Aerospace Science and Technology ••• (••••) •••–•••

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Aerospace Science and Technology

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Bounded finite-time attitude tracking control for rigid spacecraft via output feedback

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ARTICLE INFO

Article history:
Received 21 January 2016
Received in revised form 29 November 2016
Accepted 20 January 2017
Available online xxxx

Keywords:
Rigid spacecraft
Finite-time observer
Finite-time control
Attitude tracking
Actuator saturation

ABSTRACT

This paper investigates a velocity-free finite-time attitude control scheme for a rigid spacecraft with actuator saturation and external disturbances. Initially, a finite-time observer is designed to compensate the unknown angular velocity information. With the estimated values, a finite-time controller is further developed under which the time-varying reference attitude trajectory can be tracked precisely. Moreover, input saturation problem is solved by adaptive method. Rigorous Lyapunov-based analysis shows that the states of the closed-loop system converge to a small neighborhood of the origin in finite time. Finally, the dynamic performance of attitude control system is presented by numerical simulation examples and the superiority of the finite-time control scheme is verified.

1. Introduction

Attitude control is a fundamental issue for spacecraft maneuvering, which has drawn tremendous attention during the last decades. Due to its inherent nonlinear dynamic characteristics and the environmental factors, varieties of modern control approaches have been applied to improve the performance of attitude control systems [1–4]. However, most of the existing algorithms require full-state measurements, which is not easy to meet in practical engineering. Thus output feedback control and observer-based control scheme are more preferred. In [5], adaptive attitude tracking control without angular velocity measurements is discussed for a rigid spacecraft. In [6], a velocity filter is constructed and adaptive method is employed for dealing with the mismatch between the actual and estimated parameters. By using an approximate-differentiation filter, the controller for spacecraft relative rotation is designed only relay on the attitude measurements in [7], where practical asymptotic stability is proved for the closed-loop system. Based on adaptive output feedback, the attitude control laws are proposed for a rigid spacecraft with consideration of model uncertainties and external disturbances in [8].

Practically speaking, actuation limit does exist due to the physical and mechanical factors. The ignorance of such phenomenon would lead to unexpected result, which might even threaten system stability. A considerable amount of work has been dedicated to solve actuator saturation [9–12]. In [10], the anti-windup scheme is adopted to handle input constraints for attitude control system. In [11], input saturation is explicitly addressed in the presence of parametric uncertainties and external disturbances. Furthermore, control algorithm for the rigid spacecraft with consideration of fault-tolerant and input saturation is discussed in [12]. Meanwhile, effort has been put on studying control constraint problem in the absence of velocity measurements. A bounded controller combined with a filter is constructed for the fully actuated Euler–Lagrange systems without velocities information in [13], where hyperbolic tangent function is used to solve input saturation. Similar method is employed on double-integrator dynamics in [14], and the upper bound of the proposed control law is independent of the number of individual's neighbors.

Nevertheless, most of the literatures only ensure asymptotical stability of the closed-loop systems, which implies exact convergence is not guaranteed in finite time. Control strategies with rapid response and strong robustness are more desirable for guaranteeing excellent system performance. Under such a circumstance, finite-time control theory has been a favorable and hit topic ever since it was put forward. Attempts have been made in [15–17], where homogeneous theory has become a common tool for deriving finite-time control algorithms. Terminal sliding mode control is an alternative method to achieve finite-time stability. In [18], a terminal sliding mode (TSM) is proposed for single-input single-

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http://dx.doi.org/10.1016/j.ast.2017.01.020
1270-9638/

output plant. However, the singularity problem limits the application of TSM. As a result, subsequent extensions have been carried out to overcome this weakness [19,20]. In the area of spacecraft control, finite-time control strategies have been extensively studied and widely applied [21–24]. In [21], a global finite-time controller is derived for spacecraft attitude stabilization by homogeneous method. Terminal sliding mode control is adopted on spacecraft formation flying [22]. In [23], a robust sliding mode controller is designed, thus the spacecraft can track its desired attitude in finite time. To compensate the angular velocity, a dynamic finite-time observer is employed and a finite-time control law is designed for the rigid spacecraft [24].

