Aerospace Science and Technology 39 (2014) 13–21

Contents lists available at ScienceDirect

Aerospace Science and Technology

www.elsevier.com/locate/aescte

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

Qinglei Hu [a,b,*], Bo Li [b], Juntong Qi [c]

[a] School of Automation Science and Electrical Engineering, Beihang University, Beijing, 100191, China
[b] Department of Control Science and Engineering, Harbin Institute of Technology, Harbin, 150001, China
[c] State Key Laboratory of Robotics, Shenyang Institute of Automation, CAS, Shenyang, 110016, China

ARTICLE INFO

Article history:
Received 5 August 2013
Received in revised form 29 July 2014
Accepted 26 August 2014
Available online 1 September 2014

Keywords:
Spacecraft
Finite-time control
Disturbance observer
Input saturation
Misalignment

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.

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 at-

* Corresponding author at: School of Automation Science and Electrical Engineering, Beihang University, Beijing, 100191, China.

http://dx.doi.org/10.1016/j.ast.2014.08.009
1270-9638/

titude 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;

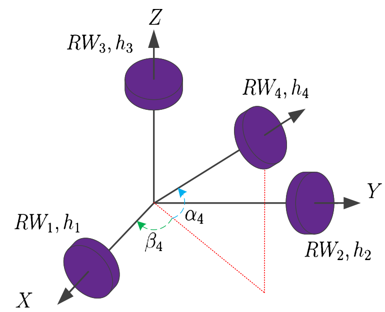

Fig. 1. Configuration of reaction wheels.

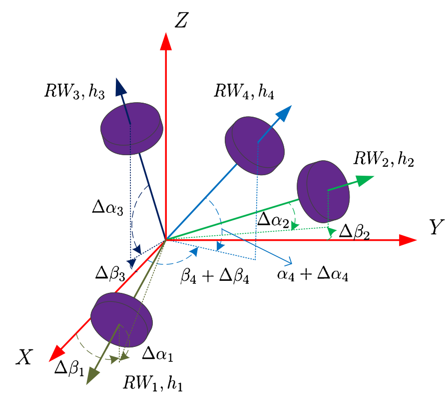

Fig. 2. Configuration with misalignments.

I is an identity matrix with proper dimensions; $J \in R^{3\times 3}$ 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 constant angles, $\Delta\alpha_1$ and $\Delta\beta_1$; also for other reaction wheels mounted left are assumed to be tilted over nominal direction with $\Delta\alpha_2$, $\Delta\beta_2$, $\Delta\alpha_3$, $\Delta\beta_3$, $\Delta\alpha_4$ and $\Delta\beta_4$ respectively. To this end, the real reaction wheel torques with misalignment can be expressed as

$$u = \tau_1 \begin{bmatrix} \cos\Delta\alpha_1 \\ \sin\Delta\alpha_1\cos\Delta\beta_1 \\ \sin\Delta\alpha_1\sin\Delta\beta_1 \end{bmatrix} + \tau_2 \begin{bmatrix} \sin\Delta\alpha_2\cos\Delta\beta_2 \\ \cos\Delta\alpha_2 \\ \sin\Delta\alpha_2\sin\Delta\beta_2 \end{bmatrix} + \tau_3 \begin{bmatrix} \sin\Delta\alpha_3\cos\Delta\beta_3 \\ \sin\Delta\alpha_3\sin\Delta\beta_3 \\ \cos\Delta\alpha_3 \end{bmatrix} + \tau_4 \begin{bmatrix} \cos(\alpha_4+\Delta\alpha_4)\cos(\beta_4+\Delta\beta_4) \\ \cos(\alpha_4+\Delta\alpha_4)\sin(\beta_4+\Delta\beta_4) \\ \sin(\alpha_4+\Delta\alpha_4) \end{bmatrix} \tag{5}$$

Generally, the misalignment angle errors $(\Delta\alpha_i, \Delta\beta_i)$ are very small in practice, and the following relationships are adopted to approximate Eq. (5)

$$\cos\Delta\alpha_i \approx \cos\Delta\beta_i \approx 1, \qquad \sin\Delta\alpha_i \approx \Delta\alpha_i, \\ \sin\Delta\beta_i \approx \Delta\beta_i \tag{6}$$

Then for the considered actuator configuration, the configuration matrix D can be represented as

$$D = D_0 + \Delta D \tag{7}$$

with

$$D_0 = \begin{bmatrix} 1 & 0 & 0 & \cos\alpha_4\cos\beta_4 \\ 0 & 1 & 0 & \cos\alpha_4\sin\beta_4 \\ 0 & 0 & 1 & \sin\alpha_4 \end{bmatrix} \tag{8a}$$

$$\Delta D = \begin{bmatrix} 0 & \Delta\alpha_2\cos\Delta\beta_2 & \Delta\alpha_3\cos\Delta\beta_3 & -\Delta\alpha_4\sin\alpha_4\cos\beta_4 - \Delta\beta_4\cos\alpha_4\sin\beta_4 \\ \Delta\alpha_1\cos\Delta\beta_1 & 0 & \Delta\alpha_3\sin\Delta\beta_3 & -\Delta\alpha_4\sin\alpha_4\sin\beta_4 + \Delta\beta_4\cos\alpha_4\cos\beta_4 \\ \Delta\alpha_1\sin\Delta\beta_1 & \Delta\alpha_2\sin\Delta\beta_2 & 0 & \Delta\alpha_4\cos\alpha_4 \end{bmatrix} \tag{8b}$$

where D_0 denotes the nominal value, and ΔD denotes the uncertainty of configuration matrix. Accordingly, the spacecraft dynamics under this uncertainty can be established as

$$J\dot{\omega} = -\omega^{\times} J\omega + (D_0 + \Delta D)\tau(t) + d(t) \tag{9}$$

For synthesis of control system design, the following reasonable assumptions are made:

Assumption 1. The external disturbances $d(t)$ including gravitational perturbations, aerodynamic torque, radiation torque, and other environmental or non-environmental torques, are assumed to be bounded. Thus, it is reasonable to assume that there always exists a constant d_0 such that

$$\|d(t)\| \le d_0 \tag{10}$$

Assumption 2. The total inertia matrix of the spacecraft $J \in R^{3\times3}$ is in the form of $J = J_0 + \Delta J$, where J_0 is a defined symmetrical matrix; ΔJ denotes the uncertainties of the spacecraft inertia, which is caused by the external environment disturbances or self-energy/fuel consuming or components/payloads releasing, also are assumed to be differentiable.

Assumption 3. Due to physical limitations on actuators, the actuator control actions are limited by certain values. For simplicity, we assume that outputs of actuator torques have the same constraint values $(\underline{\tau}, \bar{\tau})$, *i.e.*

$$\tau(t) \in \Omega := \{\tau \in R^m \mid \underline{\tau}_i \le \tau_i \le \bar{\tau}_i,\ i = 1, 2, \ldots, m\} \tag{11}$$

where $\tau(t)$ is the torque output vector of actuators, with $\bar{\tau}_i > 0$ and $\underline{\tau}_i < 0$.

