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Maintenance of satellite formations using environmental forces

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ABSTRACT

This paper examines the maintenance of satellite formations using two environmental forces: solar radiation pressure and aerodynamic forces. It is assumed that the satellites are equipped with solar flaps or aerodynamic flaps. Control using aerodynamic flaps is considered for satellite formations in LEO while solar flaps is applied to formations in LEO as well as GEO. The simple control laws based on open-loop and closed-loop control methods are designed for required rotation of the flaps to achieve desired formation keeping. The feasibility of the proposed schemes is proven via stability analyses followed by numerical simulations. A linear flap rotation scheme is found to keep the relative position errors bounded to ± 5 m. The proposed control methods show the effectiveness of the use of solar radiation pressure and aerodynamic forces for satellite formation flying.

1. Introduction

Satellite formation flying has been identified as a key technology for current and future space missions. Some of these missions include PRISMA [1], TanDEM-X [2], JC2Sat [3], PROBA-3 [4] and Magnetospheric Multiscale mission [5]. Formation flying involves significant challenges ranging from formation initialization to reconfiguration. The formations tend to break down because of various disturbances such as J2, and must be maintained by application of thrusts which require fuel. On-board fuel is a scarce commodity in space; hence, there have been some efforts to develop propellant-free formation maintenance schemes using solar radiation pressure (SRP) and/or aerodynamic forces. Leonard et al. [6] have applied along-track input in the form of differential aerodynamic drag for formation keeping. A nonlinear algorithm based on the control values $-a$, 0, and $+a$, where a is the magnitude of the differential drag accelerations, is developed for achieving desired maneuvers. Matthews and Leszkiewicz [7], and Steffy et al. [8] varied ballistic coefficients of the spacecraft for maintaining the formation. Aorpimai et al. [9] assumed satellite equipped with aerodynamic wings and varied the angle of attack to establish the formation. The New Millennium Program Earth Observation-1 (NMP EO-1) is an example of successful application of aerodynamic drag for formation flying [10]. In Ref. [10] a modified Clohessy–Wiltshire equation which includes quadratic drag is studied. This modified Clohessy–Wiltshire equation can find application in station keeping and formation flying. In Ref. [11], the use of differential drag as a means for nanosatellite formation control is studied and a simple PI control law is derived to adjust the cross-sectional area of the satellites.

On the other hand, the solar radiation pressure for formation flying is examined by Williams and Wang [12]. They considered a satellite with a solar wing and it was shown that a solar wing of the correct area can prevent the secular out-of-plane growth in a low-Earth orbit formation that is caused by differential nodal drift. Fourcade [13]

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found that SRP is successful in maintaining the desired formation. Other recent research has studied SRP based tetrahedron satellite formations [14]. This research studied the use of SRP to deploy and stabilize a three-dimensional satellite formation for a Heliocentric as well as High Earth Orbit, with the diameter of the formation and the stabilization time dependent on the properties of the solar sail.

The present paper examines the feasibility of formation maintenance using environmental forces especially SRP and aerodynamic (AERO) forces. It is assumed that the satellites are equipped with solar flaps (called SRP flaps) or aerodynamic flaps (called AERO flaps). By appropriate rotation of these SRP/AERO flaps, it is possible to influence the relative motion between satellites in a formation. For an illustration of this strategy, the control laws for rotations of the SRP and AERO flaps are developed. Open-loop and closed loop control laws (based on linear quadratic regulator (LQR) and sliding mode control (SMC)) are considered for the AERO based control while the closed loop control (proportional type) is designed for the SRP based control. It is to be noted that the consideration of the different types of control laws for the AERO and SRP based controls (such as SMC or proportional type) is just for an illustration purpose. The SRP based control can use the LQR and SMC type controllers. Finally, the feasibility of the proposed strategy is tested via numerical simulations. The major contributions of the paper are as follows: (1) simple mechanisms for harnessing SRP/AERO forces for formation control are proposed, (2) new simple control laws for the AERO-based control are designed and their efficacies are validated through stability analysis and numerical simulations, and (3) new simple control laws for the SRP-based control are designed and their performances are examined through stability analysis and numerical simulations.

The paper is organized as follows: Section 2 presents the system model followed by the design of the controller in Section 3. The results of the numerical simulations are discussed in Section 4. Finally, the conclusions of the present investigations are summarized in Section 5.

