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► **To cite this version:**

Sihem Tebbani, Maria Makarov, Cristina Stoica Maniu, Israel Hinostroza. Nanosatellite attitude control using electromagnetic actuation. 4th SONDRRA Workshop, May 2016, Lacanau, France. hal-01383419

HAL Id: hal-01383419

<https://hal.science/hal-01383419v1>

Submitted on 16 Mar 2020

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Nanosatellite Attitude Control using Electromagnetic Actuation

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Abstract—The paper deals with the attitude active control of a Low Earth Orbit nanosatellite, actuated with magnetorquers. The satellite is on a polar orbit and is subject to gradient gravity. Magnetic actuation is preferred for nanosatellite attitude control because of its relatively low cost, lightweight and low power consumption. The basic idea is to use the interaction between the magnetic field generated by the coils and the Earth's magnetic field. The developed control strategy is a Linear Quadratic controller for the three-axis stabilization of the satellite. This optimal controller is derived from a linear model of the system which is nonlinear and time-varying. Since magnetic actuation can lead to poor performances because only two axes can be controlled at each time instant, an hybrid actuation system with magnetorquers and reaction wheels is also investigated. Simulation results are provided in the case of a 2U CubeSat attitude stabilization to illustrate and compare the control strategy efficiency in three cases: (i) with magnetic torquers, (ii) with gyroscopic torquers and (iii) with the hybrid actuation system.

I. INTRODUCTION

In recent years, several nanosatellite missions were developed for astronomy studies, remote sensing, technology demonstration or student projects [1]. They present the advantage of a small size and mass (1-10 kg), low cost and a short development cycle. The miniaturization of the satellite helps also to reduce the deployment costs. The most recent nanosatellite projects use the CubeSat standard, a unit with the standard size 10x10x11.35 cm and a weight of 1.33 kg. The CubeSat uses commercial off-the-shelf (COTS) components for the structure and the electronics, which helps in reducing costs. Nanosatellites are deployed in Low Earth Orbits (LEO), i.e. with an altitude lower than 2000 km. For these altitudes, several disturbances act on the the satellite such as drag forces, gravity irregularities and gradient, internal magnetic residual torque, etc. Depending on the payload and/or satellite mission, the specification on the position and pointing accuracy can be very high. Indeed, the accuracy of the orbit and attitude controls of a nanosatellite directly affect the performance of the payload and conditions the mission success. The Attitude Control System (ACS) must then be developed to ensure high performances. The satellite control is challenging because of the use of low cost hardware actuators and sensors to ensure the nanosatellite economical viability. Several control strategies were proposed for the attitude control of a nanosatel-

lite [2]. The typically used actuators are the reaction wheels (for gyroscopic control), gas thrusters and magnetorquers (for magnetic control). The principle of the latter actuators is to create a magnetic field by three mutually perpendicular magnetic coils. A mechanical torque is produced from the interaction of coils magnetic field with the geomagnetic one, which corrects the satellite attitude.

The challenge when using magnetorquers lies in the development of reliable efficient control strategies. The linearized model of the system is time-varying, leading to complex control strategies that induce a high computation burden. In addition, at a fixed time, this actuation can lead to the uncontrollability of the system (only two directions can be controlled). Finally, when magnetometers are used for the attitude determination, using magnetorquers can distort the measure.

In this paper, a linear quadratic controller is developed to regulate the attitude. Since the model is time-varying, the control law is derived from the solution of a differential Riccati equation. In order to reduce the computation on board, two strategies are investigated. The first one consists in calculating the control law off-line based on an approached model. In the second approach, the solution calculated off-line is partially updated on board from the measurement of the Earth magnetic field. In order to increase further the control strategy performance, reaction wheels are added to the actuation system. Reaction wheels are more efficient than magnetorquers for satellite control attitude. However, they are expensive and have high power consumption. In addition, they suffer from common hardware failures and saturation. The combination of reaction wheels with magnetorquers is thus investigated in order to take advantage of the two kinds of actuation systems. The performance of control strategies when using only magnetorquers and when using only reaction wheels, and when using both of the two kinds of actuation systems are compared. The chosen configuration that uses three magnetorquers and one reaction wheel on the roll axis is shown to present the best accuracy achievement.

The paper is organised as follows. First, the system modelling is described in Section II. The controller design is then presented in Section III. In Section IV, some simulation results are provided to illustrate the control strategy efficiency. Finally,

conclusions and some perspectives end the paper.