It is worthwhile to mention that to guarantee fast convergence rate, the control output always reaches large magnitude, especially within the initial phase of system response. Thus the researchers have put effort on investigating the saturated control algorithms with finite-time convergence rate. In [25], a bounded distributed finite-time controller is developed based on double-integrator dynamics under a detailed–balanced topology. In [26], homogeneous theory is adopted to derive a proportional-derivative-type finite-time attitude stabilization law with a saturated function to ensure bounded input. A finite-time distributed attitude control law with actuator constraint is proposed for spacecraft formation by employing fast terminal sliding mode and Chebyshev neural network in [27]. However, full-state feedback scheme is used in [26] and [27]. To relax the requirement of angular velocity, a finite-time attitude coordinated controller combined with a filter is designed by using homogeneous theory in [28]. Although the input saturation is addressed explicitly, only static desired attitude has been considered. It is not straightforward to extend the control algorithm in [28] to time-varying reference attitude trajectory case. So far, constructing bounded finite-time control schemes via output feedback remains to be an open problem, especially for the attitude tracking problem.

Inspired by the realities stated above, this paper aimed at designing a velocity-free finite-time attitude tracking control law with consideration of actuator saturation and external disturbances. Initially, we propose a finite-time observer by which the lack of angular velocity measurements is compensated. Then, adding a power integrator technique is adopted to design the finite-time control law. Additionally, adaptive method is used to solve input saturation problem. The remainder of this paper is organized as follows: In Section 2, the mathematical model of the rigid spacecraft and several useful definitions and lemmas are introduced. The control algorithm is addressed with rigorous proof in Section 3. Section 4 demonstrates the efficiency of the proposed control law by presenting numerical simulation results. Conclusions and remarks are made in Section 5.

2. Dynamics and mathematical preliminaries

2.1. Spacecraft attitude dynamics and kinematics

The attitude kinematics and dynamics of a rigid spacecraft is modeled as [29]

$$\dot{q}_0 = -\frac{1}{2}\boldsymbol{q}^{\mathrm{T}}\boldsymbol{\omega} \tag{1}$$

$$\dot{\boldsymbol{q}} = \boldsymbol{Q}(\boldsymbol{q})\boldsymbol{\omega} \tag{2}$$

$$\boldsymbol{J}\dot{\boldsymbol{\omega}} + \boldsymbol{\omega}^{\times}\boldsymbol{J}\boldsymbol{\omega} = \boldsymbol{u} + \boldsymbol{d} \tag{3}$$

where $\bar{\boldsymbol{q}} = [q_0, \boldsymbol{q}^{\mathrm{T}}]^{\mathrm{T}} \in \Re^4$ is the quaternion denoting the rotation from the body frame to the inertial frame, with q_0 denotes the scalar part and $\boldsymbol{q} \in \Re^3$ the vector part satisfying $q_0^2 + \boldsymbol{q}^{\mathrm{T}}\boldsymbol{q} = 1$. $\boldsymbol{\omega} \in \Re^3$ denotes the angular velocity of the spacecraft with respect to the inertial frame expressed in the body frame. The matrix $\boldsymbol{Q}(\boldsymbol{q}) \in \Re^{3\times3}$ is defined as $\boldsymbol{Q}(\boldsymbol{q}) = \frac{1}{2}(q_0\boldsymbol{I}_{3\times3} + \boldsymbol{q}^{\times})$. $\boldsymbol{J} \in \Re^{3\times3}$ is the inertia matrix, $\boldsymbol{u} \in \Re^3$ and $\boldsymbol{d} \in \Re^3$ denote the control torque and external disturbances acting on the spacecraft, respectively. Here, the operator $\boldsymbol{a}^{\times}$ represents a skew–symmetric matrix acting on the vector $\boldsymbol{a} = \begin{bmatrix} a_1 & a_2 & a_3 \end{bmatrix}^{\mathrm{T}}$ and has the following form

$$\boldsymbol{a}^{\times} = \begin{bmatrix} 0 & -a_3 & a_2 \\ a_3 & 0 & -a_1 \\ -a_2 & a_1 & 0 \end{bmatrix} \tag{4}$$

