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Disturbance observer based finite-time attitude control for rigid spacecraft under input saturation

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ABSTRACT

A novel finite-time controller integrated with disturbance observer is investigated for a rigid spacecraft in the presence of disturbance, actuator saturation and misalignment. As a stepping-stone, a second-order disturbance observer is designed firstly such that the reconstruction of lumped disturbances is accomplished in finite time with zero error. Then, with the reconstructed information, a finite-time controller is synthesized even under actuator input saturation and misalignment, and the closed-loop system/state is proved to be finite-time stable and converges to the specified time-varying sliding mode surface. Moreover, the input saturation constraint is overcome via introducing an auxiliary variable to compensate for the overshooting. Numerical simulation results for the in-orbit rigid spacecraft show good performances, which validate the effectiveness and feasibility of the proposed schemes.
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1. Introduction

Accurate and reliable attitude stabilization is always one of the most important problems in spacecraft control system design. However, the unknown environment disturbances, spacecraft/actuator uncertainties and actuator output saturation, etc., further increase the complexity and difficulty in control scheme design. Recently, many improvements involving the design of attitude control laws have been extensively studied in literatures based on several inspiring approaches, such as optimal control [19,20], nonlinear feedback control [31], adaptive control [15], and robust control or their integrated applications [12,13]. Whereas, sliding mode control (SMC) has been widely studied and explored in practical applications for recent decades, due to its particular characters to deal with the external disturbances and model uncertainties [21,26]. Yeh [33] proposed two nonlinear attitude controllers, mainly consisting of the sliding mode controller and sliding mode adaptive controller, for a spacecraft with thrusters to follow the demanded trajectory in outer space. A robust adaptive fault tolerant control approach was also proposed for spacecraft attitude tracking in the presence of reaction wheels/actuators failures, external disturbances and time-varying inertia-parameter uncertainties [8]. However, the drawback of the predetermined sliding surface is that the robustness cannot be ensured in the reaching phase. To solve this problem, a time-varying sliding mode controller has been proposed for attitude tracking of a rigid spacecraft [14]. In Ref. [6], the authors presented a dual-stage control system design method for the flexible spacecraft attitude maneuver and vibration suppression, in which a switching mechanism was employed to design the attitude controller such that a variable structure control (VSC) law with a time-varying sliding mode surface was implemented outside the sliding region, and the VSC law with a linear sliding mode surface was activated inside the region. While, the chattering issue of SMC is still an open problem. Recently, several authors have introduced disturbance observer (DO) based SMC to alleviate the chattering problem and retain its nominal control performance [5,16,30]. In Ref. [32], a DO was proposed to reduce inherent chattering of SMC and improve the stability and robustness of the spacecraft platform. Also, the advantages of the sliding mode based DO (SMDO) are, handling the disturbances and uncertainties in a robust way simply with fast response, alleviating the chattering problem and retaining its nominal control performance.

However, one practical problem is that technically speaking, these researches involved in literatures above achieve attitude asymptotic stability of the closed-loop system, which implies that the system's trajectory converges to the equilibrium with infinite settling time, and that it is difficult to implement in practice. For this, finite-time control becomes an alternative way to obtain a faster convergence rate to the origin and achieve better robust disturbance attenuation. In Refs. [11,28], the authors utilized terminal sliding mode control (TSMC) approach to achieve finite time

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attitude convergence for spacecraft system. Then a modified robust controller with finite time convergence for the rigid spacecraft attitude tracking was discussed in Ref. [17], in order to solve the singularity phenomenon introduced by the common TSMC method. Ref. [18] proposed a class of SMC for the attitude tracking system applying DO to compensate for the uncertainties, which drove the states of the closed-loop system onto the sliding surface in finite time. In addition, the finite-time stabilization/convergence of the spacecraft attitude in the presence of disturbances has also been investigated/achieved in some other works [3,4,10,29].

While, it should be stressed that the above results have been derived from the implicit assumption that the actuators are able to provide any requested joint torques, and also the torques' axis directions and/or input scaling of the actuators (such as gas jets, reaction fly-wheels) are exactly known. This assumption is rarely satisfied in practical engineering environment because of the existing limitation of actuator output signal and possible misalignment of the actuator during installation. To overcome these difficulties, several solutions that take the actuator constraints into account have been extensively studied by Hu [7,9], Bošković [1,2], Zhu et al. [34], in which the saturation function or standard hypertangent function is commonly applied. However, an auxiliary variable is introduced to compensate for the overshooting of the actuator output by a second-order sliding mode observer in this paper.

In this work, an attempt is made to provide a simple and robust attitude control strategy for spacecraft with finite time convergence in the presence of external disturbances, inertial uncertainties, actuator misalignments and even actuator saturation. The main contributions of this paper are that: 1) a second-order observer is presented to estimate the sliding mode surface and the differentiable specified total disturbance, and zero errors of estimation can be achieved in finite time, 2) a novel time-varying SMC scheme is proposed for the spacecraft attitude stabilization in the presence of uncertainties mentioned above, 3) the input saturation rejection is explicitly considered and achieved via introducing an auxiliary variable to compensate for the overshooting. While, the specified total disturbances and input overshooting can be estimated effectively and accurately using the proposed DO, and the time-varying sliding mode technique based controller can provide fast and accurate response in view of this effective real-time compensation in this paper. A key feature of the proposed controller ensures the convergence of both attitude and velocity in finite time with simple design procedures under input saturation, which is of great interest for aerospace industry for real-time implementation.

