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Attitude control of rigid spacecraft with predefined-time stability

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Abstract

This paper investigates the attitude tracking problem of a rigid spacecraft using contemporary predefined-time stability theory. To this end, the relative attitude kinematic and dynamic models of a spacecraft are presented. Then, a sliding mode surface and predefined-time stability theory are applied to ensure that both the tracking errors of the attitude, expressed by the quaternion and the angular velocity, converge to zero within a prescribed time. Simulation results demonstrate the performance of the proposed scheme.

1. Introduction

Spacecraft attitude tracking problems have been extensively studied in the past decades due to their significant roles in space applications [1,2]. Spacecraft attitude adjustment is a complex control problem owing to the existence of the interacting nonlinear kinematic and dynamic models, and unpredictable environmental disturbances in space [3–5].

In this paper, we focus on the attitude tracking problem of a spacecraft in the presence of an unknown external disturbance. As is well established, several nonlinear control methodologies

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have been employed to solve this problem, such as PD/PID control [6], adaptive control [7–9], sliding mode control [10,11], and other methods regarding robust control methodology [12,13]. However, it should be pointed out that the majority of existing attitude tracking control schemes is designed to be asymptotically stable, implying that it requires an infinite time for the tracking errors to converge to zero [14,15]. In recent years, it is appreciable that many space applications require stringent time response constraints [16]. Driven by these requirements, the idea of finite-time stability was proposed by researchers and often combined with sliding mode control to provide finite time convergence for dynamic systems [4]. More specifically, it has been proved that closed-loop systems with finite-time controllers possess faster convergence rates and better disturbance rejection than asymptotically stable systems [17]. Nevertheless, the aforementioned finite time is usually an unbounded function with respect to the system's initial conditions, and thus it is often conservative and inaccurate [18]. To overcome this problem, fixed-time stability control was developed with the advantage that the convergence time is globally bounded [19]. However, the direct relationship between the fixed stabilization time and tuning parameters is often difficult to obtain [20]. Furthermore, the concept of prescribed-time stability control was proposed with the advantage that the control gains can be formulated as functions of the desired convergence time [21]. However, since the prescribed time is a conservative estimation for the upper bound of the stabilization time, it is often much larger than the real convergence time.

Inspired by the aforementioned results, the concept of predefined-time stability control was developed by Sanchez-Torres in [22]. Its main advantage lies in the fact that the system settling time, which can be considered as the minimum upper bound of the fixed stabilization time, appears explicitly in the tuning parameters, and thus it can be defined in advance [23]. Predefined-time sliding mode controllers were proposed in [18] for the first-order dynamical systems in the presence of uncertainty, and then they were extended to the second-order systems in [24] and [25]. It can be observed from [24] and [25] that the predefined time is composed of two parts: time for the states to reach the sliding manifolds and time for the states to converge to an equilibrium on the sliding manifolds. In addition, predefined-time stability control theory is recently employed for the tracking of robotic manipulators in [26] and for a class of uncertain chained-form nonholonomic systems in [27].

To the best of our knowledge, the predefined-time stability control theory has not been applied in the field of spacecraft attitude control. In this paper, we concentrate on developing a predefined-time stability controller for the attitude tracking problem of a spacecraft in the presence of both endogenous and exogenous uncertainties. Finally, illustrative simulations will demonstrate that with the proposed control technique, the spacecraft can track its desired attitude within the predefined time.

2. Preliminaries and problem formulation

The attitude of a rigid spacecraft can be expressed by quaternion $\boldsymbol{q}_v = [q_0 \ q_1 \ q_2 \ q_3]^{\mathrm{T}} = [q_0 \ \boldsymbol{q}^{\mathrm{T}}]^{\mathrm{T}} \in \mathbb{R}^4$. The kinematics with $\boldsymbol{q}_v$ is given by

$$\dot{\boldsymbol{q}}_v = \begin{bmatrix} \dot{q}_0 \\ \dot{\boldsymbol{q}} \end{bmatrix} = \frac{1}{2}\begin{bmatrix} -\boldsymbol{q}^{\mathrm{T}} \\ \boldsymbol{q}^{\times} + q_0\boldsymbol{I}_3 \end{bmatrix}\boldsymbol{\omega}_b \tag{1}$$

where $\boldsymbol{\omega}_b \in \mathbb{R}^3$ denotes the angular velocity of the spacecraft body-fixed frame $\mathfrak{R}_b$ with respect to $\mathfrak{R}_J$ and expressed in $\mathfrak{R}_b$, $(\cdot)^{\times}$ generates a skew symmetric matrix constructed by the vector's elements, and $\boldsymbol{I}_3$ represents the identity matrix.