Further consider the definitions and assumptions above, the attitude dynamic equation (2) can be obtained in the form of

$$(J_0 + \Delta J)\dot{\omega} = -\omega^{\times}(J_0 + \Delta J)\omega + (D_0 + \Delta D)\tau(t) + d(t) \tag{12}$$

Note that it is assumed that $(J_0 + \Delta J)^{-1} = J_0^{-1} - J_0^{-1}\Delta J(I_3 + J_0^{-1}\Delta J)^{-1}J_0^{-1}$ and $\Delta\bar{J} = J_0^{-1}\Delta J(I_3 + J_0^{-1}\Delta J)^{-1}J_0^{-1}$ [25].

2.3. Preliminaries

For the simplicity of expression, the following notations are defined as:

$$\mathrm{sig}^{\alpha}(x) = \mathrm{sign}(x)|x|^{\alpha} \\ = [\mathrm{sign}(x_1)|x_1|^{\alpha}, \mathrm{sign}(x_2)|x_2|^{\alpha}, \ldots, \mathrm{sign}(x_n)|x_n|^{\alpha}]^{\mathrm{T}} \tag{13}$$

where $x = [x_1, x_2, \ldots, x_n]^{\mathrm{T}} \in R^n$, $|x| = [|x_1|^{\alpha}, |x_2|^{\alpha}, \ldots, |x_n|^{\alpha}]^{\mathrm{T}}$, $0 < \alpha < 1$, and sign(.) is the sign function.

Then for the purpose of control system design, some definitions and lemmas are given from [10,11,18,35].

Definition 1. Consider the nonlinear system $\dot{x} = f(x, u)$, with x is a state vector, u is the input vector, $f(x)$ is a continuous function with $f(0) = 0$. If the system is Lyapunov stable, $x(t) = 0$ is constantly established for all $t > T(x)$, then the system is finite time stable. And the solution is practical finite-time stable, if there exist $\epsilon > 0$ and $T(\epsilon, x_0) < \infty$ with $x(t_0) = x_0$, such that $\|x(t)\| < \epsilon$, for all $t > T + t_0$.

Lemma 1. *Consider the system defined in Definition 1 and suppose that there exist $\lambda > 0$, $0 < \alpha < 1$ and $0 < \eta < \infty$ such that the continuous function $V(x)$ satisfies $\dot{V}(x) \le -\lambda V^{\alpha}(x) + \eta$. Then, the trajectory of system is practical finite-time stable. The reaching time is bounded by $T \le \frac{V^{1-\alpha}(x_0)}{\lambda\theta_0(1-\alpha)}$ with $0 < \theta_0 < 1$, and $V(x_0)$ is the initial value of $V(x)$.*

Lemma 2. *For $0 < p < 2$, $x = [x_1, x_2, \ldots, x_n]^{\mathrm{T}} \in R^n$, the following inequality holds*

$$(x_1^2 + x_2^2 + \ldots + x_n^2)^p \le (|x_1^p| + |x_2^p| + \ldots + |x_n^p|)^2.$$

2.4. Control objective

Consider the spacecraft attitude system given by Eqs. (1) and (12) with the actuator misalignment under Assumptions 1–3, the control objective is to design control input $\tau(t) \in \Omega$ such that, for all physically realizable initial conditions, the states of the closed-loop system are stable in finite time, which can be expressed as $\lim_{t\to T} q = \lim_{t\to T} \omega = 0$. And T is the convergence time, which is a function of the initial values of the system states.

3. Control law design

For convenience, an equivalent representation of Eq. (4) can be given as

$$J\dot{\omega} = -\omega^{\times} J\omega + B_u u(t) + d(t) \tag{14a}$$

$$B_u u(t) = B_{\tau}\tau(t) \tag{14b}$$

where $u(t)$ is the virtual control input or called combined control torque produced by the actuators, B_u is the virtual input matrix, and B_{τ} is used to describe distribution of the physical actuators with $B_\tau = D_0 + \Delta D$, here, representing the influence of each actuator on the angular acceleration of the spacecraft considered. For the considered spacecraft, the virtual input matrix B_u as the identity $B_u = I_{3\times3}$ is defined. With such a choice, the virtual input $u(t)$ represents exactly the total torques produced by the actuators, and the following virtual equivalent plant can be established

$$J\dot{\omega} = -\omega^\times J\omega + u(t) + d(t) \tag{15a}$$

$$u(t) = (D_0 + \Delta D)\tau(t) \tag{15b}$$

$$J = J_0 + \Delta J \tag{15c}$$

In what follows, attitude control schemes will be developed for spacecraft given by Eqs. (1) and (15) to achieve the control target stated in Section 2.4.

3.1. Disturbance observer design

Firstly, define the following coordinate transformation

$$z = \omega + kq \tag{16}$$

In view of Eqs. (1) and (12), it yields to

$$\begin{aligned}\dot{z} &= J_0^{-1}\left(-\omega^\times J_0\omega + \frac{1}{2}kJ_0(q_0\omega + q^\times\omega)\right)\\ &\quad + J_0^{-1}\left(-\omega^\times \Delta J\omega\tau + \frac{1}{2}k\Delta J(q_0\omega + q^\times\omega)\right)\\ &\quad + \Delta\bar{J}\left(-\omega^\times J\omega + \frac{1}{2}kJ(q_0\omega + q^\times\omega)\right) + \left(J_0^{-1} + \Delta\bar{J}\right)d\\ &\quad + J_0^{-1}D_0\tau + (J\Delta D + \Delta\bar{J}D)\tau \end{aligned}\tag{17}$$

Denote the following symbols as

$$\begin{aligned} A &= J_0^{-1}\left(-\omega^\times J_0\omega + \frac{1}{2}kJ_0(q_0\omega + q^\times\omega)\right)\\ B &= J_0^{-1}D_0\\ \bar{d} &= \left(J_0^{-1} + \Delta\bar{J}\right)d\\ E &= J_0^{-1}\left(-\omega^\times \Delta J\omega\tau + \frac{1}{2}k\Delta J(q_0\omega + q^\times\omega)\right)\\ &\quad + \Delta\bar{J}\left(-\omega^\times J\omega + \frac{1}{2}kJ(q_0\omega + q^\times\omega)\right)\\ &\quad + (J\Delta D + \Delta\bar{J}D)\tau\\ F &= \bar{d} + E \end{aligned}$$

Then, the simplified spacecraft system model can be given as

$$\dot{z} = A + B\tau + F \tag{18}$$

Remark 1. All the terms defined above especially for ΔJ and $\Delta\bar{J}$ are considered as sufficiently smooth functions, and they are assumed differentiable, then E is differentiable. Similarly, d is the differentiable disturbance and J_0^{-1} is a smooth term, thus $\bar{d}$ is also a differentiable function. So is the combination term of uncertainties and disturbances defined as $F = [F_1, F_2, F_3]^{\mathrm{T}}$.