To address attitude tracking problem, a time-varying reference attitude trajectory is described by unit quaternion $\bar{\boldsymbol{q}}_d = [q_{d0}, \boldsymbol{q}_d^{\mathrm{T}}]^{\mathrm{T}} \in \Re^4$ and desired angular velocity $\boldsymbol{\omega}_d \in \Re^3$. The attitude error $\bar{\boldsymbol{q}}_e \in \Re^4$ and angular velocity error $\boldsymbol{\omega}_e \in \Re^3$ are introduced, which are defined as $\bar{\boldsymbol{q}}_e = \bar{\boldsymbol{q}}_d^{-1} \odot \bar{\boldsymbol{q}}$ and $\boldsymbol{\omega}_e = \boldsymbol{\omega} - \boldsymbol{R}_e\boldsymbol{\omega}_d$. In addition, $\odot$ represents the quaternion multiplication and $\boldsymbol{R}_e = (q_{e0}^2 - \boldsymbol{q}_e^{\mathrm{T}}\boldsymbol{q}_e)\boldsymbol{I}_{3\times3} + 2\boldsymbol{q}_e\boldsymbol{q}_e^{\mathrm{T}} - 2q_{e0}\boldsymbol{q}_e^{\times}$ represents the rotation matrix. As for $\boldsymbol{R}_e$, it has that $\|\boldsymbol{R}_e\| = 1$, $\dot{\boldsymbol{R}}_e = -\boldsymbol{\omega}_e^{\times}\boldsymbol{R}_e$. Correspondingly, we obtain that

$$\dot{q}_{e0} = -\frac{1}{2}\boldsymbol{q}_e^{\mathrm{T}}\boldsymbol{\omega}_e \tag{5}$$

$$\dot{\boldsymbol{q}}_e = \boldsymbol{Q}(\boldsymbol{q}_e)\boldsymbol{\omega}_e \tag{6}$$

$$\begin{aligned} \boldsymbol{J}\dot{\boldsymbol{\omega}}_e &= \boldsymbol{u} + \boldsymbol{d} - (\boldsymbol{\omega}_e + \boldsymbol{R}_e\boldsymbol{\omega}_d)^{\times}\boldsymbol{J}(\boldsymbol{\omega}_e + \boldsymbol{R}_e\boldsymbol{\omega}_d) \\ &\quad - \boldsymbol{J}\left(\boldsymbol{\omega}_e^{\times}\boldsymbol{R}_e\boldsymbol{\omega}_d - \boldsymbol{R}_e\dot{\boldsymbol{\omega}}_d\right) \end{aligned} \tag{7}$$

where $\boldsymbol{Q}(\boldsymbol{q}_e) = \frac{1}{2}(q_{e0}\boldsymbol{I}_{3\times3} + \boldsymbol{q}_e^{\times})$ and $\|\boldsymbol{Q}(\boldsymbol{q}_e)\| = \frac{1}{2}$ holds. Let $\boldsymbol{v}_1 = \boldsymbol{R}_e\boldsymbol{\omega}_d$, $\boldsymbol{v}_2 = \boldsymbol{R}_e\dot{\boldsymbol{\omega}}_d$, $\boldsymbol{F}(\boldsymbol{q}_e) = \boldsymbol{Q}(\boldsymbol{q}_e)^{-1}$, and $\boldsymbol{g}(\boldsymbol{q}_e) = \boldsymbol{Q}(\boldsymbol{q}_e)\boldsymbol{J}^{-1}$, then Eqs. (1)–(3) can be rewritten as

