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Active vibration suppression in flexible spacecraft with optical measurement

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ABSTRACT

With the development of the space station technology in China, the active vibration suppression of the lager flexible antennas and solar panels that mounted on the space station is demanded. The active vibration suppression is hard to achieve because these appendages are not allowed to mount actuators on it. Further, the practical goal is usually unachievable by the controller based on the unreliability or impracticality observed vibration parameters. In this paper, the active vibration suppression during the spacecraft attitude maneuver by the optical camera monitoring the dynamic behaviors of the flexible appendages is introduced under the restriction of the freedom of the actuators. The modal parameters of the flexible appendage used in the controller are obtained by the optical monitoring approach. A referenced angular velocity is set as the virtual input by the back-stepping control and Lyapunov method, and the control law is designed to track this virtual input. The constraint of the coefficient in the controller is investigated to guarantee the asymptotic convergence of the system considering angular velocity tracking error. The control manage coefficient is defined to describe the distribution coefficient of the vibration suppression in the design of control law. The relationship between the attitude maneuver accuracy and active vibration suppression is introduced. The appropriate control manage coefficient is obtained numerically. To guarantee the reliability or practicality of the designate flexible spacecraft control system, the optical measurements are used to measure the dynamic behaviors of the large flexible structures. The absence of displacement velocity sensors is compensated by the presence of appropriate dynamics in the controller. The results of simulation validate the feasibility of the proposed control strategy.

1. Introduction

The active suppression of flexible vibration during spacecraft attitude maneuvers is an intriguing problem that has recently stimulated research activities. Both small satellites with flexible booms, and large space stations composed of light deformable structures will get benefits from the development of vibration damping control. The dynamic behaviors of the large flexible structures are difficult to be predicted analytically, not to mention the unreliability or impracticality of structural tests on Earth, the performance of controller designed on the basis of the perfect model is deteriorated. These difficulties will lead to the derivation between the on-orbit behaviors of spacecraft and the preflight ground test measures or analytic predictions [1]. To overcome these problems, the adaptive control schemes, belonging to the called centralized active vibration control approach, are proposed estimating unavailable states by the observer. An alternative to this approach is the use of structures with distributed actuators and sensors, which is also called the distributed active vibration control. With the development of space station instrumented with large antennas and solar panels in China, further investigation on the practical method accomplishing the active vibration suppression for these appendages, which are not allowed to mount actuators, is expected.

The active vibration control is consisting of the centralized active vibration control and the distributed active vibration control. The former is designed for the appendages that are not collocated with distributed actuators. Researchers derived a class of centralized active vibration controllers for the flexible spacecraft [2–6]. A control approach that integrates the command input shaping and the technique of dynamic variable structure output feedback control was put forward for the vibration control of flexible spacecraft during attitude maneuver [2]. An L1 adaptive controller was developed for the pitch angle control of an orbiting flexible spacecraft with a moment producing device located on the central rigid

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body. A state predictor was added up to the controller generating the assessments of the unknown parameters for feedback [3]. Treating the vibration mode of flexible appendages as the inherent perturbation, a robust control strategy was developed for the flexible spacecraft during large-angle attitude maneuver [4]. At the terminal of the attitude maneuver, a state feedback controller was employed to damp the residual vibration of appendages. A robust fuzzy controller for attitude stabilization of a rigid platform with a flexible appendage was introduced by proposing a fuzzy observer to estimate the unavailable states [5]. Haibo Du [6] studied on a distributed attitude cooperative control strategy to solve the problem of attitude synchronization for a group of flexible spacecraft during formation maneuvers. Based on the back-stepping design, a distributed attitude cooperative control law was explicitly constructed by a modal observer. However, the dynamic behaviors of the large flexible structures are difficult to be predicted analytically, which cause that the requirements of control in the practical mission are hardly to be satisfied by the traditional vibration suppression method without measurements of the dynamics behaviors. For the distributed active vibration control, the number of materials of actuators and sensors had been investigated and fabricated over the years; some of them were shape memory alloys, piezoelectric materials, optical fibers, electro-rheological fluids, magneto-strictive materials [7]. Among all materials, piezoelectric materials were widely used as sensor and actuator because of its numerous advantages like low cost, quick dynamic response, low power consumption, excellent electromechanical coupling, large operating range, light weight and ease in bonding on structure [8–12]. Nevertheless, these actuators and sensors have changed the structure property of the flexible appendage by mechanical interfere, which is not allowed by most large light flexible structure. It implies the vibration suppression utilizing the piezoelectric sensors is not appropriate from the engineering viewpoint. Flexible appendages cannot be allocated with actuators and the structural tests on Earth are neither unreliable nor impractical. To improve such group of flexible appendages, it is significance to find a middle ground between the centralized and decentralized active vibration control. To obtain the vibration parameters, the full-bridge strain gauges and a camera [13] and a laser displacement sensor as well as a laser vibrometer [14] were employed to sense the necessary data for vibration control. The feasibility of modal modification technology in the flight was researched by V. Wickramasinghe [15]. The methods to get the modal parameters of flexible appendages through monitoring and identification approaches were proposed [16–20]. Taking advantages of the aforementioned control schemes, the centralized active vibration suppression with the optical camera monitoring the dynamic behaviors of the flexible appendages is investigated as the first contribution of this paper. The modal parameters of the flexible appendage used in the controller are obtained by the optical monitoring approach.