The paper is organized as follows: Next section states spacecraft modeling and control problem formulation. Attitude control laws are derived in Section 3. Next, numerical simulation results are presented to demonstrate various features of the proposed control schemes. Finally, the paper is completed with some concluding comments.

2. Spacecraft modeling and problem formulation

2.1. Spacecraft attitude dynamics

Consider a rigid spacecraft system described by the following attitude kinematics and dynamics equations [24]

$$\begin{bmatrix} \dot{q}_0 \\ \dot{q} \end{bmatrix} = \frac{1}{2} \begin{bmatrix} -q^{\mathrm{T}} \\ q_0 I + q^{\times} \end{bmatrix} \omega \tag{1}$$

$$J\dot{\omega} = -\omega^{\times} J\omega + u(t) + d(t) \tag{2}$$

where q_0 and $q = [q_1 \ q_2 \ q_3]^{\mathrm{T}} \in R^3$ are the scalar and vector components of the unit attitude quaternion, respectively, satisfying the constraint $q^{\mathrm{T}} q + q_0^2 = 1$; $\omega \in R^3$ is the angular velocity vector of a body-fixed reference frame of spacecraft with respect to the inertial reference frame expressed in the body-fixed reference frame;

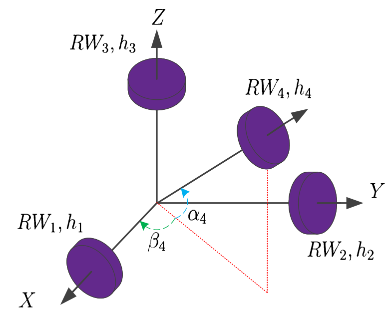

Fig. 1. Configuration of reaction wheels.

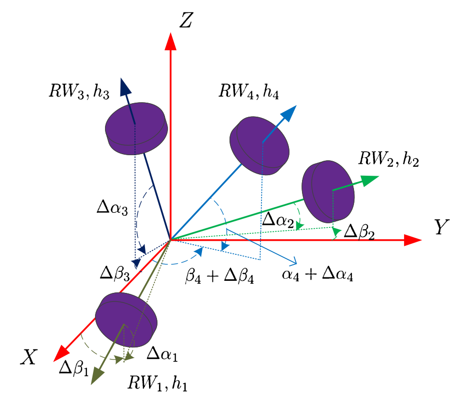

Fig. 2. Configuration with misalignments.

I is an identity matrix with proper dimensions; $J \in R^{3\times3}$ is the total inertia matrix of the spacecraft; $u(t) = [u_1 \ u_2 \ u_3]^{\mathrm{T}} \in R^3$ denotes the combined control torque produced by the actuators; and $d(t) = [d_1 \ d_2 \ d_3]^{\mathrm{T}} \in R^3$ denotes the external disturbance torque from the environment, which is assumed to be unknown but bounded. Additionally, $q^{\times}$ (or $\omega^{\times}$) denotes a skew-symmetric matrix given by

$$q^{\times} = \begin{bmatrix} 0 & -q_3 & q_2 \\ q_3 & 0 & -q_1 \\ -q_2 & q_1 & 0 \end{bmatrix} \tag{3}$$

2.2. Actuator configuration with misalignment

For orbiting spacecraft, loosely speaking, they have more than three actuators aligned with the spacecraft body axes. A common configuration with four reaction wheels is shown in Fig. 1, in which three reaction wheels' (such as reaction wheels 1, 2, and 3) rotation axes are orthogonal to the spacecraft ontology shaft and the fourth one is installed with the equiangular direction with the ontology three axis, noted as RW_1, RW_2, RW_3 and RW_4. In this circumstance, employing the configuration of four reaction wheels, as shown in Fig. 1, the spacecraft dynamics in Eq. (2) can be found as

$$J\dot{\omega} = -\omega^{\times} J\omega + D\tau(t) + d(t) \tag{4}$$

where D is configuration matrix of the reaction wheel, representing the influence of each wheel on the angular acceleration of the spacecraft, and $\tau = [\tau_1 \ \tau_2 \ \tau_3 \ \tau_4]^{\mathrm{T}}$ denotes the torques produced by the four reaction wheels. Note that the configuration matrix D is available for a given spacecraft.

However, in practice, the knowledge of orthogonal configuration of actuator will never be perfect. Whether due to finite manufacturing tolerances or warping of the spacecraft structure during launching, some misalignment can always exist. Referring to Fig. 2 and to model such uncertainties, it is assumed that the reaction wheel mounted on X axis is tilted over nominal direction with

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