as the sum of the moving masses elements. $\mathbf{P_0}$, is the position vector of the center of mass when these moving masses elements are not present in the Cubesat structure. This vector must be obtained on the laboratory, prior to launch. As established by the Cubesat standard the CoM shall be located within a 2 cm from its geometric center for the X and Y-axes, and 7 cm for Z -axis 0. The position vector $\mathbf{r}_{\text{obj}}$ can be computed with the initial distances of the moving masses equaling zero, i.e.:

$$\mathbf{r}_{\text{obj}} = \frac{1}{m}\left[(m - m_B)\mathbf{P_0} + \sum_{i=1}^{3} m_i(\boldsymbol{l}_i)\right] \qquad (14)$$

As an inherent constraint, $\mathbf{r}_{\text{off}}$ will be defined within a threshold, which will bound the range of action of the moving masses system. With the above, distance d_i, where each mass will be positioned, is defined by the error vector: $\mathbf{r}_{\text{err}}$, which can be calculated by equation 15:

$$\mathbf{r}_{\text{err}} = \mathbf{r}_{\text{off}} - \mathbf{r}_{\text{obj}} = \frac{1}{m}\sum_{i=1}^{3} m_i d_i \boldsymbol{v}_i \qquad (15)$$

4.1 Center Of Mass Adjustment

By means of eq. 13, a numeric calculation of the range of action of the ISMs can be performed, where the CoM of the satellite is computed for various positions of the moving masses within the range of ± 5 [cm] corresponding to the displacements d_i, for the moving mass elements in X, Y and Z axis. Table 3 shows the values used in these tests.

TABLE 3

INITIAL VALUES FOR NUMERIC SIMULATION OF THE INTERNAL MOVING MASSES SYSTEM

Parameter	Value
Total mass of the satellite including the adjustment masses	$m = 4.2$ [Kg]
CoM without masses	$\mathbf{P_0} = [\,2.17 \quad 2.17 \quad 2.17]$ [cm]
CoM objective to reach	$\mathbf{CM_{obj}} = [\,2.0 \quad 2.0 \quad 2.0]$ [cm]
Mass of each actuator	$m_p = 0.170$ [Kg]
Unitary vectors of the displacement axis of each mass	$U_1 = [1 \quad 0 \quad 0]$ $U_2 = [0 \quad 1 \quad 0]$ $U_3 = [0 \quad 0 \quad 1]$
Initial position of the masses	$l_1 = [0.01 \quad 0.01 \quad 0.01]$[m] $l_2 = [0.01 \quad 0.01 \quad 0.01]$ [m] $l_3 = [0.01 \quad 0.01 \quad 0.01]$ [m]

The CoM compensation depends on the mass of the moving element and the distance at which each element can be moved. From eq. 13, it can be observed that the moving masses system is able to modify the center of mass of the satellite within its entire range of adjustment of ± 5 [cm] in the X, Y and Z-axis, with a mass of $m_p = 170$ [g]. According to the results of the simulations, the maximum compensation of the CoM is 5 [mm] in each axis, for the characteristics and conditions of the proposed system.

4.2 Center Of Mass And Power Consumption On The Attitude Actuators

A series of simulations were conducted to observe the dynamic behavior of the satellite with its CoM located at distinct positions, and its relationship with the power consumption of the attitude actuators. As have been mentioned above, the interaction of the satellite with the space environment produces forces that are applied in a point called the center of pressure. In the case of the aerodynamic and solar disturbances the location of this point is considered at the geometric center of the surface of the spacecraft exposed to the atmospheric flux, and solar radiation, respectively. The distance between the center of pressure and the center of mass of the spacecraft is a lever arm that is involved in the generation of the disturbance torques that affects the attitude, producing a deviation of the desired pointing, then requiring applying the control torques by the actuators. With numerical simulations of the satellite's dynamics, a graphic representation of its attitude behavior with respect to external torques and the position of the center of mass were obtained. In the graphs of figure 6, the changes in satellite's attitude from an initial value, are introduced. With initial angular velocity equal to zero, in a period of one orbit, approximately 5500 seconds, and with a $\text{CoM} = [3.5 \quad 3.5 \quad -3.5]$ centimeters, actuators are activated once the attitude angles reach the threshold of ±0.9°, and deactivated when attitude angles are below 0.01°. In figure 7 disturbance torques are showed for this case, and, figure 8 shows the duty cycles of the actuators if these had to actuate the entire orbit period.