For the purpose of the attitude control system design, the sliding mode surface is selected as

$$s = z = \omega + kq \tag{19}$$

where k is a time-varying variable but $k > 0$ is necessary and required to ensure the stability of the sliding mode surface. And the reaching law is involved

$$\dot{s} = -k_1 s - k_2\,\mathrm{sig}(s)^\alpha \tag{20}$$

with $k_1 > 0$, $k_2 > 0$, and $0 < \alpha < 1$. In view of above sliding mode surface and reaching law, the derivative of s can be established as

$$\dot{s} = \dot{z} = A + B\tau + F \tag{21}$$

It is clear that the sliding motion can converge to a neighborhood of the equilibrium in finite time, and then remain there by the reaching motion controller [18].

To estimate the uncertainty term F, the following lemma should be recalled.

Lemma 3. *(See [5,18,22,23].) Consider the simplified control system model governed by Eq. (18) and Eq. (21), the following second-order observer can estimate the total disturbances F*

$$\begin{aligned} \dot{\zeta}_0 &= v_0 + B\tau + A\\ v_0 &= -a_0\,\mathrm{diag}\left(L^{1/3}|\zeta_0 - s|^{2/3}\right)\mathrm{sign}(\zeta_0 - s) + \zeta_1\\ \dot{\zeta}_1 &= v_1\\ v_1 &= -a_1\,\mathrm{diag}\left(L^{1/2}|\zeta_1 - v_0|^{1/2}\right)\mathrm{sign}(\zeta_1 - v_0) + \zeta_2\\ \dot{\zeta}_2 &= -a_2 L\,\mathrm{sign}(\zeta_2 - v_1) \end{aligned}\tag{22}$$

where ζ_0, ζ_1 and ζ_2 are the observed values of s, F and $\dot{F}$; $\mathrm{diag}(.)$ denotes the diagonal matrix of the corresponding vector, that is, $\mathrm{diag}(L) = \mathrm{diag}(L_1, L_2, L_3)$ with $L = [L_1, L_2, L_3]^{\mathrm{T}}$, $L_i \geq \|\ddot{F}_i\|$ are Lipschitz constants with $i = 1, 2, 3$.

For the second-order DO in Eq. (22), further define auxiliary variables $e_0 = \zeta_0 - s$, $e_1 = \zeta_1 - F$ and $e_2 = \zeta_2 - \dot{F}$, the error dynamics can be found in the form of

$$\begin{aligned} \dot{e}_0 &= \dot{\zeta}_0 - \dot{s} = -a_0\,\mathrm{diag}\left(L^{1/3}|e_0|^{2/3}\right)\mathrm{sign}(e_0) + \zeta_1 - F\\ &= -a_0\,\mathrm{diag}\left(L^{1/3}|e_0|^{2/3}\right)\mathrm{sign}(e_0) + e_1 \end{aligned}\tag{23a}$$

$$\begin{aligned} \dot{e}_1 &= \dot{\zeta}_1 - \dot{F} = -a_1\,\mathrm{diag}\left(L^{1/2}|\zeta_1 - v_0|^{1/2}\right)\mathrm{sign}(\zeta_1 - v_0) + \zeta_2 - \dot{F}\\ &= -a_1\,\mathrm{diag}\left(L^{1/2}|e_1 - \dot{e}_0|^{1/2}\right)\mathrm{sign}(e_1 - \dot{e}_0) + e_2 \end{aligned}\tag{23b}$$

$$\dot{e}_2 = \dot{\zeta}_2 - \ddot{F} = -a_2 L\,\mathrm{sign}(e_2 - \dot{e}_1) - \ddot{F} \tag{23c}$$

then the stability of the disturbance observer can be achieved by selecting appropriate parameters a_0, a_1, a_2 and L such that the errors e_0, e_1 and e_2 can converge to zero in finite time T_d.

Remark 2. Generally speaking, the sliding-mode based disturbance observer is an effective and useful method to deal with the disturbance/uncertainty rejection problem. According to Ref. [16], the disturbance observer error states defined in Eq. (23) will converge to zero in finite time. Note that here ζ_0, ζ_1 and ζ_2 are the real-time estimated values of s, F and $\dot{F}$ respectively in this work.

3.2. Finite-time attitude controller design

In this subsection, a novel sliding mode based finite time control algorithm is investigated to achieve the target given in Section 2.4 with the help of DO designed in Eq. (23). Now come to the main result of this paper summarized and presented as follows.

Theorem 1. *Consider the spacecraft system governed by Eq. (18) with the second-order DO given in Eq. (22), there exist the observer gains such that the estimated states ζ_0, ζ_1 and ζ_2 converge to s, F_2 and $\dot{F}_2$ in finite time, respectively. And also under the control laws proposed in Eqs. (24) and (25)*

$$u_0 = -k_2\,\mathrm{sig}(s)^\alpha - J_0\zeta_1 - \frac{s}{\|s\|}\mu_1\|\omega\| - \beta k^2\,\mathrm{Tanh}(s/p^2) \tag{24}$$

$$\dot{k} = -\gamma_k\left(s^\mathrm{T}\,\mathrm{Tanh}(s/p^2) + 3\kappa(\beta k+1)p^2 + \mathrm{sig}(k)^\alpha\right) \tag{25}$$

with $\mu_1 = \|\omega^\times J_0\| + \|\frac{1}{2}k(q_0 + q^\times)\|$, $k(0) = k_0 > 0$ *and* $\gamma_k > 0$, *the following can be achieved:*

(a) *the controller will drive the sliding surface s to zero in finite time;*
(b) *the signals q and ω will also converge to zero and* q_0 *tending to 1 is guaranteed in finite time, in the closed-loop system.*

Note that here $p^2(t)$ *is strictly a bounded above zero scalar sharpness function [27], which governs the magnitude of the control rates, and* $0 < p^2_{\min} \le p^2 \in L_\infty$, $\dot{p} \in L_\infty$.

Proof. The proof of Theorem 1 is organized in the following two steps.

Step 1: Consider a Lyapunov function candidate as

$$V_1 = \frac{1}{2}s^\mathrm{T} J_0 s \tag{26}$$

which satisfies $\frac{1}{2}\phi_{J\min}\|s\|^2 \le \frac{1}{2}s^\mathrm{T} J_0 s \le \frac{1}{2}\phi_{J\max}\|s\|^2$.

