Aerospace Science and Technology 55 (2016) 144–157

Contents lists available at ScienceDirect

Aerospace Science and Technology

www.elsevier.com/locate/aescte

Smooth finite-time fault-tolerant attitude tracking control for rigid spacecraft

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ARTICLE INFO

Article history:
Received 20 October 2015
Received in revised form 24 February 2016
Accepted 19 May 2016
Available online 27 May 2016

Keywords:
Finite time control
Actuator fault
Unwinding phenomenon
Attitude tracking
Spacecraft

ABSTRACT

This paper investigates the problem of attitude tracking control for a rigid spacecraft subject to parametric uncertainties, external disturbances, actuator faults and actuator saturation constraints. By combining the finite-time passivity technique into adaptive sliding mode control approach, a novel smooth fault-tolerant control algorithm with finite time convergence is proposed. Then, the finite-time convergence of the relative attitude errors will be achieved by implementing the proposed smooth fault-tolerant controller, even under actuator faults and magnitude constraints. In particular, a solution to the unwinding phenomenon, arising from the usage of the redundant four-parameter attitude representation, is explored in the sense of finite-time passivity. Besides detailed controller design procedures and rigorous theoretical proofs of all related closed-loop finite-time stability, numerical simulation results are exhibited to demonstrate the effectiveness and superior control performance of the proposed control scheme.

1. Introduction

Attitude tracking control problem of rigid spacecraft has attracted an increasing attention due to its extensive applications in aerospace engineering, such as satellite surveillance, formation flying, on-orbit service, etc. However, it is a challenge to achieve rapid and high-accuracy attitude tracking, this is mainly due to the nonlinear and highly coupled attitude dynamics, and the existence of parametric uncertainties and external disturbances. Recently, various advanced nonlinear control schemes, including sliding mode control [1–3], inverse optimal control [4], attracting-manifold-based adaptive control [5], output feedback control [6,7], etc., have been carried out to improve the closed-loop performance. It is worth mentioning that, all of the above control schemes are derived under the Lipschitz continuity condition such that the attitude tracking errors converge to the equilibrium point with infinite-settling time. In Refs. [8–10], the authors have pointed out that the finite-time control has faster convergence rate, higher steady-state accuracy and better disturbance rejection properties. Obviously, in view of the superior merits of the finite-time control technique, the attitude control with finite-time convergence is highly preferable and can lead to a better control performance. In Ref. [11], by using the technique of adding a power integrator, a continuous finite-time attitude tracking control scheme was proposed. Finite-time output feedback control was investigated in Ref. [12] on the basis of the geometric homogeneity theory and the same technique used in Ref. [11]. By using the geometric homogeneity approach, a second order sliding mode controller with finite-time convergence was proposed in Ref. [13]. Based on the novel fast nonsingular terminal sliding mode and the adaptive control technique, the finite-time convergence of the relative errors was also obtained in Ref. [14] for the problem of attitude tracking of spacecraft. In Ref. [15], Song and Li proposed a finite-time attitude control scheme with double closed loops structure.

Note that the system faults, especially, actuator faults, were not considered in the above listed attitude control schemes. In practice, kinds of uncontrollable actuator faults are frequently encountered due to the harsh working environment, and the occurrence of actuator faults might result in a series of potential problems (e.g., safety and reliability). Extensive fault-tolerant control schemes have been carried out for the spacecraft attitude control, and actuator failure compensation by using the adaptive control technique for attitude tracking was addressed in Ref. [16]. In Refs. [17,18], robust fault-tolerant attitude control schemes were also carried out for flexible spacecraft such that the asymptotical stability of the closed-loop system was obtained under the actuator faults and disturbances. A nonregressor-based attitude tracking control scheme with fault-tolerant capability was further presented in Ref. [19] to deal with thruster faults and thruster saturation limits, moreover parametric uncertainties and external disturbances also were simultaneously taken into consideration. However, these

