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Spin-axis pointing of a magnetically actuated spacecraft

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ABSTRACT

Attitude regulation proves to be a challenging problem, when magnetic actuators alone are used as attitude effectors, since they do not provide three independent control torque components at each time instant. In this paper a rigorous proof of global exponential stability is derived for a magnetic control law that leads the satellite to a desired spin condition around a principal axis of inertia, pointing the spin axis toward a prescribed direction in the inertial frame. The technique is demonstrated by means of numerical simulation of a few example maneuvers. An extensive Monte Carlo simulation is performed for random initial conditions, in order to investigate the effect of changes in control law gains.

1. Introduction

The objective of this study is the determination of a control law that allows for the acquisition of a desired pure spin condition of a rigid spacecraft around one of its principal axes of inertia by means of magnetic actuators only, while aiming the spin axis in a prescribed direction in the inertial space. The use of magnetic actuators on satellites flying low earth orbits (LEOs) poses several problems in the selection of suitable control strategies because of the fact that the interaction between the local geomagnetic field and the coils generates torques that lie on a plane that is orthogonal to the magnetic field itself. This makes the system inherently underactuated, with the inability to provide three independent control torques at each time instant. As a consequence, the application of well known control strategies is no longer possible in this case for both regulation and tracking of desired attitudes.

The interest in magnetic actuators is due to different reasons [1]. First of all, savings in weight, cost and complexity with respect to other systems are significant. At the same time, the possibility of a smooth modulation of control torques greatly limits the interaction of the attitude control system with flexible modes. In spite of the presence of only two magnetic control components, attitude stabilization is possible because, on average, the system proves to be controllable if the orbit possesses an adequate inclination with respect to the geomagnetic equator [2]. This mechanism is based on the fact that the magnetic field vector periodically rotates over inclined orbits as the spacecraft completes its movement around the Earth, making the problem intrinsically time-varying.

Thrusters allow for a faster angular momentum dumping, but this comes at the cost of large propellant consumption and the added complexity of a fuel system (tanks, pipes, valves, etc.) feeding a minimum of eight thrusters required for full three-axis control. Moreover, pulse modulation may cause strong interactions of attitude dynamics with fuel sloshing and flexible modes, thus harming pointing precision [3]. When momentum and reaction wheels are used, for example in large satellites with stringent pointing requirements, the need of redundancy and the presence of high spin-rate elements make the system more complex, heavy and prone

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to (possibly multiple) failure(s). In addition, the problem of wheel desaturation arises when the effects of environmental torques need to be compensated, so that other means are needed to unload the wheel cluster, such as thrusters or magnetic torquers [4,5].

When three independent control torques are available, the problem of attitude control can be conveniently expressed in terms of Euler axis/angle, where a nominal Euler axis rotation directly takes the spacecraft from an initial attitude to a desired final one by means of the minimum angular path. This maneuver strategy can be implemented by means of a quaternion feedback control [6]. If, on the converse, the spacecraft is underactuated, e.g. when magnetic torquers are the only attitude effectors available, the nominal rotation cannot be performed.

Different approaches have been investigated in the past for tackling the problem of control in underactuated conditions [7,8], which represent an interesting solution also as a strategy for failure mitigation systems (e.g. after loss of a reaction wheel in a three-axes stabilized spacecraft with no redundancy) [9–11], possibly at the cost of reduced closed-loop performance in terms of pointing accuracy and/or maneuver time. In [12] it is shown that a feasible rotation always exists around a non-nominal Euler axis that lies on the plane perpendicular to the torqueless direction and allows for exact pointing of a given body-fixed axis toward any prescribed target direction.

Such a capability may be useful in several mission scenarios. As an example, Earth pointing may be required in the case of Earth observation, with the accurate alignment of the boresight of a sensor payload. This is the case of ALMASat-EO, a microsatellite under development at the University of Bologna, mounting an innovative multispectral camera and testing a novel micropropulsion system for orbit control [13]. In other cases accurate pointing may be required for communication purposes with a directional antenna. Another application may be finally represented by the orientation of a thruster nozzle for precise orbit maneuvers, where the accuracy must lie within a fraction of a degree.