Differentiating the above Lyapunov function, it yields to

$$\begin{aligned}\dot{V}_1 &= s^\mathrm{T}\left(-\omega^\times J_0\omega + \frac{1}{2}k\left(q_0\omega + q^\times\omega\right)\right) + s^\mathrm{T} D_0\tau + s^\mathrm{T} J_0 F \\ &\le \left(\left\|\omega^\times J_0\right\| + \left\|\frac{1}{2}k\left(q_0 + q^\times\right)\right\|\right)\|s\|\|\omega\| \\ &\quad + s^\mathrm{T} u_0 + s^\mathrm{T} J_0 F\end{aligned} \tag{27}$$

In view of control law equation (24), one has

$$\dot{V}_1 \le -k_2 s^\mathrm{T}\,\mathrm{sig}(s)^\alpha - s^\mathrm{T}\left(J_0 e_1 + \beta k^2\,\mathrm{Tanh}(s/p^2)\right) \tag{28}$$

From Lemma 3, it can be obtained that F is compensated by the real-time second-order disturbance observer. In other words, the observer error $e_1 = \zeta_1 - F = 0$ after finite time T_d. Under this, from the second term of Eq. (28), one can obtain

$$-s^\mathrm{T}\left(J_0 e_1 + \beta k^2\,\mathrm{Tanh}(s/p^2)\right) = -\beta k^2 \sum_{i=1}^{3} s_i\,\mathrm{Tanh}(s_i/p^2) \tag{29}$$

To this end, one has $\dot{V}_1 \le -k_2\|s\|^{\alpha+1} - \beta k^2 \sum_{i=1}^{3} s_i\,\mathrm{Tanh}(s_i/p^2)$. Then, if the parameter $\beta > 0$ is chosen such that $\beta k^2 > 0$, one has $\dot{V}_1 < 0$, which implies the sliding mode state s is bounded for all the time under the control law equation (24).

Step 2: Consider the following extended Lyapunov function

$$V = \frac{1}{2}s^\mathrm{T} J_0 s + \frac{1}{2\gamma_k}k^2 \tag{30}$$

Differentiating the above Lyapunov function with respect to time, it follows that

$$\begin{aligned}\dot{V} &= s^\mathrm{T}\left(-\omega^\times J_0\omega + \frac{1}{2}k_0\left(q_0\omega + q^\times\omega\right)\right) + s^\mathrm{T} D_0\tau + s^\mathrm{T} J_0 F \\ &\quad - \frac{1}{\gamma_k}k\gamma_k\left(s^\mathrm{T}\,\mathrm{Tanh}(s/p^2) + 3\kappa(\beta k+1)p^2 + \mathrm{sig}(k)^\alpha\right) \\ &\le \left(\left\|\omega^\times J_0\right\| + \left\|\frac{1}{2}k_0\left(q_0 + q^\times\right)\right\|\right)\|s\|\|\omega\| + s^\mathrm{T} u_0 \\ &\quad + s^\mathrm{T} J_0 F - ks^\mathrm{T}\,\mathrm{Tanh}(s/p^2) - 3\kappa(\beta k+1)kp^2 - k\,\mathrm{sig}(k)^\alpha\end{aligned} \tag{31}$$

According to the control law equation (24) and the auxiliary time-varying gain function in Eq. (25), it follows:

$$\begin{aligned}\dot{V} &\le -k_2 s^\mathrm{T}\,\mathrm{sig}(s)^\alpha - s^\mathrm{T}\left(\beta k^2 + k\right)\mathrm{Tanh}(s/p^2) \\ &\quad - 3\kappa(\beta k+1)kp^2 - k\,\mathrm{sig}(k)^\alpha\end{aligned} \tag{32}$$

Using the following factors

$$0 \le |m|\left(1 - \tanh|m/n|\right) \le \kappa|n| \tag{33}$$

where m and $n \ne 0$ are real scalar variables, and κ is a positive constant with minimum value $\kappa_0 = m_0(1 - \tanh m_0) < e^{-1}$ for any m_0 satisfying $e^{-2m_0} + 1 - 2m_0 = 0$. Thus, one can obtain that

$$\begin{aligned}&-s^\mathrm{T}(\beta k+1)k\,\mathrm{Tanh}(s/p^2) \\ &\quad = -(\beta k+1)k\|s\|_1 + (\beta k+1)k\left(\sum_{i=1}^{3}|s_i|\left(1 - \mathrm{Tanh}(s_i/p^2)\right)\right) \\ &\quad \le -(\beta k+1)k\|s\|_1 + 3\kappa(\beta k+1)kp^2\end{aligned} \tag{34}$$

In view of the results of Step 1, the sliding mode state s is bounded by $\|s\|_1 \le c_m$ with c_m a positive scalar constant. Then one can get

$$\begin{aligned}\dot{V} &\le -\frac{k_2}{(1/2\phi_{J\max})^{\frac{\alpha+1}{2}}}\left(\frac{1}{2}s^\mathrm{T} J_0 s\right)^{\frac{\alpha+1}{2}} \\ &\quad - (2\gamma_k)^{\frac{\alpha+1}{2}}\left(\frac{1}{2\gamma_k}k^2\right)^{\frac{\alpha+1}{2}} + (\beta k+1)kc_m \\ &\le -\chi V^{\alpha+1/2} + \eta\end{aligned} \tag{35}$$

with $\gamma_k = \frac{1}{2}\chi^{\frac{2}{\alpha+1}}$, $\chi = \frac{k_2}{(1/2\phi_{J\max})^{\frac{\alpha+1}{2}}}$, $\eta = \beta k^2 c_m$, and appropriate value of the parameters should be chosen by the constraint of $(\beta k+1)kc_m > 0$. According to Lemma 1, the decreasing Lyapunov function $V(x)$ can drive the states of the closed-loop system into $V^{\alpha+1/2}(x) \le \frac{\eta}{(1-\theta_0)\chi}$, which implies that the trajectories of the spacecraft attitude stabilization system are bounded in finite time under the control scheme, Eqs. (24) and (25), as

$$\lim_{t\to T} s(t) \in \left(\|s\| \le \sqrt{\frac{\phi_{J\max}}{\phi_{J\min}}\left(\frac{\eta}{(1-\theta_0)k_2}\right)^{\alpha+1/2}}\right)$$

which is a small set involving the origin of the closed-loop attitude control system. □

3.3. Finite time controller design under actuator saturation

Another important problem encountered in practice is that of actuator saturation issue. In theory, actuators are required to produce fast and large enough joint torques under desired control schemes, which will be hardly achieved due to limited magnitude and slew rate constraints for practical actuators. As a consequence, the demanded control signals will quickly saturate actuators striving to maintain fine performances to the best of their abilities. Then, the occurrence of actuator saturation will subsequently reduce performances and even destabilize the closed-loop system, if the system is not equipped with an appropriate control scheme to reject the issue of saturation. However, the actuator saturation problem, which can lead to severe discrepancies between commanded input signals and actual control efforts, is not considered in the above controller design explicitly. Therefore, it is important to study the stability, robustness, and effective control law of those systems with actuator input saturation. While, taking account of some practical problems, such as energy saving (minimizing energy consumption), capability for actuator saturation avoidance, a modified DO based finite time controller will be developed in this subsection.

Table 1
Main parameters of rigid spacecraft.

