Politecnico di Milano

Dipartimento di Ingegneria Aerospaziale

Spacecraft Attitude Dynamics Project

Project n°149

Team n°81

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1.Introduction

1.1) Mission

Our satellite is designed to …

1.2) Orbit Specifications

The Orbit on which our Satellite is, is the following :

- Altitude : 1200 km
- Inclination : 100.38 °
- Période orbitale : 1h49min

1.3) Spacecraft description

Our satellite is a 6U cubesat.
Our satellite is composed of :

- Mandatory :
  - 8 Constant Thrust Jets
  - 1 Sun sensor
- Non Mandatory
  - 1 Magnetometer
  - 4 Reaction Wheels

2.Spacecraft Geometry

3.Dynamics

we have used Euler

To determine the attitude dynamics of the CubeSat we use the Euler's equations. The Euler equations are the analytic form adopted to describe the attitude motion of a rigid body.

The fundamental equation in body axes is : $Id\omega = I\omega \wedge \omega + M_c + M_{ext}$

With M_c the moment of control, M_{ext} the moment of the external disturbances applied on the spacecraft and ω the angular position of the spacecraft in the inertial frame.

$$I_x \omega_x = (I_y - I_z)\omega_y \omega_z + M_{c,x} + M_{ext,x}$$

$$I_y \omega_y = (I_z - I_x)\omega_x \omega_z + M_{c,y} + M_{ext,y}$$

$$I_z \omega_z = (I_x - I_y)\omega_x \omega_y + M_{c,z} + M_{ext,z}$$

4.Kinematics

with/over time

Kinematics describe how the attitude parameters change in time, according to angular velocities. Our model has been developed using direction cosines matrix (DCM) $A_{B/N}$:

$$\frac{dA_{B/N}}{dt} = -[\omega \wedge]A_{B/N}$$

Due to computational errors, at each time-step $A_{b/n}$ must be orthonormalized, according to a single-step iteration:

$$A_{B/N}(t) = \frac{3}{2}A_{B/N} - \frac{1}{2}A_{B/N}(A_{B/N}{}^{t})A_{B/N}$$

5.Environmental Perturbation

Even if the satellite is orbiting around the Earth, over the atmosphere boundaries, it is still subjected to some disturbance torques due to more perturbations: some of them can be neglected but others must be considered. We have chosen to include the Solar radiation pressure, the gravity gradient and the Earth's magnetic field. The Solar radiation pressure is the disturbance that has the greatest effect on our spacecraft, but the gravity gradient and the Earth's magnetic field have an impact on the spacecraft that we can neglect so we take them into account.

The sat is orbiting to high to be subjected to a relevant atmospheric drag

5.1) Magnetic Disturbance

Earth's magnetic field interacts with the spacecraft, causing a disturbance torque equal to:

$$M = m \wedge B$$

where B is Earth's magnetic field and m is the magnetic dipole of the Earth.

The magnetic field can be modeled in a direct way, as seen during lectures:

$$B_N(r) = \frac{R_E^3 H_0}{|r|^3}\left[3(\widehat{m} \cdot \widehat{r})\,\widehat{r} - \widehat{m}\right]$$

5.2) Gravity Gradient Disturbance

The Earth's gravity field causes a disturbance torque on the satellite: in fact, since the field is not uniform, there will be a different attraction force acting in different points of the spacecraft. The effect is more relevant for satellites of large dimension, but even if the resulting torque is small the effect can be considerable due to the long time of action.

The total torque can be computed as the sum the gravity effects acting on the elementary mass dm:

$$M = \int dM = -\int r \wedge \frac{Gm_t}{|R+r|^3}(R + r)dm$$

5.3) Solar Radiation Pressure Disturbance

over a surface

Solar radiation illuminating a surface of the spacecraft generates a pressure and therefore a torque on the satellite itself. Radiation on spacecraft comes from 3 different sources: directly from the Sun, radiation reflected by the Earth and Earth radiation.

6.Sensors

Add gyoscopes Safran Stim277H random walk 0.15-0.30
Add CubeSpace CubeSense N

the sensors allow the spacecraft to compute its position...