$$\dot{\boldsymbol{q}}_e = \boldsymbol{v}_e \tag{8a}$$

$$\dot{\boldsymbol{v}}_e = \boldsymbol{f}(\boldsymbol{q}_e, \boldsymbol{v}_e) + g(\boldsymbol{q}_e)\boldsymbol{u} + \boldsymbol{d}^* \tag{8b}$$

where $\boldsymbol{v}_e$ is introduced as the first derivative of the vector part of attitude error quaternion, $\boldsymbol{f}(\boldsymbol{q}_e, \boldsymbol{v}_e) = \dot{\boldsymbol{Q}}_e\boldsymbol{F}_e\boldsymbol{v}_e - \boldsymbol{Q}_e\boldsymbol{J}^{-1}\{(\boldsymbol{F}_e\boldsymbol{v}_e + \boldsymbol{v}_1)^{\times}\boldsymbol{J}(\boldsymbol{F}_e\boldsymbol{v}_e + \boldsymbol{v}_1) - \boldsymbol{J}[(\boldsymbol{F}_e\boldsymbol{v}_e)^{\times}\boldsymbol{v}_1 - \boldsymbol{v}_2]\}$ and $\boldsymbol{d}^* = g(\boldsymbol{q}_e)\boldsymbol{d}$. In the following, $\boldsymbol{F}(\boldsymbol{q}_e)$ and $\boldsymbol{Q}(\boldsymbol{q}_e)$ are written into $\boldsymbol{F}_e$ and $\boldsymbol{Q}_e$ for the sake of brevity. To facilitate the control algorithm design, the following assumptions are made.

Assumption 1. The inertia tensor of the spacecraft is a symmetric positive definite matrix, which is precisely known. It is reasonable to suppose that $\|\boldsymbol{J}\| \le \bar{J}$ and $\|\boldsymbol{J}^{-1}\| \le \mu_J$ with $\bar{J}$ and μ_J being positive scalars.

Assumption 2. The external disturbance $\boldsymbol{d}$ is assumed to be bounded by a known positive scalar $\bar{d}$, i.e. $\|\boldsymbol{d}\| \le \bar{d}$. Noticing that $\|\boldsymbol{g}(\boldsymbol{q}_e)\| = \|\boldsymbol{Q}_e\boldsymbol{J}^{-1}\| \le \frac{1}{2}\mu_J$, which leads to $\|\boldsymbol{g}(\boldsymbol{q}_e)\boldsymbol{d}\| \le \frac{1}{2}\mu_J\bar{d}$. Thus, it has that $\|\boldsymbol{d}^*\| \le \varphi_d$ with $\varphi_d = \frac{1}{2}\mu_J\bar{d}$ being a positive scalar.

Remark 1. To ensure the existence of $\boldsymbol{F}_e$, the matrix $\boldsymbol{Q}_e$ should be invertible, which implies that $\det(\boldsymbol{Q}_e(\boldsymbol{q}_e)) = \frac{1}{2}q_{e0}(t) \ne 0$ for $t \ge 0$. Thus, the initial state and control strategy should be chosen to guarantee $q_{e0}(t) \ne 0$ for all time. In fact, the initial attitudes can be always selected as $q_{e0}(t) \ne 0$ and the control parameters are tunable to ensure $q_{e0}(t) \ne 0$ for $t > 0$. So this remark is a mild and reasonable restriction.

2.2. Definitions and lemmas

Definition 1. (See [30].) Consider the following system

$$\dot{\boldsymbol{x}} = \boldsymbol{f}(\boldsymbol{x}), \quad \boldsymbol{f}(\boldsymbol{0}) = \boldsymbol{0}, \quad \boldsymbol{x} \in R^n \tag{9}$$