Aerospace Science and Technology 42 (2015) 342–352

Contents lists available at ScienceDirect

Aerospace Science and Technology

www.elsevier.com/locate/aescte

Deployment and control of spacecraft solar array considering joint stick-slip friction

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ARTICLE INFO

Article history:
Received 19 May 2014
Received in revised form 12 January 2015
Accepted 1 February 2015
Available online 7 February 2015

Keywords:
Solar array
Dynamic modeling
Fuzzy adaptive PD control
Joint friction
Stick-slip

ABSTRACT

For solar array system, traditional modeling theories usually adopt the Cartesian coordinates to establish the dynamic equation of the system. The order of this equation is often too high and inconvenient to design control laws and the Cartesian coordinates are difficult to be measured in practice. This paper presents a modeling method for solar array system. In this method, joint coordinates of the solar panels, yoke and the spacecraft main-body are defined as the generalized variables. In this way, the dynamic equation derived here possesses lower order than those derived by other methods used before, which facilitates further control design. Moreover, the joint coordinates can be easily measured in practice. On the other hand, joint friction is an important issue to be considered. Joint friction could affect the stability and control precision of the system. Most existing studies on joint friction of solar array system are over-simplified into plane problems, and as a result, inaccuracy of friction force calculation output was incurred. In this paper, a three-dimensional revolute joint model is introduced, the calculation of joint friction is discussed in detail, and the relationship between ideal constraint force and Lagrange multipliers is derived. In addition, the control design of the solar array system is discussed and the fuzzy adaptive PD control method is used for controller design. At the end of this paper, the validity of the above studies is verified by the comparisons of numerical simulations with the ADAMS software.

1. Introduction

Solar array, which provides necessary power for the whole system, is a vital component of the spacecraft. The first task for a spacecraft in space is to deploy its solar array. Hence, the failure of the solar array deployment would be a disaster for a space mission. Multiple-panel solar arrays are folded during spacecraft launch and ascent. After the spacecraft and launch vehicle are separated and the spacecraft is turned into the free flying orbit, solar panels will be freed from their fixation position and deployed by driving torsion spring. Transient impact occurs at the moment of full deployment of solar arrays, and it will disturb the spacecraft flight or even may destroy the structure.

Up to now, the deployment of solar array has been studied by many researchers. For example, Wie et al. [14] presented the dynamic and digital simulation of the deployment of rigid solar panels on the INTELSAT-V and INSAT spacecrafts, and described two different deployment mechanisms used on the two spacecrafts. Wallrapp and Wiedemann [12] simulated the deployment of a satellite solar array using the multibody program SIMPACK and the comparison of the results shows that flexible bodies cause a slightly changed torque in the closed cable loops. Kojima et al. [6] simulated the ADEOS spacecraft attitude response due to the stick-slip effect and evaluated their mathematical model using the ADEOS flight data and ground-based test data. Kuang et al. [8] investigated the multibody dynamics of a satellite and the numerical simulation showed that the chaotic dynamics is very sensitive to the initial conditions from the time evolution history of the variables of the attitude motions. Kote et al. [7] analyzed the influence of deployment and latching of solar panel on spacecraft's attitude by deriving the equations of motion with Lagrangian formulation and solving the equations using numerical method. Kwak et al. [9] researched the dynamic modeling of satellite with deployable solar arrays equipped with strain energy hinges and the simulation result was similar to the ground experimental results. Ding et al. [3] provided a method of using a root hinge drive assembly (RHDA) to control the solar array deployment and established multi-DOF mechanism dynamic model of the solar array deployment system. However, those studies mentioned above have seldom focused on the effect of joint friction of solar array. The solar panels are connected by revolute joints. So friction inevitably exists in the joints. In addition, the deployment may cause the change of spacecraft attitude, so control should be taken into account to eliminate this

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http://dx.doi.org/10.1016/j.ast.2015.02.001
1270-9638/

change. Certain reasonable control design may achieve the goal of adjusting the attitude of spacecraft effectively by using relatively less energy.

In this paper, dynamics and control of a solar array system are studied by considering the joint friction. Dynamic model of the system is established and the contribution of joint friction to the dynamic equation of the system is derived. The fuzzy adaptive PD controller is designed to control the attitude change of the system caused by the deployment of solar arrays. This paper is organized as follows. Section 2 introduces the structure of solar array system, including the closed cable loops and the latch mechanisms. In Section 3, the dynamic equation of the system is established using the single direction recursive construction method and the contribution of joint friction to the system equation is deduced using the virtual power theory. Section 4 presents the control design for the system. In Section 5, numerical simulations are carried out to validate the theoretical studies in this paper. Finally, a concluding remark is given in Section 6.

2. Introduction of solar array system

In this section, the structure of spacecraft with solar arrays adopted in this paper is firstly introduced, then the equipment of closed cable loop (CCL) [14] used for synchronizing deployment of solar arrays is introduced, and finally a fine description of the latch mechanisms of solar arrays is given [2].

2.1. Structure of spacecraft system

The spacecraft system adopted in this paper is shown in Fig. 1. The system consists of a main-body, a yoke and three solar panels. The solar array is in a topological-tree configuration. The deployment mechanism consists of four torsion springs and three close cable loops. The preloaded torsion springs located at each revolute joint provide the energy to deploy the arrays.