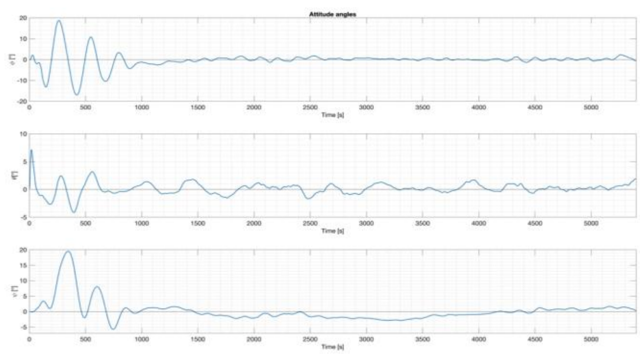

Fig. 6. *Satellite attitude with $CoM = [3.5 \quad 3.5 \quad -3.5]$ cm. The simulation shows large oscillations in the attitude angles due to the disturbance torques, requiring that the attitude subsystem apply stabilization torques.*

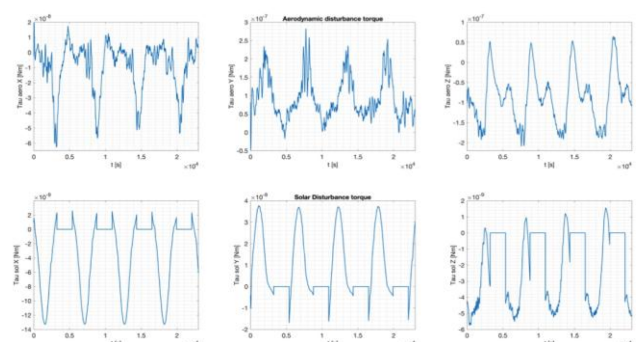

Fig. 7. *Aerodynamic and Solar disturbance torques actuating on the satellite for four orbits, when the CoM is located on: $CoM = [3.5 \quad 3.5 \quad 3.5]$ cm.*

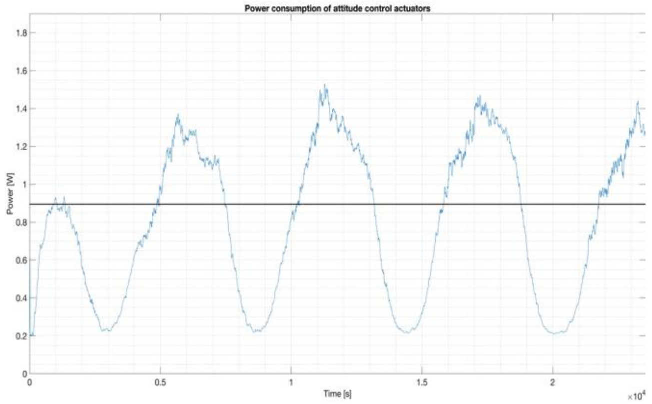

Fig. 8. *Power consumption of the actuators when the CoM is located in this position: $CoM = [3.5 \quad 3.5 \quad 3.5]$ cm. ACS turns-on the actuators to produce a corrective torque, with an average of 0.89 W, during the 4 orbit simulation.*

On the other hand, if the center of mass is adjusted to: $\mathrm{CoM} = [0.03 \quad 0.03 \quad 0.03]$, we obtain the graphs of figures 9 to 11.

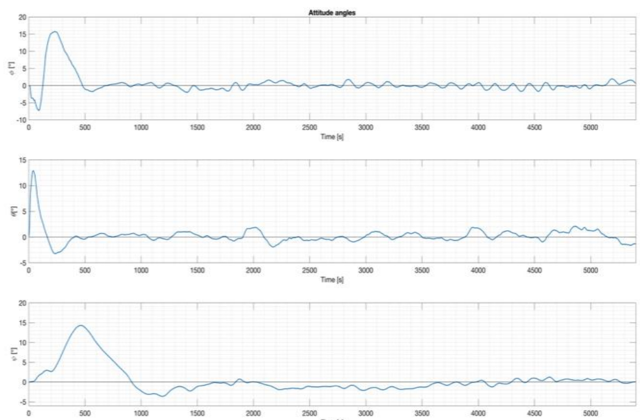

Fig. 9. *Satellite attitude, $CoM = [3.0 \quad 3.0 \quad 3.0]$ cm. In this case the attitude is more stable, as the disturbance torques has smaller magnitude.*