As for the attitude tracking problem, the desired attitude quaternion is denoted as $\boldsymbol{q}_{dv} = \left[q_{d0}\ \boldsymbol{q}_d\right]^{\mathrm{T}} \in \mathbb{R}^4$, and $\boldsymbol{\omega}_d \in \mathbb{R}^3$ denotes the desired angular velocity. Then, the desired kinematics is written as

$$\dot{\boldsymbol{q}}_{dv} = \begin{bmatrix} \dot{q}_{d0} \\ \dot{\boldsymbol{q}}_d \end{bmatrix} = \frac{1}{2}\begin{bmatrix} -\boldsymbol{q}_d^{\mathrm{T}} \\ \boldsymbol{q}_d^{\times} + q_{d0}\boldsymbol{I}_3 \end{bmatrix}\boldsymbol{\omega}_d \tag{2}$$

Similarly, with $\boldsymbol{q}_{ev} = \left[q_{e0}\ \boldsymbol{q}_e\right]^{\mathrm{T}} \in \mathbb{R}^4$ being the attitude error quaternion and $\boldsymbol{\omega}_e \in \mathbb{R}^3$ denoting the error angular velocity, the error kinematics is expressed as

$$\dot{\boldsymbol{q}}_{ev} = \begin{bmatrix} \dot{q}_{e0} \\ \dot{\boldsymbol{q}}_e \end{bmatrix} = \frac{1}{2}\begin{bmatrix} -\boldsymbol{q}_e^{\mathrm{T}} \\ \boldsymbol{q}_e^{\times} + q_{e0}\boldsymbol{I}_3 \end{bmatrix}\boldsymbol{\omega}_e \tag{3}$$

Meanwhile, $\boldsymbol{q}_{ev}$ can be obtained by

$$\boldsymbol{q}_{ev} = \begin{bmatrix} q_{e0} \\ \boldsymbol{q}_e \end{bmatrix} = \begin{bmatrix} \boldsymbol{q}_d^{\mathrm{T}}\boldsymbol{q} + q_0 q_{d0} \\ q_{d0}\boldsymbol{q} - \boldsymbol{q}_d^{\times}\boldsymbol{q} - q_0\boldsymbol{q}_d \end{bmatrix} \tag{4}$$

and $\boldsymbol{\omega}_e$ is given by

$$\boldsymbol{\omega}_e = \boldsymbol{\omega}_b - \boldsymbol{C}_{bd}\boldsymbol{\omega}_d \tag{5}$$

with $\boldsymbol{C}_{bd}$ being the transformation matrix from $\mathfrak{R}_d$ to $\mathfrak{R}_b$ and calculated by $\boldsymbol{C}_{bd} = (q_{e0}^2 - \boldsymbol{q}_e^{\mathrm{T}}\boldsymbol{q}_e)\boldsymbol{I}_3 + 2\boldsymbol{q}_e\boldsymbol{q}_e^{\mathrm{T}} - 2q_{e0}\boldsymbol{q}_e^{\times}$.

The attitude dynamics of the spacecraft can be written as

$$\boldsymbol{J}_b\dot{\boldsymbol{\omega}}_b + \boldsymbol{\omega}_b^{\times}\boldsymbol{J}_b\boldsymbol{\omega}_b = \boldsymbol{u} + \boldsymbol{d} \tag{6}$$

where $\boldsymbol{J}_b \in \mathbb{R}^{3\times3}$ is the spacecraft inertial matrix with respect to $\mathfrak{R}_b$. Moreover, $\boldsymbol{u} \in \mathbb{R}^3$ and $\boldsymbol{d} \in \mathbb{R}^3$ denote the control torque and the external disturbance torque, respectively.

Combining Eqs. (3), (5), and (6), the error kinematics and dynamics can be rewritten as:

$$\begin{cases} \dot{\boldsymbol{q}}_e = \dfrac{1}{2}E\left(\boldsymbol{q}_{ev}\right)\boldsymbol{\omega}_e \\ \dot{\boldsymbol{\omega}}_e = \boldsymbol{\omega}_e^{\times}\boldsymbol{C}_{bd}\boldsymbol{\omega}_d - \boldsymbol{J}_b^{-1}\left[(\boldsymbol{C}_{bd}\boldsymbol{\omega}_d)^{\times}\boldsymbol{J}_b\boldsymbol{\omega}_b + \boldsymbol{\omega}_e^{\times}\boldsymbol{J}_b\boldsymbol{\omega}_b - \boldsymbol{u} - \boldsymbol{d}\right] \end{cases} \tag{7}$$

where $E\left(\boldsymbol{q}_{ev}\right) = \left[\boldsymbol{q}_e^{\times} + q_{e0}\boldsymbol{I}_{3\times3}\right] \in \mathbb{R}^{3\times3}$ and $\boldsymbol{d}$ denotes a vector of unknown but bounded external lumped uncertainties with $\|\boldsymbol{d}\| \le \bar{d}$, where $\|\cdot\|$ denotes the standard Euclidean norm.

