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IFAC-PapersOnLine 49-17 (2016) 278–283

Control of Magnetic Space Tug

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Abstract: Magnetic tugging of a target satellite without thrust capacity can be interesting in various contexts. In this paper, the dynamics of such a 2-satellites formation is derived and linearised about a nominal configuration which is not necessarily constant. Analytical expressions are given for the different forces and torques differentials. Two LQ-based controllers are given, depending on the capacity of the target to control its own attitude. Linear simulations of the closed loop system are realised and compared with the full order non-linear model. The results obtained are promising and consistent with previous research.

Keywords: Formation flying; Spacecraft dynamics, navigation, guidance and control

1. INTRODUCTION

Satellite tugging can be motivated by various reasons: de-orbiting or re-orbiting, necessary in the case of satellites end-of-life; orbit control for formations of several satellites in which only one is equipped with thrusters; or to finalize launches, in which case this manoeuvre would replace the last stage of the launcher.

Several means can be considered to modify the orbit of a satellite by tugging it with another satellite. Indeed, one could simply dock a chaser/tug satellite to the target/tugged satellite. Contactless solutions however could be more interesting, as they could provide a way to avoid standardized interfaces and hazardous docking phases. They may also help preventing the creation of new debris.

In the same context as Voirin et al. (2012), we propose using magnetic forces to tug the target. Indeed many satellites, especially in Low Earth Orbit, are equipped with Magnetic Torque Bars (MTQs), used for attitude control. A chaser equipped with a powerful magnetic dipole could generate forces and torques on the target.

Electromagnetic Formation Flying has been studied since the beginning of the 21st century. Schweighart (2005) computed the dipoles to apply to make a N-satellites formation follow a given trajectory in free space; Elias et al. (2007) gave a way to control the relative position of a formation, while controlling each satellite attitude with reaction wheels; Sakai et al. (2008) solved the guidance to keep the same position in time and suggested to modulate the dipole with sine waves to avoid the problem caused by the constant torque due to the Earth magnetic field; Ahsun et al. (2010) improved the work done by Elias et al. (2007) and applied an idea similar to Sakai et al. (2008). Recently, Huang et al. (2016) started looking for configurations enabling to reduce the total momentum on a 2-satellites formation.

All the previously cited references assumed the dipoles to be located at the center of mass of the satellites, and supposed all satellites equally capable of steering their dipole and controlling their attitude. In this study, we consider a lever-arm on the target dipole, and assume both the value and orientation of the target dipole fixed in its body frame. The target attitude will be supposed uncontrolled in some examples. Finally, a constant thrust from the chaser is considered, which changes the dynamics compared to the given references.

In a paper to be published, the authors will demonstrate the existence of nominal relative configuration trajectories enabling to magnetically tug a target satellite, while avoiding accumulating angular momentum because of the Earth magnetic field, without waving the dipoles.

This paper focuses on the control of the formation around these configuration trajectories. The system considered is described in section 2; the equations of motion are derived in section 3; they are linearised in section 4 while the efforts are differentiated in section 5; finally, section 6 presents two possible controllers.

2. SYSTEM DESCRIPTION

The system considered is composed of a target satellite denoted by the subscript T and a chaser satellite subscripted C. As presented in Fig. 1, the target is equipped with an MTQ turned on which dipole is equal to the constant $\boldsymbol{\mu}_T = [0\ 500\ 0]^T$ A m^2 in its body frame (value reached by Sentinel 2 satellite for example[1]), located at the body-frame constant position $\boldsymbol{\gamma}_{\mu_T} = [1\ 0\ 0]^T$ m; the target mass is $\mathbf{m}_T = 2300$ kg; its inertia tensor is $\mathbf{J}_T = \mathrm{diag}([1300\ 1100\ 700])$ kg m^2. The chaser is characterised by $\mathbf{m}_C = 1000$ kg; $\mathbf{J}_C = 700\ \mathbf{I}_3$ kg m^2, where $\mathbf{I}_3$ is the identity matrix. Its dipole is located at its center of mass ($\boldsymbol{\gamma}_{\mu_C} = \mathbf{0}$).

⋆ E. Fabacher wishes to thank Airbus DS and the European Space Agency for the financial support provided for his NPI PhD.

[1] http://emits.sso.esa.int/emits-doc/ESTEC/Sentinel-1-FP7-Industry-Day-Nov-07.pdf

10.1016/j.ifacol.2016.09.048

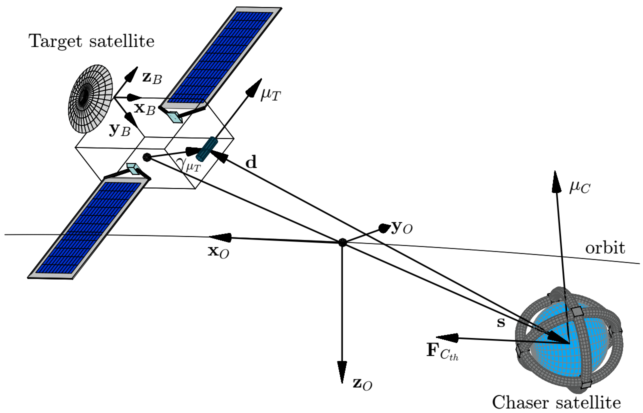

Fig. 1. Vectors and frames used in this article.