Mission	Attitude stabilization
Mass (kg)	874.56
Inertia moments ($kg\,m^2$)	
Principal moments of inertia	$J_{11}=20,\ J_{22}=17,\ J_{33}=15$
Products of inertia	$J_{12}=0,\ J_{23}=0,\ J_{13}=0.9$
Uncertainties of inertia	$\Delta J_{11}=1.2,\ \Delta J_{22}=1.0,\ \Delta J_{33}=0.8$ $\Delta J_{12}=0.1,\ \Delta J_{13}=-0.1,\ \Delta J_{21}=0.2$ $\Delta J_{23}=0.2,\ \Delta J_{31}=-0.3,\ \Delta J_{32}=0.1$
Orbit	
Type	Circular
Attitude (km)	500
The inclination (deg)	97.4
The right ascension of the ascending node	10.30 am
Attitude	
Attitude control type	Three axis control by four reaction fly-wheels
Initial attitude quaternion	$Q_0=[q_0,q^T]^T=[0.9,\ -0.3,\ 0.26,\ 0.18]$
Initial angular velocity(rad/s)	$w_0=[0,0,0]^T$
Reaction fly-wheels	
Inertia moments ($kg\,m^2$)	0.409
The assembling location/angles (deg)	$\alpha_4=35.26,\ \beta_4=45$
The misalignment error angles (deg)	$\Delta\alpha_i=[0.2;\ 0.1;\ 0.2;\ 0.1],\ \Delta\beta_i=[0.1;\ 0.2;\ 0.1;\ 0.2]$
The limitation output torques (N m)	$\bar{\tau}_i=1.5,\ \underline{\tau}_i=-1.5$

Consider the sliding mode surface in Eq. (19) with actuator input saturation, Eq. (21) has the form of

$$\dot{s}=\dot{z}=A+B\,\mathrm{sat}(\tau)+F_1 \tag{36}$$

with

$$A=J_0^{-1}\left(-\omega^\times J_0\omega+\frac{1}{2}kJ_0(q_0\omega+q^\times\omega)\right)$$
$$B=J_0^{-1}D_0$$
$$\bar{d}=(J_0^{-1}+\Delta\bar{J})d,\qquad F_1=\bar{d}+E_1\quad\text{and}$$
$$E_1=J_0^{-1}\left(-\omega^\times\Delta J\omega\,\mathrm{sat}(\tau)+\frac{1}{2}k\Delta J(q_0\omega+q^\times\omega)\right)$$
$$+\Delta\bar{J}\left(-\omega^\times J\omega+\frac{1}{2}kJ(q_0\omega+q^\times\omega)\right)$$
$$+(J\Delta D+\Delta\bar{J}D)\,\mathrm{sat}(\tau).$$

Note that here $\mathrm{sat}(\tau)=[\mathrm{sat}(\tau_1)\ \ \mathrm{sat}(\tau_2)\ \ \mathrm{sat}(\tau_3)\ \ \mathrm{sat}(\tau_4)]^T$ is the actual actuator control vector, in which $\mathrm{sat}(\tau_i)$, $i=1,2,3,4$, is defined as

$$\mathrm{sat}(\tau_i)=\begin{cases}\bar{\tau}_i, & \tau_i>\bar{\tau}_i\\ \tau_i, & \underline{\tau}\le\tau_i\le\bar{\tau}_i\\ \underline{\tau}_i, & \tau_i<\underline{\tau}\end{cases} \tag{37}$$

Further define an auxiliary variable $\delta=[\delta_1\ \delta_2\ \delta_3\ \delta_4]^T$ as

$$\delta_i=\begin{cases}\bar{\tau}_i-\tau_i, & \tau_i>\bar{\tau}_i\\ 0, & \underline{\tau}\le\tau_i\le\bar{\tau}_i\\ \underline{\tau}_i-\tau_i, & \tau_i<\underline{\tau}\end{cases} \tag{38}$$

Eqs. (37) and (38) yield to

$$\mathrm{sat}(\tau)=\delta+\tau \tag{39}$$

Then, submitting Eq. (39) into Eq. (36), it follows that

$$\dot{s}=\dot{z}=A+B\tau+F_2 \tag{40}$$

where $E_2=J_0^{-1}(-\omega^\times\Delta J\omega\,\mathrm{sat}(\tau)+\frac{1}{2}k\Delta J(q_0\omega+q^\times\omega))+\Delta\bar{J}(-\omega^\times J\omega+\frac{1}{2}kJ(q_0\omega+q^\times\omega))+(J\Delta D+\Delta\bar{J}D)\,\mathrm{sat}(\tau)+B\delta$, and $F_2=\bar{d}+E_2$ is the uncertainty which can be estimated by the second-order DO designed in Eq. (22). Therefore, one can obtain another main result of this paper stated as follows.

Table 2
Control parameters for numerical simulation.

Control schemes	Control gains
Observer	$a_0=3.2,\ a_1=1.0,\ a_2=0.6,\ L=\mathrm{diag}(0.6,\ 0.6,\ 0.6)$
Controller	$k_2=8.6,\ \alpha=0.6,\ \kappa=0.02,\ p^2=1.2,\ \beta=0.12,$ $\gamma_k=0.05,\ k_0=0.9,\ \beta=0.02,\ \omega_d=0.1.$

Theorem 2. *Consider the spacecraft system governed by Eq. (18) with the second-order DO given in Eq. (22) and the proposed control laws in Eqs. (24) and (25). If there exist observer gains such that the estimated states ζ_0, ζ_1 and ζ_2 converge to s, F_2 and $\dot{F}_2$ in finite time, respectively, then the following results are achieved.*

(a) *The controller will drive the sliding surface s to zero in finite time.*
(b) *The signals q and ω will also converge to zero and q_0 tending to 1 is guaranteed in finite time.*
(c) *The actuator output will be prevented from saturation via the compensation of the observer.*

Proof. The proof of Theorem 2 is similar to that of Theorem 1, which is omitted here. □

4. Simulation and analysis

To verify the effectiveness and performance of the proposed second-order DO in Eq. (22) (presented in Lemma 3) and time-varying attitude stabilization controller in Eqs. (24) and (25) (noted as DOFTC), numerical simulations have been conducted in this section using the rigid spacecraft system governed by Eqs. (1) and (15). The simulation parameters are presented in Tables 1 and 2, and the external disturbances are assumed to be

$$d(t)=0.2\times10^{-3}*\begin{bmatrix}3\cos(10\omega_d t)+4\sin(3\omega_d t)-10\\ -1.5\sin(2\omega_d t)+3\cos(5\omega_d t)+15\\ 3\sin(10\omega_d t)-8\sin(4\omega_d t)+10\end{bmatrix}\ \mathrm{N\,m}.$$

For the purpose of comparison, both the finite-time controller presented in Ref. [18] (noted as FTC-Ref. [18]) and the robust controller under control input saturation in Ref. [21] (noted as SRC-Ref. [2]) are also applied to the control issue considered above.