The drive torque $\boldsymbol{T}_{drive}(i,1) = T^i_{drive}$ on the i-th $(i = 1,\dots,4)$ joint can be represented as

$$T^i_{drive} = k_{drive}(\bar{\theta}_i - \theta_i) \tag{1}$$

where k_{drive} is the torsional stiffness of torsion spring in unit of N m/rad; $\bar{\theta}_i$ and θ_i are the preload angle and practical angle of the i-th joint in unit of radians, respectively. When the deployment completed, the preload torque should be 0 N m, so the preload angle of the spring between the yoke and the spacecraft main-body is 0.5π, and the preload angle of spring between any two adjacent panels is π.

2.2. Closed cable loops (CCL)

The closed cable loops (Fig. 1) restrain the deployment by synchronizing the deployment angles during the deployment [14]. The first cable connects the first panel and the spacecraft main body, the second cable connects the second panel and the yoke, and the third cable connects the third and first panels. These cables synchronize the deployment angles of each panel by applying a "passive-control" torque which is proportional to the angle difference. These cable torques shown in Fig. 2 are simply modeled as

$$T^1_{ccl} = k^1_{ccl}(2\theta_1 - \theta_2) \tag{2a}$$

$$T^2_{ccl} = k^2_{ccl}(\theta_2 - \theta_3) \tag{2b}$$

$$T^3_{ccl} = k^3_{ccl}(\theta_3 - \theta_4) \tag{2c}$$

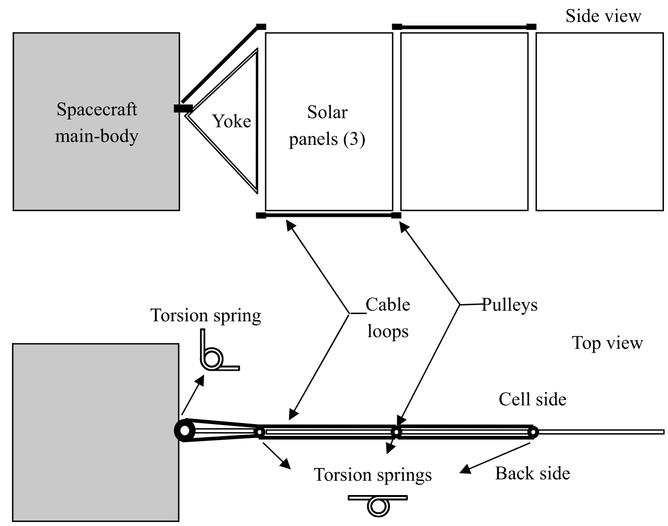

Fig. 1. The structure of solar array system.

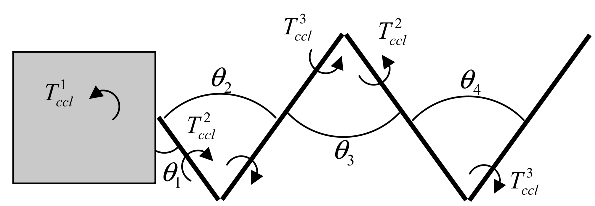

Fig. 2. Cable torque analysis model.

where T^i_{ccl} is the i-th cable torque in unit of N m, k^i_{ccl} $(= 4500 r^{i\,2}_{ccl})$ is the equivalent cable torsional stiffness in unit of N m/rad, and r^i_{ccl} is the cable pulley radius at joints in unit of m $(r^1_{ccl} = 2r^2_{ccl} = 2r^3_{ccl})$ [14], so

$$\frac{1}{4}k^1_{ccl} = k^2_{ccl} = k^3_{ccl} \tag{3}$$

2.3. Latch mechanisms

Latch mechanisms are widely used in the solar array deployment system. A schematic diagram of the latch mechanism is shown in Fig. 3. The joint connects two bodies which are fixed on A and B separately, and the two bodies can rotate relatively round the joint D. The cam C is fixed on A, and the pin E can move on the surface of C. When the joint is deployed and reaches the expected angle (0.5π or π), the pin slide into the groove G, thus to realize the locking of the two bodies (Fig. 3c).

A *step* function and a *bistop* function are introduced to represent the latch status and the torque, respectively, which are given by [2]

$$T_{lock}(\theta_i) = step(\theta_i, x_1, 0, x_2, 1) \times bistop(\theta_i, \dot{\theta}_i, x_3, x_4, k_{bs}, e, c, d) \tag{4}$$

where

$$step(\theta_i, x_1, h^1_s, x_2, h^2_s) = \begin{cases} h^1_s & \text{if } \theta_i < x_1 \\ h^1_s - (h^1_s - h^2_s)\left(\frac{\theta_i - x_1}{x_2 - x_1}\right)^2 \left(3 - 2 \times \frac{\theta_i - x_1}{x_2 - x_1}\right) & \text{if } x_1 \le \theta_i \le x_2 \\ h^2_s & \text{if } \theta_i > x_2 \end{cases} \tag{5}$$

and