For a given vector $\mathbf{x}$, x is the norm of the vector, $\hat{\mathbf{x}}$ is the unitary vector associated, $\dot{\mathbf{x}}$ is its time derivative in the specified reference frame. $[\mathbf{x}^{\times}]$ is the skew-symmetric matrix denoting the cross-product $\mathbf{x}\times$, $\cdot$ denotes the scalar product. Four frames are used in this article; I, an inertial frame centred on the Earth; O, the orbital frame centred on the center of mass of the formation; B_i, the body frame linked to satellite i (target if no i specified).

3. DYNAMICS

3.1 Relative Position Dynamics

Law describing the evolution of the relative position $\mathbf{s}$ between the target and the chaser has been showed in Fabacher et al. (2015). In inertial frame, this evolution is given by:

$$\left.\frac{d^2\mathbf{s}}{dt^2}\right|_I = -\frac{\mu}{r^3}\mathbf{M}\mathbf{s} + \frac{\mathbf{F}_{C_{th}}}{m_C} - \frac{\mathbf{F}_{T_{\epsilon\mu}}}{m_{CT}} \quad (1)$$

where $\mathbf{M}$ is the Jacobian matrix describing the linearisation of the Earth gravity with regard to the position around the center of mass of the formation, and can be obtained from Wie (1998). μ is the standard gravitational parameter of the Earth; r is the distance between the center of the Earth and the formation's center of mass; $\mathbf{F}_{C_{th}}$ is the force created by the chaser's thrusters; $\mathbf{F}_{T_{\epsilon\mu}}$ is the magnetic force created by the chaser's magnetic dipole on the target's magnetic dipole; m_{CT} is the reduced mass of the system ($m_{CT} = \frac{m_C m_T}{m_C + m_T}$).

Differentiating twice $\mathbf{s}$ in the orbital frame gives (2):

$$\left.\frac{d^2\mathbf{s}}{dt^2}\right|_I = \left.\frac{d^2\mathbf{s}}{dt^2}\right|_O + \boldsymbol{\omega}\times(\boldsymbol{\omega}\times\mathbf{s}) + 2\boldsymbol{\omega}\times\left.\frac{d\mathbf{s}}{dt}\right|_O + \left.\frac{d\boldsymbol{\omega}}{dt}\right|_O \times \mathbf{s} \quad (2)$$

where $\boldsymbol{\omega}$ is the rotational rate vector from inertial to orbital frame. We define $\boldsymbol{\eta} = \frac{d\boldsymbol{\omega}}{dt}$ which, as $\boldsymbol{\omega}$, depends on the position of the formation in its orbit. The equation of the relative motion in the orbital frame is therefore:

$$\ddot{\mathbf{s}} + \boldsymbol{\omega}\times(\boldsymbol{\omega}\times\mathbf{s}) + 2\boldsymbol{\omega}\times\dot{\mathbf{s}} + \boldsymbol{\eta}\times\mathbf{s} + \frac{\mu}{r^3}\mathbf{M}\mathbf{s} = \frac{\mathbf{F}_{C_{th}}}{m_C} - \frac{\mathbf{F}_{T_{\epsilon\mu}}}{m_{CT}} \quad (3)$$

3.2 Attitude Dynamics

In the body frame, the evolution of the attitude of one of the satellites is classically described by:

$$\mathbf{J}\frac{d\boldsymbol{\omega}_{B/I}}{dt} + \boldsymbol{\omega}_{B/I}\times\mathbf{J}\boldsymbol{\omega}_{B/I} = \sum\boldsymbol{\tau} \quad (4)$$

In the case of Electromagnetic Formation Flight, $\sum\boldsymbol{\tau}$ can be developed in:

$$\sum\boldsymbol{\tau} = \boldsymbol{\tau}_{\epsilon\mu} + \boldsymbol{\tau}_{\gamma} + \boldsymbol{\tau}_{\epsilon\mu_E} + \boldsymbol{\tau}_{rw} + \boldsymbol{\tau}_g + \boldsymbol{\tau}_p \quad (5)$$

with $\boldsymbol{\tau}_{\epsilon\mu}$ the torque on the satellite due to the magnetic field created by the other satellite; $\boldsymbol{\tau}_{\gamma} = \gamma_{\mu_T}\times\mathbf{F}_{\epsilon\mu}$ the torque created by the cross product of the satellite center of mass to dipole lever-arm with the magnetic force created by the other satellite; $\boldsymbol{\tau}_{\epsilon\mu_E}$ the torque on the satellite due to the Earth magnetic field; $\boldsymbol{\tau}_{rw}$ the torque created by a reaction wheel system (or other similar devices); $\boldsymbol{\tau}_g$ the torque due to the gravity gradient; $\boldsymbol{\tau}_p$ the rest of the perturbing torques.