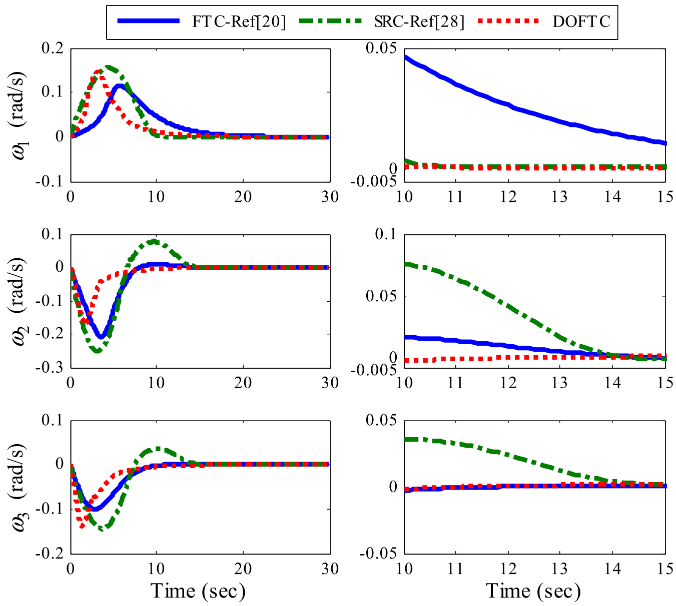

Fig. 3. Time responses of the spacecraft attitude angular velocity $\boldsymbol{\omega}$.

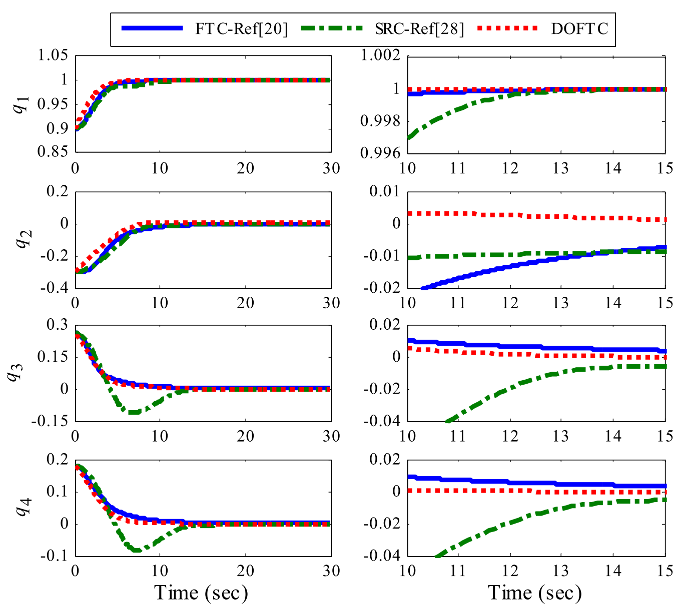

Fig. 4. Time responses of the spacecraft attitude quaternion $\boldsymbol{q}$.

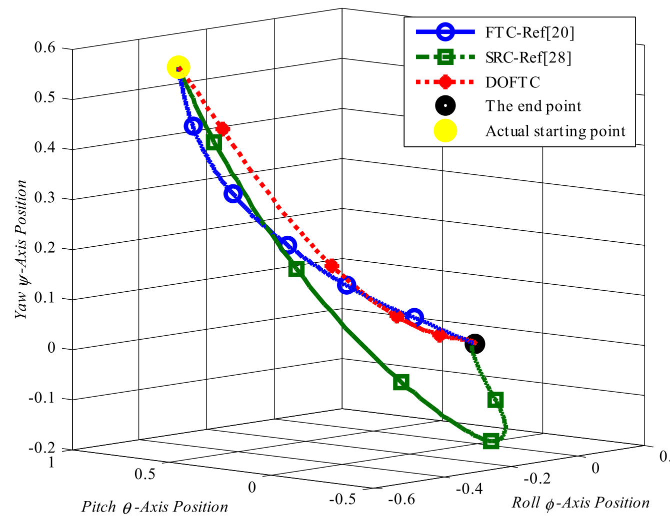

Fig. 5. Time responses of the attitude angles.

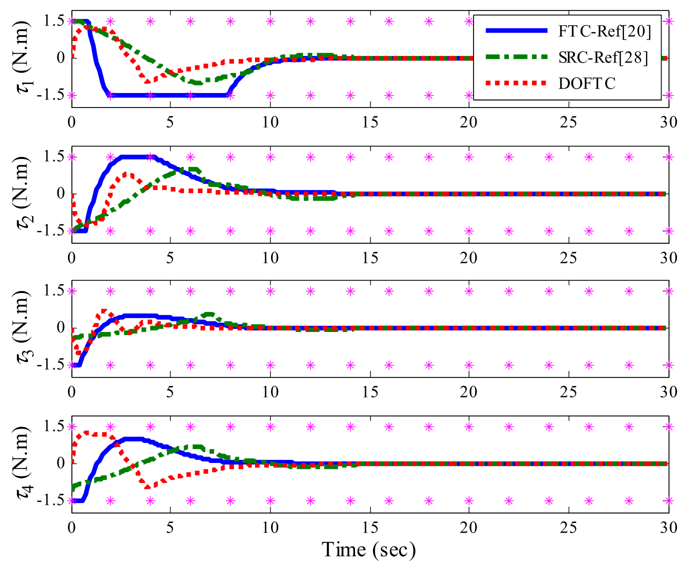

Fig. 6. Time responses of the demand control torques $\boldsymbol{\tau}$.

More specially, the simulations (performances) of the attitude control system with actuator/inertia uncertainties and external disturbances even actuator saturation are carried out as follows, using the same values of parameters designed above.

All computations and plots are performed using the MATLAB/SIMULINK software package. Figs. 3 and 4 show time responses of spacecraft attitude angular velocity and the attitude quaternion. As we can see, the spacecraft attitude system achieves stabilization in finite time smoothly with a settling time less than 10 s with a high accuracy with 10^{-3} in steady-state error. Although the other two schemes can achieve the attitude stabilization, more time or larger control torque (system energy/fuel) need to be required. From the point view of practice, the spacecraft attitude responses using Euler angle $[\phi\ \theta\ \psi]^{\mathrm{T}}$ (ϕ, θ and ψ are, respectively, the roll, pitch, and yaw angles), are shown as Fig. 5. It is clear that the results of SRC-Ref. [2] and FTC-Ref. [18] schemes show poorer performances with great stabilization error or drifting or non-optimal tracking trajectory than the proposed control scheme. Accordingly, the case of DOFTC achieves the best performance among of the three cases above, which further illustrates the effectiveness and feasibility of the proposed method.

Time responses of the demand control torques $\boldsymbol{\tau}$ are depicted in Fig. 6. Note that the restrictions on the actuator output torque magnitude $(\underline{\tau}, \bar{\tau})$ with $\bar{\tau}_i = 1.5$ N m and $\underline{\tau}_i = -1.5$ N m are considered explicitly. From the curves of the control torques $\boldsymbol{\tau}$, we can see that the actuator output derived from the proposed strategy DOFTC has been avoided from saturation successfully due to compensation design in Subsection 3.3. As shown in Fig. 6, comparing the cases among FTC-Ref. [18], SRC-Ref. [2] and DOFTC, the output control signals $[\tau_1, \tau_2, \tau_3, \tau_4]^{\mathrm{T}}$ of the controller FTC-Ref. [18] are obviously larger than the other two controllers with input saturation, and it also leads to the saturated reaction wheel torque in τ_1 and τ_2 with a longer period. Whereas, the output control torques of the controller SRC-Ref. [2] still generate saturation of the reaction wheels at the beginning period of the time responses, although the control input saturation is considered in the robust controller design, especially in τ_1 and τ_2.