Sensors are what allows the spacecraft to understand its position, orientation, and speed in space and therefore they are very important to compute the attitude dynamics. In fact, there cannot be a reasonable control and a correct pointing of the spacecraft during its orbit without knowing exactly its orientation in space with respect to Earth, Sun, or the stars.

Our assignment requires us to use a Sun sensor. However we will also implement a Magnetometer.

The model of each sensor is not made using their physical behavior but is made in term or errors on the measurement they provide. To compute each sensor, we compute an error on the real position of the spacecraft in its body frame.

6.1) The Magnetic Field Sensor or Magnetometer

The model of the magnetometer is quite simple and consists in adding a random noise on top of the exact value of the magnetic field modeled in the environment. This random added noise should represent the noise caused by the limited accuracy of the device.

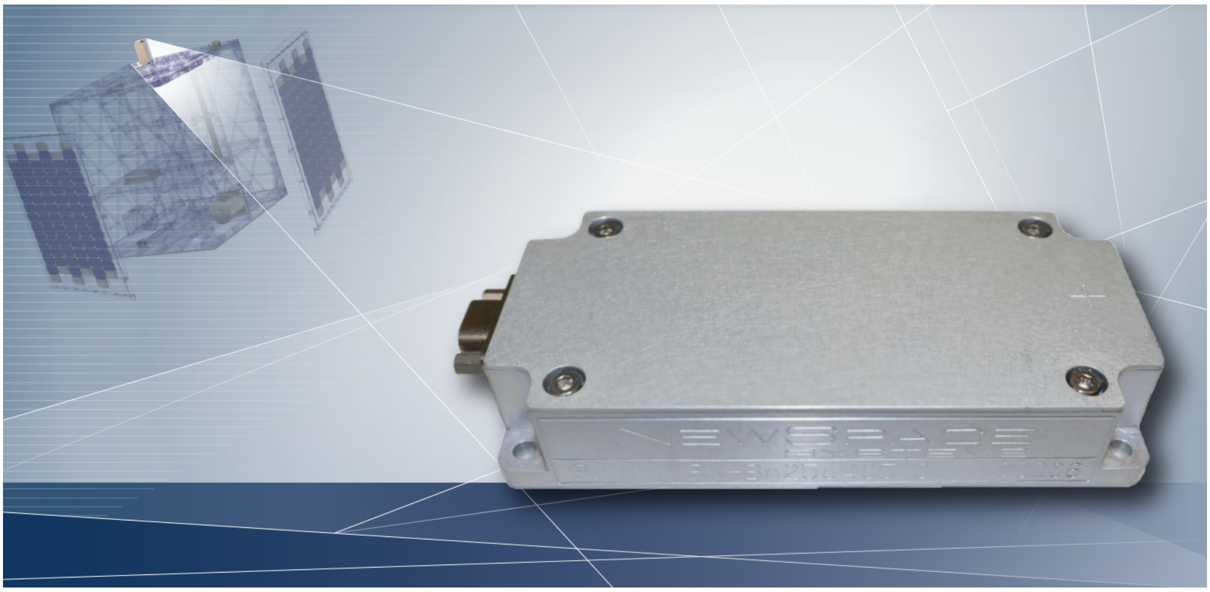

Figure 1 : New Space Systems NMRM-Bn25o485 Magnetometer

Accuracy	Update Rate [Hz]	Mass [kg]
	< 18 Hz	85 g

+ - 1°

6.2) Sun sensor

As said in class, Sun sensors are normally based on materials that produce an electric signal when illuminated by Sun radiation.
In our case, we followed the method to simulate one explained during the lecture that consists in multiplying the direction of the sun in the body frame by a matrix error.

Figure 2 : New Space Systems NFSS-411 Sun sensor

Field of view [deg]	Accuracy [deg]	Update Rate [Hz]	Mass [kg]
114 °	< 0.01 °	5 Hz	< 35 g

+- 0.1

7.Attitude Determination

In order to get the attitude matrix $A_{B/N}$ of the spacecraft, we need at least two measurements coming from our two sensors. We decided to implement **an algebraic method attitude algorithm** (see annexe), using measurements of:

- the Sun direction S_B coming from the Sun sensor
- the magnetic field vector B_B coming from the Magnetometer

Given these two measurements, the attitude determination algorithm compares them with their exact values in inertial frame and gets the rotation matrix $A_{B/N}$ by constructing two orthogonal frames.