2. System model

2.1. System equations of motion

The system comprises of a leader satellite and a follower satellite (Fig. 1). Both the satellites are equipped with SRP flaps or AERO flaps. A simple mechanism using a servo is used to rotate each flap; by rotating these flaps, differential SRP/AERO forces between the satellites are created for required formation keeping or manoeuvring. The orbital motion of the leader satellite is defined by a radial distance r_c from the center of the Earth and a true anomaly θ. The motion of the follower satellite is described relative to the motion of the leader satellite by a reference frame $S-xyz$ fixed at the center of the leader satellite. The x-axis points along the local vertical, the z-axis is taken along normal to the orbital plane, and the y-axis represents the third axis of this right-handed frame. The system equations of motion are as follows [15,16]:

$$\ddot{r}_c = r_c\dot{\theta}^2 - \frac{\mu}{r_c^2}, \quad \ddot{\theta} = -2\dot{\theta}\frac{\dot{r}_c}{r_c} \tag{1}$$

$$\ddot{x} = x\dot{\theta}^2 + y\ddot{\theta} + 2\dot{y}\dot{\theta} - \frac{\mu(r_c+x)}{r^3} + \frac{\mu}{r_c^2} + f_x + f_{dx} \tag{2}$$

$$\ddot{y} = y\dot{\theta}^2 - 2\dot{x}\dot{\theta} - x\ddot{\theta} - \frac{\mu y}{r^3} + f_y + f_{dy} \tag{3}$$

$$\ddot{z} = -\frac{\mu z}{r^3} + f_z + f_{dz} \tag{4}$$

where r is the radial distance of the follower satellite from the center of the Earth, $r = \left[(r_c+x)^2 + y^2 + z^2\right]^{(1/2)}$. f_j and f_{dj} are the control input and external disturbance about the jth direction (for $j = x,y,z$), respectively.

The linear equations of motion with inclusion of J_2 perturbations, given by Schweighart and Sedwick [17] are as follows:

$$\ddot{x} - 2(nc)\dot{y} - (5c^2-2)n^2x = -\frac{3}{2}n^2J_2\left(R_e^2/r_c\right) \times \left\{1 - 3\sin^2(kt) - \frac{1}{4}[1+3\cos(2i_c)]\right\} + f_{dx} \tag{5}$$

$$\ddot{y} + 2(nc)\dot{x} = -3n^2J_2\left(R_e^2/r_c\right)\sin^2(i_c)\sin(kt)\cos(kt) + f_{dy} \tag{6}$$

$$\ddot{z} + q^2 z = 2lq\cos(qt+\varphi) + f_{dz} \tag{7}$$

where n is the mean orbital rate of the circular reference orbit, $n = \sqrt{\mu/r_c{}^3}$; $c = \sqrt{1+s}$; $s = \left(3J_2R_e^2/8r_c^2\right)[1+3\cos(2i_c)]$; J_2 is the second spherical harmonic of Earth's geopotential; R_e is the radius of the Earth; i_c is the initial inclination of the circular reference orbit; $k = nc + \left(3nJ_2R_e^2/2r_c^2\right)\cos^2(i_c)$; t is the time; ϕ is the initial phasing angle for cross track motion (set to 0 in the simulation);

$$q = nc - \left(\cos\gamma_0 \sin\gamma_0 \cot\Omega_{s0} - \sin^2\gamma_0 \cos i_{sat}\right) \times \left(\dot{\Omega}_{sat} - \dot{\Omega}_{ref}\right) - \dot{\Omega}_{sat}\cos i_{sat};$$

$$\gamma_0 = \cot^{-1}\left[\frac{\cot i_{sat}\sin i_c - \cos i_{sat}\cos\Omega_{s0}}{\sin\Omega_{r0}}\right];$$

$$\Omega_{s0} = \frac{z_0}{r_c \sin i_c}; \quad i_{sat} = \frac{\dot{z}_0}{kr_c} + i_c;$$

$$\Phi_0 = \cos^{-1}[\cos i_{sat}\cos i_c + \sin i_{sat}\sin i_c\cos\Omega_{s0}];$$

$$\dot{\Omega}_{sat} = \frac{-3nJ_2R_e^2}{2r_c^2}\cos i_{sat};$$

$$l = -r_c\frac{\sin i_{sat}\sin i_c\sin\Omega_{s0}}{\sin\Phi_0}\left(\dot{\Omega}_{sat} - \dot{\Omega}_c\right);$$

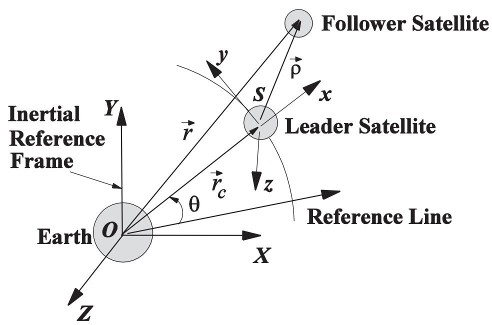

Fig. 1. Geometry of orbit motion of leader and follower satellites [15].