II. SYSTEM MODELLING

A. Nonlinear model

The system is modelled via kinematics and dynamic equations [3], [4]. The satellite is regarded as a rigid body. The kinematics equation is described in terms of the quaternion vector:

$$\dot{\mathbf{q}} = \frac{1}{2} \mathbf{Q}(\boldsymbol{\Omega}_o) \mathbf{q} \quad (1)$$

where $\boldsymbol{\Omega}_o$ is the angular velocity in the orbit frame, \mathbf{q} is a four order quaternion vector and $\mathbf{Q}(\boldsymbol{\Omega}_o)$ a skew-symmetric matrix. The attitude dynamics of the satellite is described by:

$$\dot{\boldsymbol{\omega}} = \mathbf{J}^{-1} (\boldsymbol{\tau}_c + \boldsymbol{\tau}_{gg} + \boldsymbol{\tau}_d - \boldsymbol{\omega} \times (\mathbf{J}\boldsymbol{\omega})) \quad (2)$$

where $\boldsymbol{\omega}$ is the angular velocity in the inertial frame, \mathbf{J} is the inertia tensor, $\boldsymbol{\tau}_c$ the control torque, $\boldsymbol{\tau}_{gg}$ the gravity gradient torque and $\boldsymbol{\tau}_d$ the disturbance torque (e.g. aerodynamic drag, solar radiation, residual magnetic torques, etc). The cross product is denoted by \times . The angular velocities are related by:

$$\boldsymbol{\omega} = \boldsymbol{\Omega}_o + \mathbf{R}_o^B(\mathbf{q}) \begin{bmatrix} 0 \\ -\omega_0 \\ 0 \end{bmatrix} \quad (3)$$

where ω_0 is the orbital rate. $\mathbf{R}_o^B(\mathbf{q})$ is the rotation matrix between the orbit and body frames and it depends on the quaternion vector \mathbf{q} .

The gravity gradient torque can be approached by:

$$\boldsymbol{\tau}_{gg} = 3\omega_0^2 \mathbf{k} \times (\mathbf{J}\mathbf{k}) \quad (4)$$

where \mathbf{k} is the projection of the z-axis in the body frame.

The control torque delivered by magnetorquers is given by:

$$\boldsymbol{\tau}_m = \mathbf{m} \times \mathbf{B}_e(t) \quad (5)$$

where \mathbf{m} is the magnetic dipole moment generated in the coils (the control input of the system), and $\mathbf{B}_e(t)$ is the Earth's magnetic field, with $\mathbf{B}_e(t) = [B_x, B_y, B_z]^T$. The latter is function of the altitude, longitude and latitude of the satellite. In this study, the IGRF model is used [5].

The control torque delivered by reaction wheels is given by:

$$\boldsymbol{\tau}_r = \dot{\mathbf{h}} + \boldsymbol{\omega} \times \mathbf{h} \quad (6)$$

where \mathbf{h} is the angular momentum of the wheels.

B. Linear model

The model described previously is linearized around the equilibrium:

$$[\mathbf{q}_e^T, \boldsymbol{\omega}_e^T] = [0 \ 0 \ 0 \ 1 \ 0 \ 0 \ 0] \quad (7)$$

This equilibrium represents in this case the nadir pointing. Since the fourth quaternion is redundant, it will be removed from the state vector in the linear model. The inertia tensor \mathbf{J} is considered diagonal. Inertia momentum along each axis are

denoted I_x, I_y and I_z . The system is then represented with a linear state-space equation:

$$\dot{\mathbf{x}} = \mathbf{A}\mathbf{x} + \mathbf{B}_j(t)\mathbf{u}_j, j = 1..3 \quad (8)$$

with $\mathbf{x} = [\delta q_1, \delta \dot{q}_1, \delta q_2, \delta \dot{q}_2, \delta q_3, \delta \dot{q}_3]^T$. The control input is either $\mathbf{u}_1 = \delta \mathbf{m}$ (when only magnetorquers are used), $\mathbf{u}_2 = \delta \dot{\mathbf{h}}$ (when only reaction wheels are used) or $\mathbf{u}_3^T = [\delta \mathbf{m}^T \ \delta \dot{\mathbf{h}}^T]$ (the hybrid configuration when the both kinds of actuators are used). In this section, three reaction wheels and/or three magnetorquers are considered (the model can be easily reduced when using less actuators as presented in Section IV). The evolution matrix is given by:

$$\mathbf{A} = \begin{bmatrix} 0 & 1 & 0 & 0 & 0 & 0 \\ a_1 & 0 & 0 & 0 & 0 & a_2 \\ 0 & 0 & 0 & 1 & 0 & 0 \\ 0 & 0 & a_3 & 0 & 0 & 0 \\ 0 & 0 & 0 & 0 & 0 & 1 \\ 0 & a_4 & 0 & 0 & a_5 & 0 \end{bmatrix}, \quad (9)$$

with $a_1 = -4\omega_0^2(I_y - I_z)/I_x$, $a_2 = \omega_0(I_x - I_y + I_z)/I_x$, $a_3 = -3\omega_0^2(I_x - I_z)/I_y$, $a_4 = -\omega_0(I_z + I_x - I_y)/I_z$ and $a_5 = -\omega_0^2(I_y - I_x)/I_z$.

The control matrices are given by:

$$\mathbf{B}_1(t) = \frac{1}{2} \mathbf{J}^{-1} \begin{bmatrix} 0 & 0 & 0 \\ 0 & B_z(t) & -B_y(t) \\ 0 & 0 & 0 \\ -B_z(t) & 0 & B_x(t) \\ 0 & 0 & 0 \\ B_y(t) & -B_x(t) & 0 \end{bmatrix}, \quad (10)$$

$$\mathbf{B}_2 = \frac{1}{2} \mathbf{J}^{-1} \begin{bmatrix} 0 & 0 & 0 \\ 1 & 0 & 0 \\ 0 & 0 & 0 \\ 0 & 1 & 0 \\ 0 & 0 & 0 \\ 0 & 0 & 1 \end{bmatrix}, \quad (11)$$

$$\text{and } \mathbf{B}_3(t) = [\mathbf{B}_1(t) \ \mathbf{B}_2] \quad (12)$$

The matrix $\mathbf{B}_1(t)$ is time-varying since the Earth's magnetic field depends on the satellite position.

The system is thus modeled by a first-order differential matrix equation (8) with the dynamics (9), and a control matrix given by (10), or (11), or (12) depending on the considered actuation system. It is assumed that all the state variables are available, either measured or estimated by the Attitude Determination System (ADS).

III. CONTROLLER DESIGN

The controller is designed by considering the linear model described in Section II-B. Consequently, the proposed controller aims at stabilizing the satellite for slight variations around the equilibrium. The nonlinear model could be considered for the detumbling control of the satellite which is out the scope of the present paper.

A. Linear Quadratic controller

The multi input multi output linear system is controlled using a Linear Quadratic controller (LQR). The aim is to maintain the attitude at a given reference value. In this paper, the regulation of the state vector at a null value is considered, corresponding to aligned orbital and body frames. This could be of interest in case of pointing the nadir for Earth observation missions for example. The control input is derived by the minimization of a quadratic performance index:

$$\min J = \frac{1}{2} \int_0^{t_f} (\mathbf{x}^\top \mathbf{W}_x \mathbf{x} + \mathbf{u}^\top \mathbf{W}_u \mathbf{u}) dt \quad (13)$$

where $\mathbf{W}_x \succeq 0$ and $\mathbf{W}_u \succ 0$ are constant weighing matrices and are the tuning parameters of the controller. t_f is the final time. The optimal control law is then given as a linear state feedback:

$$\mathbf{u}(t) = -\mathbf{W}_u^{-1} \mathbf{B}^\top(t) \mathbf{P}(t) \mathbf{x}(t) \quad (14)$$

where $\mathbf{P}(t)$ satisfies the Riccati differential equation:

$$-\dot{\mathbf{P}}(t) = \mathbf{P} \mathbf{A} + \mathbf{A}^\top \mathbf{P} - \mathbf{P} \mathbf{B}(t) \mathbf{W}_u^{-1} \mathbf{B}(t)^\top \mathbf{P} + \mathbf{W}_x \quad (15)$$

with the final condition $\mathbf{P}(t_f) = \mathbf{0}$. The controller consists in solving online the differential equation (15) and deducing the control input from (14).

B. Magnetic control

In the case of magnetic control, the control matrix \mathbf{B}_1 (see (10)) is time-varying because of the time-evolution of the Earth magnetic field. Solving online (15) is memory and power demanding. In order to reduce further the computation burden, two approaches are proposed.