In order to increase the precision of this method, it is necessary to have accurate sensors and the two vectors given as inputs must be as orthogonal as possible.

8.Missions

In this project we study three phases of attitude control. Each phase has its own control law as they all have a different objective. The aim of a control law is to determine the control torque to apply to the spacecraft to control its attitude.

8.1) Detumbling

Reducing the angular velocities after launcher release is the first operation that a spacecraft must do for keeping a stable attitude and continuing its mission. There are different ways to proceed, but we developed …

8.2) Slew Maneuver

The Slew maneuver phase consists in achieving a desired attitude through the rotation of the spacecraft at a certain angle while the satellite is not spinning rapidly.

it's defined as a rest to rest motion between two attitudes

8.3) Pointing

9.Actuators

The moment of control which is given by the control law is not directly applied to the dynamics of the spacecraft. The actuators transform this desired control M_c into a real control torque M_{real}, which is applied to the spacecraft by the actuators.

9.1) Constant Thrust Jets

The assigned actuators for our spacecraft were thrusters with constant thrust. We selected XXXXXXX. The configuration chosen for the actuators leads to 8 actuators, that are sufficient to guarantee a correct control torque.

The 8 thrusters are placed as in the image :

This design leads us to a configuration matrix computed as:

$$A = \begin{bmatrix} 0 & 1 & -1 \\ 0 & -1 & 1 \\ 0 & -1 & -1 \\ 0 & 1 & 1 \\ -1 & 0 & 1 \\ -1 & 0 & -1 \\ 1 & 0 & 1 \\ 1 & 0 & -1 \end{bmatrix}$$

9.2) Reaction Wheels

As mentioned, to better fulfill our mission requirements we decided to limit the usage of thrusters only in initial detumbling maneuvers and selected a new kind of actuators for the nominal phase: reaction wheels.

The reasons are the following:

- limit jets fuel consumption in the first part of the mission
- reaction wheels continuous torque guarantees more precision than impulsive behavior of constant jets, and can be a fail-over option in case of jets failure

The reaction wheel selected is XXXXX and reaches a maximum torque of XXXXX. That unit, developed into a pyramidal configuration, avoids singularities in each maneuver and perfectly manages the disturbances for the whole mission without overcoming maximum torques or reaching saturation for the rotational speed.

This leads to the following configuration matrix:

$$R = \frac{1}{\sqrt{3}} \begin{bmatrix} -1 & 1 & 1 & -1 \\ -1 & -1 & 1 & 1 \\ 1 & 1 & 1 & 1 \end{bmatrix}$$

10.Annexe

10.1) Algebraic Method Attitude Algorithm

The method used is the Algebraic method, with this method we can determine the attitude using only 2 vectors measured by the sensors :

$$\left[s_1\, s_2\, s_3\right] = \left[v_1\, v_2\, v_3\right]$$

We call s_i the unit vectors measured by the sensors, in body-fixed frame, v_i the unit vectors of the same celestial bodies but referred to the inertial frame.

In our case we only have 2 measurements from the sensors, we call them p and q. We call a and b their corresponding directions in inertial space.

We define p^* and q^* :

$$p^* = \frac{p+q}{2}, \; q^* = \frac{p-q}{2}$$

Use p^* and q^* instead of p and q in the algorithm permits to minimize errors. We define the s_i and v_i vectors :

$$s_1 = p^*, \; s_2 = \frac{p^* \wedge q^*}{\left|p^* \wedge q^*\right|} \; \& \; s_3 = p^* \wedge s_2$$

$$v_1 = a, \; v_2 = \frac{a \wedge b}{\left|a \wedge b\right|} \; \& \; v_3 = a \wedge v_2$$

$$A_{B/N} = SV^T$$