3.3 Nominal States

The guidance of the formation is developped in Fabacher et al. (2015) and will be further studied in a future reference. Because r, $\boldsymbol{\omega}$, $\boldsymbol{\eta}$ and the Earth magnetic field vary in orbit, the nominal parameters states also depend on the time. They solve the system of differential equation formed by (3), (4) adapted for the chaser and (4) adapted for the target. In the scope of this article, we will consider that for every time t, a nominal configuration trajectory has been found and is described by a combination $\left(\mathbf{s}, \dot{\mathbf{s}}, \boldsymbol{\theta}_C, \boldsymbol{\omega}_{B/O_C}, \boldsymbol{\theta}_T, \boldsymbol{\omega}_{B/O_T}, \mathbf{F}_{C_{th}}, \boldsymbol{\tau}_{C_{rw}}, \boldsymbol{\mu}_C, \boldsymbol{\tau}_{T_{rw}}\right)_{nom}(t)$.

4. LINEARISATION: DYNAMICS AND KINEMATICS

In this section, the dynamics equations are differentiated around a nominal trajectory. The aim is to obtain a time-varying state space system representation which will be used to synthesize controllers.

4.1 Translational Motion

Let's differentiate (3):

$$\boldsymbol{\delta}\ddot{\mathbf{s}} + 2\left[\boldsymbol{\omega}^{\times}\right]\boldsymbol{\delta}\dot{\mathbf{s}} + \left(\left[\boldsymbol{\omega}^{\times}\right]^2 + \left[\boldsymbol{\eta}^{\times}\right] + \frac{\mu}{r^3}\mathbf{M}\right)\boldsymbol{\delta}\mathbf{s} = \frac{\boldsymbol{\delta}\mathbf{F}_{C_{th}}}{m_C} - \frac{\boldsymbol{\delta}\mathbf{F}_{T_{\epsilon\mu}}}{m_{CT}} \quad (6)$$

$\boldsymbol{\delta}\mathbf{F}_{T_{\epsilon\mu}}$ is the differentiation of the magnetic force considered in the problem. Part of it comes from the relative position, the rest comes from the control inputs $\boldsymbol{\delta\mu}_C$.

$$\boldsymbol{\delta}\mathbf{F}_{T_{\epsilon\mu}} = \frac{\partial\mathbf{F}_{T_{\epsilon\mu}}}{\partial\mathbf{s}}\boldsymbol{\delta}\mathbf{s} + \frac{\partial\mathbf{F}_{T_{\epsilon\mu}}}{\partial\boldsymbol{\mu}_C}\boldsymbol{\delta\mu}_C \quad (7)$$

Similarly, $\boldsymbol{\delta}\mathbf{F}_{C_{th}}$ is the differentiation of the thruster force considered in the problem. It is considered as a control input. The previous equation can be derived in a state space representation as follows:

$$\begin{bmatrix}\boldsymbol{\delta}\dot{\mathbf{s}}\\ \boldsymbol{\delta}\ddot{\mathbf{s}}\end{bmatrix} = \mathbf{A}_s\begin{bmatrix}\boldsymbol{\delta}\mathbf{s}\\ \boldsymbol{\delta}\dot{\mathbf{s}}\end{bmatrix} + \mathbf{B}_s\begin{bmatrix}\boldsymbol{\delta}\mathbf{F}_{C_{th}}\\ \boldsymbol{\delta\mu}_C\end{bmatrix} \quad (8)$$

With:

$$\mathbf{A}_s = \begin{bmatrix}\mathbf{0}_3 & \mathbf{I}_3\\ -\left(\left[\boldsymbol{\omega}^{\times}\right]^2 + \left[\boldsymbol{\eta}^{\times}\right] + \frac{\mu}{r^3}\mathbf{M}\right) & -2\left[\boldsymbol{\omega}^{\times}\right]\end{bmatrix} + \begin{bmatrix}\mathbf{0}_3 & \mathbf{0}_3\\ -\frac{1}{m_{CT}}\frac{\partial\mathbf{F}_{T_{\epsilon\mu}}}{\partial\mathbf{s}} & \mathbf{0}_3\end{bmatrix} \quad (9)$$

and

$$\mathbf{B}_s = \begin{bmatrix}\mathbf{0}_3 & \mathbf{0}_3\\ \frac{1}{m_C}\mathbf{I}_3 & \frac{-1}{m_{CT}}\frac{\partial\mathbf{F}_{T_{\epsilon\mu}}}{\partial\boldsymbol{\mu}_C}\end{bmatrix} \quad (10)$$