1) *Fixed controller gain*: In the first approach, the Earth magnetic field model is used to solve off-line the differential equation (15). The derived gain $\mathbf{L}_f(t) = \mathbf{W}_u^{-1} \mathbf{B}_1(t)^\top \mathbf{P}(t)$ is then stored on board and the law (14) is applied online without solving the Riccati equation, leading to:

$$\mathbf{u}_1(t) = -\mathbf{L}_f(t) \mathbf{x}(t) \quad (16)$$

2) *Updated controller gain*: In the second approach, the LQ gain \mathbf{L}_f is updated online from the measurement of the Earth magnetic field:

$$\mathbf{u}_1 = -\mathbf{W}_u^{-1} \hat{\mathbf{B}}_1(t)^\top \mathbf{P}(t) \mathbf{x}(t) \quad (17)$$

where $\mathbf{P}(t)$ is computed off-line from the model considered for the Earth magnetic field (as in the fixed gain approach). It should be mentioned that the geomagnetic field is usually measured online with magnetometers since it is used for the attitude determination system [6].

C. Gyroscopic control

In the case of reaction wheels actuation system, the model is time-invariant. In this case, when $t_f \rightarrow \infty$, the matrix \mathbf{P} tends to a stationary matrix, solution of the Riccati algebraic equation:

$$\mathbf{P} \mathbf{A} + \mathbf{A}^\top \mathbf{P} - \mathbf{P} \mathbf{B}_2 \mathbf{W}_u^{-1} \mathbf{B}_2^\top \mathbf{P} + \mathbf{W}_x = \mathbf{0} \quad (18)$$

The controller gain $\mathbf{W}_u^{-1} \mathbf{B}_2^\top \mathbf{P}$ is thus constant and can be calculated off-line and stored on-board.

It should be mentioned that the approach used for the gyroscopic actuation could be used for the magnetic one, by considering an average model for the system [7]. The idea is to replace the Earth magnetic field by its average value over a given time duration, exploiting periodicity assumptions on the Earth magnetic field. This approach was tested in this work but failed to stabilize the system. Indeed, the controller robustness and performance highly depend on the choice of the averaging window length. Further study should be carried out to choose an adequate value depending on the considered mission characteristics.

D. Hybrid control

The hybrid configuration leads to a time-varying model and consequently the control approach is similar to the one developed in Section III-B.

IV. SIMULATION RESULTS

In this section, the LQ controller is applied in the case of attitude control of a 2U CubeSat (two units of cubeSat) nanosatellite. It has a polar circular orbit with an altitude of 300km. Control inputs must satisfy upper and lower bounds that depend on the used magnetic coils and reaction wheels ($m_{max} = 0.3 \text{ Am}^2$, $\dot{h}_{max} = 0.01 \text{ Nm}$). Euler angles at initial time are all chosen equal to 15 deg. Initial angular velocities are null. The orbital satellite initial position is the ascending node. The matrices \mathbf{W}_x and \mathbf{W}_u are diagonal and are chosen empirically. They are the same for all tests: $\mathbf{W}_x = \text{diag}([1, 0, 1, 0, 1, 0])$, $\mathbf{W}_{u1} = \text{diag}([1, 1, 1])$, $\mathbf{W}_{u2} = 10\mathbf{W}_{u1}$. A disturbance torque is applied: $\boldsymbol{\tau}_d = 10^{-8} \text{diag}([1, 1, 1])$. The Earth magnetic field considered for the control gain calculation uses a first order spherical harmonic model, whereas the real model uses a 10th order spherical harmonic model. The control law is applied to the nonlinear model of the satellite. Three cases are studied: using only magnetorquers (Figs. 1 and 2), using only reaction wheels (Fig. 3) and using hybrid actuation system (Fig. 4). In the latter case, the hybrid configuration consists in three magnetorquers associated with a reaction wheel on the roll axis.

In the case of magnetic control (Figs. 1 and 2), the satellite is stabilized within 1 orbit. The controller with updated gain leads to better results than the one with fixed gain especially in the roll stabilization. Nevertheless, the two approaches perform well and the fixed control gain approach could be used in case of strict memory or power limitations.