The problem to be solved in this paper is stated as follows: with the control input $\boldsymbol{u}$, the attitude error states which include the error quaternion $\boldsymbol{q}_e$ and the error angular velocity $\boldsymbol{\omega}_e$, converge to zero in a predefined-time T_c.

3. Main results

Before proceeding to the main theorem, the following lemma is introduced to summarize the existing results on the predefined-time stability. Its proof can be taken directly from [18], thus omitted here for the sake of brevity.

Lemma 1. *Suppose there exists a candidate Lyapunov function $V(\boldsymbol{x}) : \mathbb{R}^n \to \mathbb{R}$ with $V(\boldsymbol{0}) = 0$, $V(\boldsymbol{x}) > 0(\forall\ \boldsymbol{x} \neq \boldsymbol{0})$ for an autonomous system $\dot{x} = f(x, t)$ with $x \in \mathbb{R}$ being the state. Then, it can be concluded that the underlying system is globally predefined-time stable w.r.t. T_c in the sense that $x(t) = 0$ for all $t \ge T_c$ and for all $x(t_0) \neq 0$, provided that*

$$\dot{V}(\boldsymbol{x}) \le -\frac{1}{pT_c}\exp\left[V^p(\boldsymbol{x})\right]V^{1-p}(\boldsymbol{x}),\ \forall\ \boldsymbol{x} \neq \boldsymbol{0} \tag{8}$$

where T_c is the prescribed time constant, and $0 < p < 1$.

Inspired by Lemma 1, the attitude tracking of a spacecraft can also be made predefined-time stable. More specifically, as will be demonstrated in the following theorem, the application of predefined-time stability theory to the underlying problem actually captures a tradeoff between stability and computational complexity.

Theorem 1. *Considering the error kinematics and dynamics of a spacecraft in Eq. (6), there is an attitude tracking control which can achieve predefined-time stability w.r.t. T_c in the sense that,*

$$\lim_{t \geq T_c} \boldsymbol{q}_e(t) = 0, \ \lim_{t \geq T_c} \boldsymbol{\omega}_e(t) = 0$$

provided that

$$\boldsymbol{u} = \boldsymbol{\omega}_e^{\times} \boldsymbol{J}_b \boldsymbol{\omega}_b + (\boldsymbol{C}_{bd}\boldsymbol{\omega}_d)^{\times} \boldsymbol{J}_b \boldsymbol{\omega}_b - \boldsymbol{J}_b \boldsymbol{\omega}_e^{\times} \boldsymbol{C}_{bd} \boldsymbol{\omega}_d - \boldsymbol{J}_b \dot{\boldsymbol{\alpha}} - \bar{d} \boldsymbol{\Lambda} \boldsymbol{e}_3 - \frac{1}{2p_2 T_{c_2}} \exp\left(V_2^{p_2}\right)\left(V_2^{-p_2}\right) \boldsymbol{J}_b \boldsymbol{\sigma} \quad (9)$$

with $0 < p_2 < 1$ and $\boldsymbol{e}_3$ denoting a vector of ones and $\boldsymbol{\alpha} = \frac{1}{p_1 T_{c_1}} \exp\left(V_1^{p_1}\right) V_1^{-p_1} \cdot E^{-1}\left(\boldsymbol{q}_{ev}\right)\boldsymbol{q}_e$, where $0 < p_1 < 1$ and $V_1 = \frac{1}{2}\boldsymbol{q}_e^{\mathrm{T}}\boldsymbol{q}_e$. Moreover, $\dot{\boldsymbol{\alpha}}$ denotes the time derivative of $\boldsymbol{\alpha}$, and $\boldsymbol{\sigma}$ represents:

$$\boldsymbol{\sigma} = \boldsymbol{\omega_e} + \boldsymbol{\alpha} \quad (10)$$

$\boldsymbol{\Lambda} \in \mathbb{R}^{3\times 3}$ is a diagonal matrix with the ith element being $\mathrm{sign}\left(\boldsymbol{\sigma}^{\mathrm{T}} \boldsymbol{J}_b^{-1}\right)_i$, $(i = 1, 2, 3)$. $V_2 = \frac{1}{2}\boldsymbol{\sigma}^{\mathrm{T}}\boldsymbol{\sigma}$, $T_{c_1} > 0$ and $T_{c_2} > 0$ are two predefined times satisfying $T_c = T_{c_1} + T_{c_2}$.