Furthermore, for analyzing the effectiveness of the second-order DO proposed in Eq. (22), Figs. 7 and 8 show the state observed

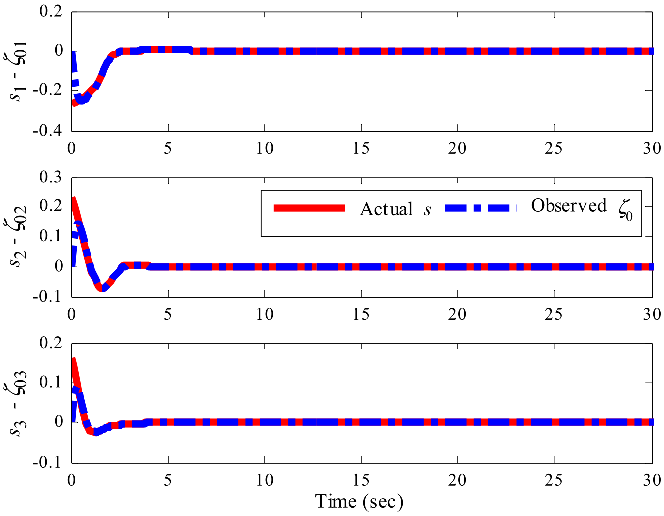

Fig. 7. Time responses of the observe value $\boldsymbol{s} - \zeta_0$.

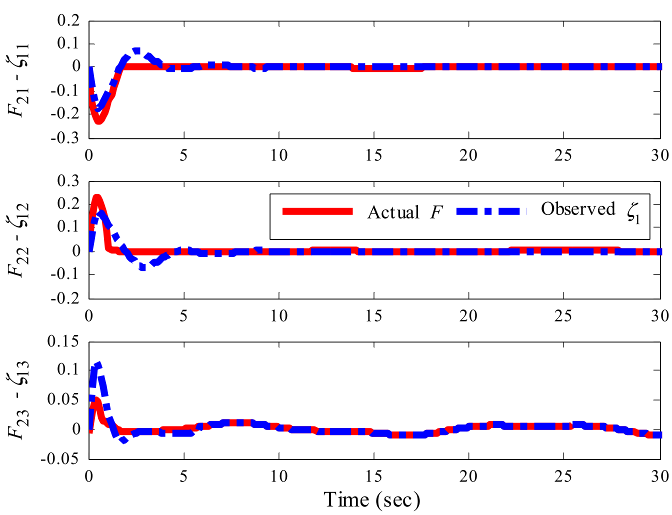

Fig. 8. Time responses of the observe value $\boldsymbol{F}_2 - \zeta_1$.

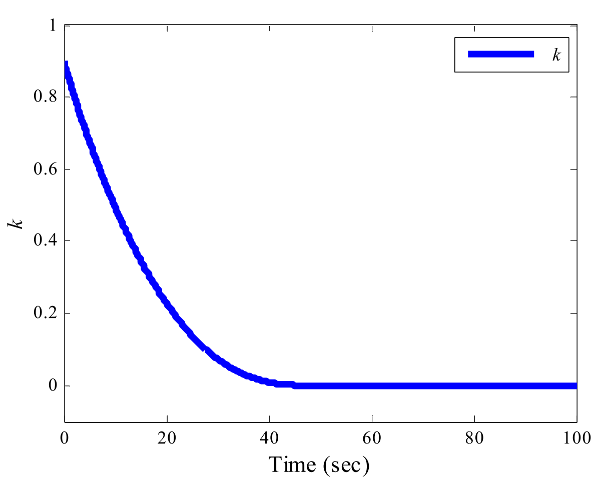

Fig. 9. Time responses of the time-varying gain $\boldsymbol{k}$.

values of the sliding-mode $\boldsymbol{s}$ and the combined disturbances $\boldsymbol{F}_2$ defined in Eq. (39), respectively. As one can see, the observer states ζ_0 and ζ_1 achieve a fine and fast tracking to the actual variables $\boldsymbol{s}$ and $\boldsymbol{F}_2$, and ζ_0 tracks to $\boldsymbol{s}$ entirely and exactly within 5 s then ζ_1 to $\boldsymbol{F}_2$ within 10 s. In brief, the DO can estimate the states such as time-varying sliding mode and total disturbances effectively. Additionally, the time response of the adaptive variable $\boldsymbol{k}$ (the time-varying gain of sliding mode surface) is shown in Fig. 9, which decreases to zero monotonically in finite time.

Summarizing all the cases above, it is noted that the proposed controller (DOFTC) can significantly achieve better performances than FTC-Ref. [18] or SRC-Ref. [2] method in both theory and simulation. Moreover, the flexibility in the choice of control parameters can be utilized to obtain desirable performance while meeting the constraints on the control magnitude and uncertainties. These control approaches provide the theoretical basis for the practical application of the advanced control theory to spacecraft control system design.

5. Conclusion

In this paper, a novel time-varying sliding mode control based finite-time controller incorporated a second-order disturbances observer is proposed for a rigid spacecraft attitude stabilization control system. More specially, a second-order observer/differentiator is designed firstly to estimate the time-varying sliding mode manifold and the combined disturbances caused by external disturbances, spacecraft inertia uncertainties and actuator misalignments. With the estimated values derived from the observer, a finite-time controller and an adaptive law of the time-varying variable gain are presented, and Lyapunov stability analysis shows that finite-time convergence of the uncertain spacecraft attitude control system to the equilibrium point can be accomplished with great robustness to disturbances and uncertainties even actuator saturations. Numerical simulation results for in-orbit rigid spacecraft model show fine performance, which validates the effectiveness and feasibility of the proposed scheme. However, the actuator faults such as lose of effectiveness or stuck are not considered directly in this paper, this is one of subjects for future research.

Conflict of interest statement

We declare that we have no financial and personal relationships with other people or organizations that can inappropriately influence our work, there is no professional or other personal interest of any nature or kind in any product, service and/or company that could be construed as influencing the position presented in, or the review of, the manuscript entitled, "Disturbance observer based finite-time attitude control for rigid spacecraft under input saturation".

Acknowledgements

This present work was supported by the National Natural Science Foundation of China (61273175), Program for New Century Excellent Talents in University (NCET-11-0801), Heilongjiang Province Science Foundation for Youths (QC2012C024), Research Fund for Doctoral Program of Higher Education of China (20132302110028), and Open Project Program of State Key Laboratory of Robotics (2013-O11).