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http://dx.doi.org/10.1016/j.ast.2016.05.019
1270-9638/

fault-tolerant control schemes could merely achieve asymptotical stability of the closed-loop system. By employing the integral-type sliding mode manifold and the adaptive control technique, a fault-tolerant controller for attitude stabilization was designed in Ref. [20], and the practical finite-time stability of the attitude trajectory was guaranteed. A sliding mode fault-tolerant attitude tracking control scheme was proposed in Ref. [21], in which the convergence time as an explicitly specified parameter can be chosen by designers. Finite-time fault-tolerant controllers were also presented in Refs. [22,23] by incorporating the adaptive technique into the fast terminal sliding mode control, and the actuator saturations were considered simultaneously.

Most of the aforementioned investigations employed unit quaternions for attitude control to avoid the inherent geometric singularity. Unfortunately, the use of redundant four-parameter representation inevitably renders two antipodal equilibrium points for a given physical equilibrium orientation, and thus may result in an undesired or unnecessary full rotation. In Ref. [24], rotation matrix as a global and unique attitude representation was employed to deal with this unwinding problem and a continuous attitude control law was presented. In Ref. [25], the attitude of the spacecraft can be regulated to the desired physical orientation through a short rotation path with a suitable choice of the error variables. In addition, the unwinding phenomenon was also avoided by using a hybrid feedback control structure in Ref. [26].

The main contribution of this paper is the design of a novel smooth finite-time fault-tolerant control algorithm, which explicitly takes into account parametric uncertainties and external disturbances, assures fast and accurate response, achieves effective compensation for the effect of actuator faults and actuator saturation constraints, and has the following properties: 1) the proposed controller is fundamentally smooth, hence better performance of the control input can be guaranteed; 2) finite-time convergence of the relative attitude errors can be obtained under the designed controller; 3) the spacecraft attitude can be regulated to the closest equilibrium point in terms of rotation path, which leads to a smaller angle rotation and less energy consumption. In addition, the control scheme proposed in this paper achieves better performance (faster response, higher precision performance, etc.) with comparison to the other existing strategies. A pivotal feature of the proposed controller is that it can ensure the finite-time convergences of the attitude tracking errors and the relative angular velocities with simple implementation of the proposed controller and less online computations, such that it is indeed user/designer friendly, which is of great interest for aerospace industry. The superiorities of the presented control scheme are analytically authenticated and also illustrated via simulation results.

The paper is organized as follows. Section 2 states the attitude mathematical model of a rigid spacecraft, preliminaries and control problems formulation. The next section presents the novel attitude control law formulations and the proofs of all related finite-time stability. Simulation results are given in Section 4 to verify the superior features of the proposed scheme. Finally, the paper is completed with some concluding comments.

2. Mathematical model and control problem

2.1. Spacecraft attitude dynamics

The rigid spacecraft is driven by actuators which can provide torques along three mutually perpendicular axes that define a body-fixed frame. The kinematics and dynamics equations of the rigid spacecraft in terms of unit quaternion are governed by ([29], Chapter 4):

$$\dot{q}_v = \frac{1}{2}\left(q_v^{\times} + q_4 I_3\right)\omega \tag{1}$$

$$\dot{q}_4 = -\frac{1}{2}q_v^T\omega \tag{2}$$

$$J\dot{\omega} = -\omega^{\times}J\omega + u + T_d \tag{3}$$

where $q = [q_1, q_2, q_3, q_4]^T = [q_v^T, q_4]^T \in \mathbb{R}^3 \times \mathbb{R}$ denotes the unit quaternion describing the spacecraft attitude in the body frame with respect to an inertia frame and satisfies the constraint $q_v^T q_v + q_4^2 = 1$; $I_3 \in \mathbb{R}^{3\times 3}$ is the identity matrix; ω is the angular velocity of the spacecraft with respect to the inertial frame and expressed in the body frame; $J \in \mathbb{R}^{3\times 3}$ is the inertia matrix that defines the mass moment of inertia of the spacecraft; $u \in \mathbb{R}^3$ and $T_d \in \mathbb{R}^3$ denote the control torque and the external disturbance, respectively; the notation $a^{\times} \in \mathbb{R}^{3\times 3}$ represents a skew-symmetric matrix and can be characterized by the vector cross-product operation between any two vectors, that is to say, $a^{\times}b = a \times b$ for any $a, b \in \mathbb{R}^3$.