In Cheon et al. [14], the problem of target pointing is tackled in the case where only magnetic devices are used, more specifically magnetometers and magnetic torquers, with the function, respectively, of attitude sensors and actuators. As a major limitation, the approach, derived after a linearization of the governing equations of motion, provides local asymptotic stability only for the resulting controller. In [15] a pointing control law is proven to asymptotically stabilize an axisymmetric spacecraft under controller saturation and multiple failures of up to two magnetic torque-rods, with good numerical results also extended to satellites with triaxial inertia properties.

In this paper a continuous magnetic torque command is proposed that leads a three-inertial spacecraft to a desired spin condition around a principal axis of inertia that is aligned toward a target direction, fixed in the inertial reference frame. A simple control law based on angular momentum shaping successfully achieves the mission task. The analysis follows the approach introduced in [16], where convergence toward a desired spin rate was proven by demonstrating robustness of global exponential stability (GEAS) of a nominal system by means of generalized exponential asymptotic stability in variations (GEASVs) tools [17,18]. In this framework, the error dynamics equation is first derived for two error signals, namely the angular momentum error in the body frame and the angular momentum error with respect to the desired direction of the spin axis in the inertial frame. The error dynamics can be cast in the classical form of a nominal system perturbed by a vanishing perturbation term. Then, after proving the generalized exponential stability for the nominal system, such result is extended to the perturbed system [19].

The control law is then tested by means of numerical simulations, in order to demonstrate performance and stability characteristics of the method. In particular, a Monte Carlo approach is used to empirically evaluate the convergence capability of the controller to obtain single-axis pointing from arbitrary initial conditions and determine average convergence time as a function of control gains.

2. Mathematical preliminaries

The dynamic model of a rigid satellite expressed in a set of principal axes of inertia, $\mathbb{F}_B = \{P; \hat{\boldsymbol{e}}_1, \hat{\boldsymbol{e}}_2, \hat{\boldsymbol{e}}_3\}$, centered in the spacecraft center of mass P, is given by Euler equation

$$\dot{\boldsymbol{h}} = \boldsymbol{M} - \boldsymbol{\omega} \times \boldsymbol{h} \tag{1}$$

where $\boldsymbol{h} = (h_1, h_2, h_3)^T = \boldsymbol{J}\boldsymbol{\omega}$ is the absolute angular momentum vector represented in terms of components in the body-fixed frame, $\mathbb{F}_B$, $\boldsymbol{\omega} = (\omega_1, \omega_2, \omega_3)^T$ is the absolute velocity vector, $\boldsymbol{J} = \mathrm{diag}(J_1, J_2, J_3)$ is the spacecraft inertia matrix, and $\boldsymbol{M} = \boldsymbol{M}^{(c)} + \boldsymbol{M}^{(d)}$ is the external torque vector, given by the sum of control and disturbance torques, respectively. No external disturbances will be considered in the present analysis (that is, $\boldsymbol{M}^{(d)} = \boldsymbol{0}$).

For a spacecraft controlled by means of three mutually orthogonal magnetic coils, the control torque generated by the magnetorquers is given by

$$\boldsymbol{M}^{(c)} = \boldsymbol{m} \times \boldsymbol{b} \tag{2}$$

where $\boldsymbol{m}$ is the commanded magnetic dipole moment vector generated by the coils and $\boldsymbol{b} = \mathbb{T}_{BO}\boldsymbol{b}_O$ is the local geomagnetic field vector expressed in terms of body-frame components.

A circular low earth orbit of radius r_c, period T_{orb}, and orbit rate $\Omega = 2\pi/T_{orb}$ is considered. The components $\boldsymbol{b}_O$ of the geomagnetic field are expressed in the local-vertical/local-horizontal orbit frame, $\mathbb{F}_O$, by means of a simple tilted dipole model [20,21], where the z_O-axis lies along the local vertical, the y_O-axis is normal to the orbit plane, in a direction opposite to the orbital angular speed $\boldsymbol{\omega}^{orb}$, and the transverse axis x_O completes a right-handed triad, in the direction of the orbital velocity. The geometry of the problem is sketched in Fig. 1.

The coordinate transformation matrix between $\mathbb{F}_O$ and $\mathbb{F}_B$

$$\mathbb{T}_{BO} = (\overline{q}^{\,2} - \boldsymbol{q}^T\boldsymbol{q})\mathbb{I}_3 + 2\boldsymbol{q}\boldsymbol{q}^T - 2\overline{q}\,\tilde{\mathbb{Q}} \tag{3}$$