In the case of gyroscopic control, the satellite reaches the desired orientation in less than 0.01 orbit with very good performances. The closed-loop behavior presents a slight overshoot and fluctuations. However, the angular velocity is in this case more important than in the case of magnetic control configuration. It should be reminded that reaction wheels are more expensive and more power consuming than coils. The hybrid configuration is then studied to limit the use of reaction wheels. In the last test case, i.e. with the hybrid configuration,

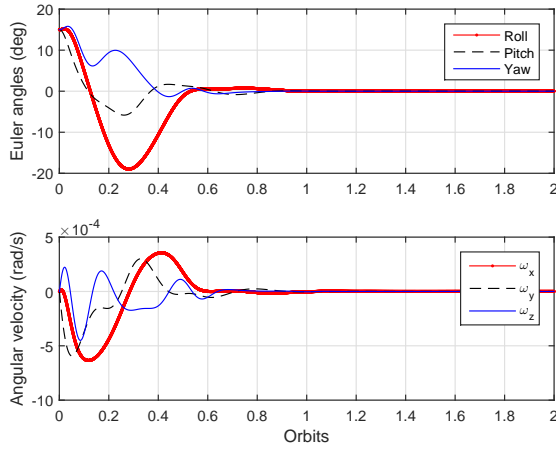


Fig. 1. Simulation results with magnetorquers - Fixed controller gain.

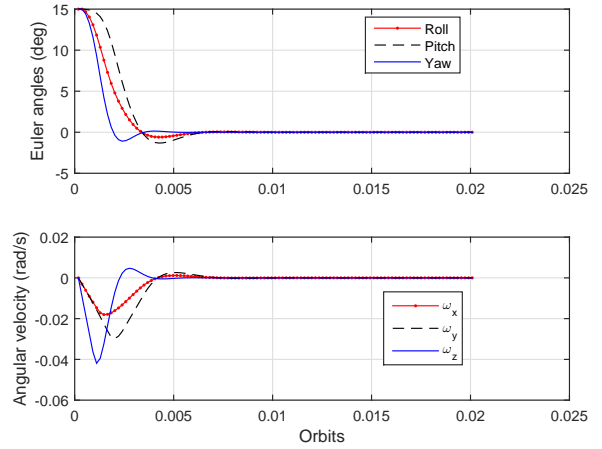


Fig. 3. Simulation results with reaction wheels.

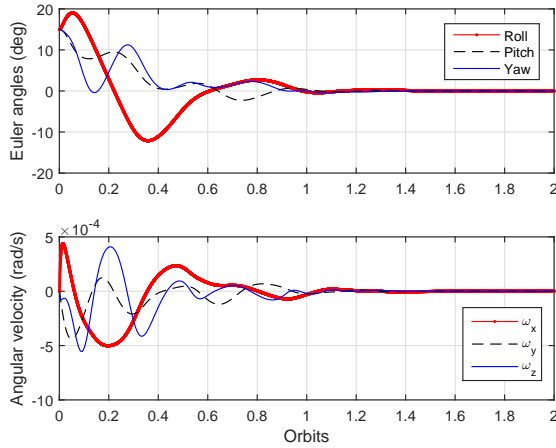


Fig. 2. Simulation results with magnetorquers - Updated controller gain.

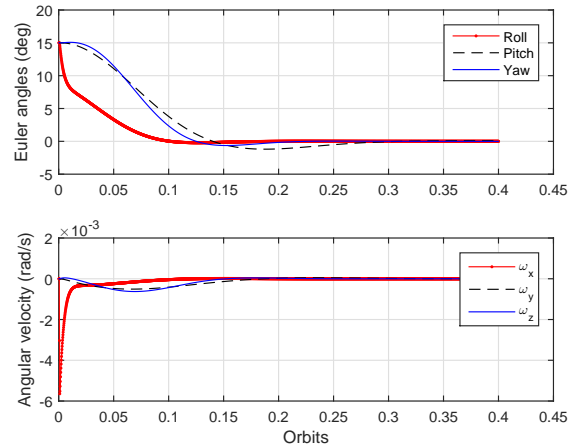


Fig. 4. Simulation results with hybrid actuation system.

the satellite is stabilized within 0.3 orbits and the angular velocities remain small. The roll angle reaches the equilibrium faster than the two other Euler angles thanks to the gyroscopic control. This configuration takes benefits from the gyroscopic control in terms of stabilisation performance, while reducing cost, weight and risk of failure.

V. CONCLUSION

In this paper, the three-axis stabilization of a nanosatellite in a polar circular Low Earth Orbit was studied. The Attitude Control System considers a linear controller and magnetic actuation system. Since this kind of actuation system is less efficient than the gyroscopic one, an hybrid actuation system is studied. It combines three magnetic coils and one reaction wheel and is shown to present the best trade-off between performance and power consumption. Further work will consider the development of control laws that ensure high performances of the magnetic control, based on the study and control of time-varying systems. The robustness of the control law with respect to measurement errors and noise will also be studied.

More specifically, the attitude determination system accuracy should be taken into account in the design and analysis of the control law performance and robustness.

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