2.2. Relative attitude error dynamics

In order to address the attitude tracking control issue, like in Refs. [4,19,28], the error quaternion $q_e = [q_{ev}^T, q_{e4}]^T \in \mathbb{R}^3 \times \mathbb{R}$ is defined as the relative attitude error between the body-fixed frame and a desired frame with attitude orientation $q_d = [q_{dv}^T, q_{d4}]^T \in \mathbb{R}^3 \times \mathbb{R}$, whose attitude motion is governed by $\dot{q}_{dv} = \frac{1}{2}(q_{dv}^{\times} + q_{d4}I_3)\omega_d$ and $\dot{q}_{d4} = -\frac{1}{2}q_{dv}^T\omega_d$, and ω_d is the desired angular velocity, then one can have

$$q_e = q \otimes q_d^* = \begin{bmatrix} q_{d4}q_v - q_4 q_{dv} + q_v^{\times} q_{dv} \\ q_{d4}q_4 + q_{dv}^T q_v \end{bmatrix} \tag{4}$$

where q_d^* is the conjugate quaternion of the unit quaternion q_d; q_d and q_e satisfy $q_{dv}^T q_{dv} + q_{d4}^2 = 1$ and $q_{ev}^T q_{ev} + q_{e4}^2 = 1$, respectively. The corresponding rotation matrix can be obtained from the attitude error quaternion vector through the following equations

$$C = \left(q_{e4}^2 - q_{ev}^T q_{ev}\right)I_3 + 2q_{ev}q_{ev}^T - 2q_{e4}q_{ev}^{\times} \tag{5}$$

where the rotation matrix C satisfies $\|C\| = 1$ and $\dot{C} = -\omega_e^{\times}C$.

Define the relative angular velocity vector $\omega_e \in \mathbb{R}^3$ between the body-fixed frame and the desired frame as

$$\omega_e = \omega - C\omega_d \tag{6}$$

Hence, the relative attitude error dynamics can be given as

$$\dot{q}_{ev} = \frac{1}{2}\left(q_{ev}^{\times} + q_{e4}I_3\right)\omega_e \tag{7}$$

$$\dot{q}_{e4} = -\frac{1}{2}q_{ev}^T\omega_e \tag{8}$$

$$J\dot{\omega}_e = -\omega^{\times}J\omega + J\left(\omega_e^{\times}C\omega_d - C\dot{\omega}_d\right) + u + T_d \tag{9}$$

In what follows, attitude tracking control law will be developed for the nonlinear system given by Eqs. (7)–(9). To proceed the controller design and stability analysis, the following property and assumptions are given.

Property 1. *The inertial matrix can be lower and upper bounded as*

$$\lambda_{\underline{J}}\|x\|^2 \le x^T J x \le \lambda_{\bar{J}}\|x\|^2, \quad \forall x \in \mathbb{R}^3 \tag{10}$$

where $\lambda_{\underline{J}}$ and $\lambda_{\bar{J}}$ are some positive scalars.

Assumption 1. During the orbiting mission, the inertia matrix is time-varying and uncertain, but remains positive definite and bounded all the time. Therefore, it is rational to assume that there exist some unknown constants such that $\varepsilon_{\underline{J}} \le \|J\| \le \varepsilon_{\bar{J}} < \infty$ and $\|\dot{J}\| \le \varepsilon_f < \infty$, where $\varepsilon_{\underline{J}} > 0$, $\varepsilon_{\bar{J}} > 0$ and $\varepsilon_f \ge 0$.