References

[1] J.D. Bošković, S.M. Li, R.K. Mehra, Robust adaptive variable structure control of spacecraft under control input saturation, J. Guid. Control Dyn. 24 (1) (2001) 14–22.
[2] J.D. Bošković, S.M. Li, R.K. Mehra, Robust tracking control design for spacecraft under control input saturation, J. Guid. Control Dyn. 27 (4) (2004) 627–633.
[3] S.H. Ding, S.H. Li, Stabilization of the attitude of a rigid spacecraft with external disturbances using finite-time control techniques, Aerosp. Sci. Technol. 13 (4–5) (2009) 256–265.
[4] H.B. Du, S.H. Li, Finite-time attitude stabilization for a spacecraft using homogeneous method, J. Guid. Control Dyn. 35 (3) (2012) 740–748.

[5] L. Guo, W.H. Chen, Disturbance attenuation and rejection for systems with nonlinearity via DOBC approach, Int. J. Robust Nonlinear Control 15 (3) (2005) 109–125.
[6] Q.L. Hu, Variable structure maneuvering control with time-varying sliding surface and active vibration damping of flexible spacecraft with input saturation, Acta Astronaut. 64 (11–12) (2009) 1085–1108.
[7] Q.L. Hu, Robust adaptive sliding mode attitude maneuvering and vibration damping of three-axis-stabilized flexible spacecraft with actuator saturation limits, Nonlinear Dyn. 55 (4) (2009) 301–321.
[8] Q.L. Hu, Robust adaptive sliding mode fault tolerant control with L_2 gain performance for flexible spacecraft using redundant reaction wheels, IET Control Theory Appl. 4 (6) (2010) 1055–1070.
[9] Q.L. Hu, M.I. Friswell, Robust variable structure attitude control with L_2-gain performance for a flexible spacecraft including input saturation, Proc. Inst. Mech. Eng., Part I, J. Syst. Control Eng. 224 (2) (2010) 153–167.
[10] Q.L. Hu, X. Huo, B. Xiao, et al., Robust finite-time control for spacecraft attitude stabilization under actuator fault, Proc. Inst. Mech. Eng., Part I, J. Syst. Control Eng. 226 (3) (2012) 416–428.
[11] Q.L. Hu, B. Li, A.H. Zhang, Robust finite-time control allocation in spacecraft attitude stabilization under actuator misalignment, Nonlinear Dyn. 73 (1) (2013) 53–71.
[12] Q.L. Hu, B. Li, Y.M. Zhang, Robust attitude control design for spacecraft under assigned velocity and control constraints, ISA Trans. 52 (4) (2013) 480–493.
[13] E.D. Jin, Z.W. Sun, Robust controllers design with finite time convergence for rigid spacecraft attitude tracking control, Aerosp. Sci. Technol. 12 (4) (2008) 324–330.
[14] Y.Q. Jin, X.D. Liu, W. Qiu, et al., Time-varying sliding mode controls in rigid spacecraft attitude tracking, Chin. J. Aeronaut. 21 (4) (2008) 352–360.
[15] J.L. Junkins, M.R. Akella, R.D. Robinert, Nonlinear adaptive control of spacecraft maneuvers, J. Guid. Control Dyn. 20 (6) (1997) 1104–1110.
[16] A. Levant, Higher-order sliding modes, differentiation and output feedback control, Int. J. Control 76 (9–10) (2003) 924–941.
[17] S.H. Li, Z. Wang, S.M. Fei, Comments on the paper: robust controllers design with finite time convergence for rigid spacecraft attitude tracking control, Aerosp. Sci. Technol. 15 (3) (2011) 193–195.
[18] K.F. Lu, Y.Q. Xia, Z. Zhu, et al., Sliding mode attitude tracking of rigid spacecraft with disturbances, J. Franklin Inst. 349 (2) (2012) 413–440.
[19] W.C. Luo, Y.C. Chu, K.V. Ling, H_∞ inverse optimal attitude-tracking control of rigid spacecraft, J. Guid. Control Dyn. 28 (3) (2005) 481–493.
[20] C. Pukdeboon, A.S.I. Zinober, Control Lyapunov function optimal sliding mode controllers for attitude tracking of spacecraft, J. Franklin Inst. 349 (2) (2012) 456–475.
[21] A. Sabanovic, Variable structure systems with sliding modes in motion control–a survey, IEEE Trans. Ind. Inform. 7 (2) (2011) 212–223.
[22] Y.B. Shtessel, I.A. Shkolnikov, A. Levant, Smooth second-order sliding modes: missile guidance application, Automatica 43 (8) (2007) 1470–1476.
[23] Y.B. Shtessel, I.A. Shkolnikov, A. Levant, Guidance and control of missile interceptor using second-order sliding modes, IEEE Trans. Aerosp. Electron. Syst. 45 (1) (2009) 110–124.
[24] M.J. Sidi, Spacecraft Dynamics and Control, Cambridge University Press, London, 1997, pp. 88–97.
[25] C. Song, S.J. Kim, S.H. Kim, et al., Robust control of the missile attitude based on quaternion feedback, Control Eng. Pract. 14 (7) (2006) 811–818.
[26] V.I. Utkin, Variable structure systems with sliding modes, IEEE Trans. Autom. Control 22 (2) (1977) 212–222.
[27] R.J. Wallsgrove, M.R. Akella, Globally stabilizing saturated attitude control in the presence of bounded unknown disturbances, J. Guid. Control Dyn. 28 (5) (2005) 957–963.
[28] J.Y. Wang, Z.W. Sun, 6-DOF robust adaptive terminal sliding mode control for spacecraft formation flying, Acta Astronaut. 73 (0) (2012) 76–87.
[29] J.Y. Wang, H.Z. Liang, Z.W. Sun, et al., Finite-time control for spacecraft formation with dual-number-based description, J. Guid. Control Dyn. 35 (3) (2012) 950–962.
[30] X.J. Wei, L. Guo, Composite disturbance observer based control and terminal sliding mode control for nonlinear systems with disturbances, Int. J. Control 82 (6) (2009) 1082–1098.
[31] B. Wie, J. Lu, Feedback control logic for spacecraft eigenaxis rotations under slew rate and control constraints, J. Guid. Control Dyn. 18 (6) (1995) 1372–1379.
[32] S.N. Wu, X.Y. Sun, Z.W. Sun, et al., Sliding mode control for staring mode spacecraft using a disturbance observer, Proc. Inst. Mech. Eng., G J. Aerosp. Eng. 224 (2) (2010) 215–224.
[33] F.K. Yeh, Sliding–mode adaptive attitude controller design for spacecrafts with thrusters, IET Control Theory Appl. 4 (7) (2010) 1254–1264.
[34] Z. Zhu, Y.Q. Xia, M.Y. Fu, Adaptive sliding mode control for attitude stabilization with actuator saturation, IEEE Trans. Ind. Electron. 58 (10) (2001) 4898–4907.
[35] Z. Zhu, Y.Q. Xia, M.Y. Fu, Attitude stabilization of rigid spacecraft with finite-time convergence, Int. J. Robust Nonlinear Control 21 (6) (2011) 686–702.