Despite the displacement information on the vibration can be obtained by traditional piezoelectric sensors and optical measurements, it is hard for these sensors to obtain the velocity of vibration. Although the micro-electromechanical systems (MEMS) may be used to measure the velocity of vibration in the near future, they are unacceptable for most of large light flexible appendage, because of their mechanical interface and complexity. Therefore, the alternative way is to compensate the absence of velocity vibration by the dynamics property. The controller is constructed by the back-steeping control and Lyapunov control methods for the flexible spacecraft with an actuator collocated at the spacecraft body. The constraint of the control coefficients is introduced to guarantee the asymptotical stability of the whole system, which is ignored in the previous literatures. Because the challenge of the vibration suppression during the attitude maneuver lies in the restriction of the freedom of the actuators, the relationship between the attitude maneuver and the vibration suppression should be investigated. Therefore, as the second contribution of this paper, a new variable named control manage coefficient is defined to describe the relationship between the attitude maneuver and active vibration suppression. The appropriate control manage coefficient is obtained numerically, which not only effectively damps the vibration but also greatly improves the accuracy of the attitude maneuver and the performance of the control torque. The remainder of this paper is outlined as follows. In section 2, the mathematical model of a spacecraft with flexible appendages and the optical measurement is introduced. In section 3, the referenced angular velocity is constructed by back-steeping control and Lyapunov methods, the control law is derived and the relationship between the attitude maneuver and active vibration suppression is introduced. In Section 4, the controller is tested and the appropriate control manage coefficient is obtained, followed by analyses and conclusions.

2. Mathematical models

The mathematical models of a flexible spacecraft are consisting of the kinematic and the dynamic model. The mathematical models proposed in this paper were introduced in the Ref. [8]. The dynamic behaviors of the large flexible structures are difficult to be predicted analytically, not to mention the unreliability or impracticality of structural, and the optical measurements are used to measure the displacement of the flexible appendages for the distributed sensors are not allowed to mount on the appendages. By setting the position of these optical signs, the displacements of the whole appendages can be monitored.

2.1. Mathematical equations

The flexible spacecraft is composed of a rigid main body and flexible appendages. The kinematics equations are defined by the attitude motion of the main body and expressed by the unit quaternion as

$$\begin{bmatrix} \dot{q}_0 \\ \dot{\boldsymbol{q}} \end{bmatrix} = \frac{1}{2}\boldsymbol{Q}_0(q_0, \boldsymbol{q})\boldsymbol{\omega} \tag{1}$$

where $\boldsymbol{\omega}$ is the angular velocity of the spacecraft in the body fixed frame, $[q_0\ \boldsymbol{q}^T]^T$ the unit quaternion vector, $\boldsymbol{q}$ the vector components of unit quaternion, and

$$\boldsymbol{Q}_0(q_0, \boldsymbol{q}) = \begin{bmatrix} -\boldsymbol{q}^T \\ q_0\boldsymbol{E}_3 + \tilde{\boldsymbol{q}} \end{bmatrix}$$

with $\tilde{\boldsymbol{q}}$ as the cross product as

$$\tilde{\boldsymbol{q}} = \begin{bmatrix} 0 & -q_z & q_y \\ q_z & 0 & -q_x \\ -q_y & q_x & 0 \end{bmatrix}$$

and $\boldsymbol{E}_3$ the identity matrix.

By Lagrangian approach, the dynamical equations of the flexible spacecraft with the piezoelectric actuators were introduced in Ref. [1]. This paper considers the situation without the piezoelectric actuators. With $\boldsymbol{q}_{fi}$ as the modal coordinate vector, the dynamical equations of the spacecraft with flexible appendages are expressed as

$$\begin{cases} \boldsymbol{I}_{bt}\dot{\boldsymbol{\omega}} + \boldsymbol{H}\ddot{\boldsymbol{q}}_{fi} + \tilde{\boldsymbol{\omega}}\boldsymbol{I}_{bt}\boldsymbol{\omega} + \tilde{\boldsymbol{\omega}}\boldsymbol{H}\dot{\boldsymbol{q}}_{fi} = \boldsymbol{T}_b \\ \ddot{\boldsymbol{q}}_{fi} + \boldsymbol{C}_{fi}\dot{\boldsymbol{q}}_{fi} + \boldsymbol{K}_{fi}\boldsymbol{q}_{fi} = -\boldsymbol{H}^T\dot{\boldsymbol{\omega}} \end{cases} \tag{2}$$

where $\boldsymbol{I}_{bt}$ is the symmetric inertia matrix of the whole structure, $\boldsymbol{H}$ the coupling matrix between the rigid body and the second derivative of the modal coordinates, $\boldsymbol{T}_b$ the control torque, $\boldsymbol{C}_{fi}$ the damping matrices, and $\boldsymbol{K}_{fi}$ the stiffness matrices.