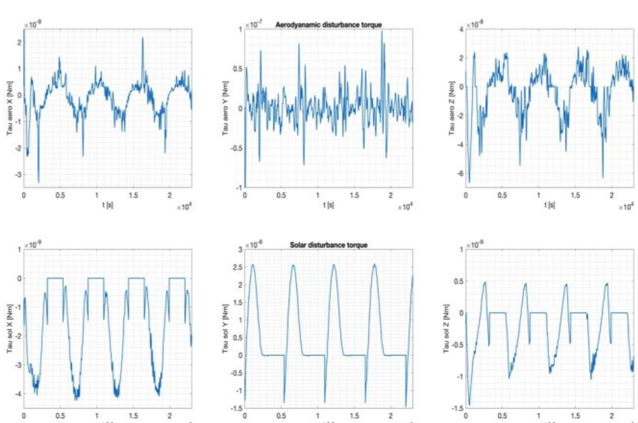

Fig. 10. *Aerodynamic and Solar disturbance torques on the satellite, $CoM = [3.0 \quad 3.0 \quad 3.0]$ cm. In this case, disturbance torques has approximately an order of magnitude less than the previous case.*

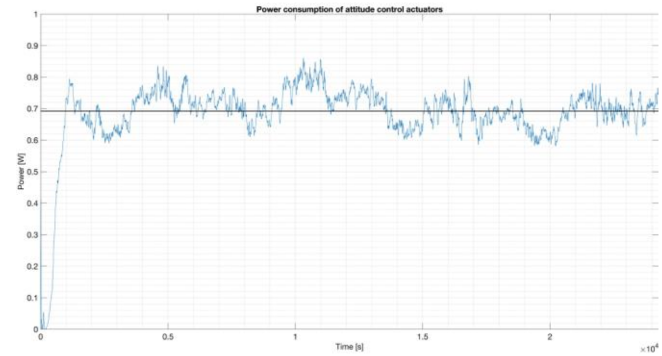

Fig. 11. *Power consumption of the actuators, $CoM = [3.0 \quad 3.0 \quad 3.0]$ cm. In this graph is shown that attitude actuators require less power to maintain the attitude, an average of 0.68 [W], and with less oscillations.*

5 Discussion

The moving mass system proposed here, is intended to perform a fine adjustment of the center of mass of the satellite, reducing the lever arm involved in the disturbance torque generation due to the space environment. In contrast with previous works that involves internal moving masses, the ISM proposed here is not intended to generate any control torques but to reduce the environmental torques, by reallocating the CoM closest to the geometrical center of the satellite's structure. This will be helpful when a reconfiguration in orbit occurs, or in case of the failure of the deployment of a component like antennas or solar panels. The system described in this paper represents a novel approach for external torques reduction suitable for nano satellites [29]. Results from the numeric simulations indicate that the implementation of this system will reduce the load on the attitude control subsystem, which traduces into a reduction of the power consumption on the attitude actuators, being this, the principal advantage. In the numeric simulations presented here, changing the CoM from $\mathbf{CoM} = [\mathbf{3.5} \quad \mathbf{3.5} \quad \mathbf{3.5}]$cm to $[\mathbf{3.0} \quad \mathbf{3.0} \quad \mathbf{3.0}]$ cm, represents a reduction of approximately 20% of the power required to maintains attitude stability.

6 Conclusions

A set of three sliding masses, each one can be moved along an orthogonal-axis, is proposed as a system to reallocate the center of mass of a satellite once it was deployed in orbit. The first step in this method is to estimate the current satellite's CoM. For this purpose, a center of mass estimation method based in the recursive least square algorithm has been developed. This ID method uses the equations of motion of the satellite, including the torque contribution of three reaction wheels, and the external disturbances from the gravity gradient, aerodynamic drag, and solar wind. The estimation method was simulated in a Matlab-Simulink dynamic model for a 3U Cubesat. The result from these simulations shows a convergence of an acceptable value of the position of the CoM of the satellite in approximately 120 s, while the relative error is less than 5%. Then, the required displacement of each mass is computed, and after the shifting elements are moved, the system will have a new CoM that needs to be estimated. This loop repeats until the estimated CoM is positioned in a pre-established threshold of 5 mm. The numeric tests, showed that the proposed system could adjust the position of the CoM inside the boundary of 5 mm in each axis, for a 3U Cubesat. From the analysis of the CoM position and the effect of the disturbance torques on the attitude, we observe that, when the CoM is near to the geometric center of the satellite structure, this will be traduced in the decrement of the frequency of activation of the actuators. This represents a reduction of 20% of power consumption in the ACS in its long-term operation while maintaining the nanosatellite stabilized in three-axis.

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