Proof. We start the proof by incorporating the sliding mode surface which is chosen as $\boldsymbol{\sigma}$. The first candidate Lyapunov function is chosen as V_2. Taking the derivative of V_2 with respect to time along the trajectories of system (7) yields:

$$\begin{aligned} \dot{V}_2 &= \boldsymbol{\sigma}^{\mathrm{T}} \dot{\boldsymbol{\sigma}} \\ &= \boldsymbol{\sigma}^{\mathrm{T}} \begin{bmatrix} \boldsymbol{\omega}_e^{\times} \boldsymbol{C}_{bd} \boldsymbol{\omega}_d - \boldsymbol{J}_b^{-1} (\boldsymbol{C}_{bd}\boldsymbol{\omega}_d)^{\times} \boldsymbol{J}_b \boldsymbol{\omega}_b \\ -\boldsymbol{J}_b^{-1} \boldsymbol{\omega}_e^{\times} \boldsymbol{J}_b \boldsymbol{\omega}_b + \boldsymbol{J}_b^{-1} \boldsymbol{u} + \boldsymbol{J}_b^{-1} \boldsymbol{d} + \dot{\boldsymbol{\alpha}} \end{bmatrix} \end{aligned} \quad (11)$$

Substituting Eq. (9) into the above relation yields

$$\dot{V}_2 \leq -\frac{1}{p_2 T_{c_2}} \exp\left(V_2^{p_2}\right) V_2^{1-p_2} \quad (12)$$

With Lemma 1, it can be concluded that $V_2 \to 0$ as $t > T_{c_2}$, implying that $\boldsymbol{\sigma} = \boldsymbol{0}$ is reached in the predefined T_{c_2}. Then, we have

$$\boldsymbol{\omega}_e = -\frac{1}{p_1 T_{c_1}} \exp\left(V_1^{p_1}\right) V_1^{-p_1} E^{-1}\left(\boldsymbol{q}_{ev}\right) \boldsymbol{q}_e \quad (13)$$

The second candidate Lyapunov function is proposed as V_1. When $t > T_{c_2}$, with Eqs. (7) and (13), the derivative of V_1 becomes

$$\begin{aligned} \dot{V}_1 &= \boldsymbol{q}_e^{\mathrm{T}} \dot{\boldsymbol{q}}_e \\ &= -\frac{1}{p_1 T_{c_1}} \exp\left(V_1^{p_1}\right) V_1^{1-p_1} \end{aligned} \quad (14)$$

According to Lemma 1, $\boldsymbol{q}_e \to \boldsymbol{0}$ in the predefined time T_{c_1}. Consequently, the attitude tracking system achieves stability with the proposed $\boldsymbol{u}$ in the predefined-time $T_c = T_{c_1} + T_{c_2}$, and this concludes the proof of Theorem 1. □

Remark 1. Note that albeit its effectiveness, the proposed controller (9) can not ensure that $\boldsymbol{q}_e$ or $\boldsymbol{\omega}_e$ converges to absolute zero within the predefined time simply due to the existence of uncertainties. Rather, they will converge to an arbitrarily small region of the equilibrium, respectively.

4. Numerical simulations

In this section, illustrative numerical examples are presented to demonstrate the performance of the proposed predefined-time stability control technique. The desired attitude reference frame is chosen as the spacecraft orbit coordinate. The desired attitude angular velocity is calculated as $\boldsymbol{\omega}_d = [0 \ \ -\|\boldsymbol{r}\times\boldsymbol{v}\|/\|\boldsymbol{r}\|^2 \ 0]^{\mathrm{T}}$ with $\boldsymbol{r}$ and $\boldsymbol{v}$ denoting the spacecraft position and velocity vectors with respect to $\mathfrak{R}_J$, respectively. The orbit recursion is referenced in [28] with the initial spacecraft position and velocity given as $\boldsymbol{r}_0 = [2295.7382 \ 5446.8229 \ 3521.8472]^{\mathrm{T}}$ km and $\boldsymbol{v}_0 = [2.5399 \ 3.1242 \ -6.4666]^{\mathrm{T}}$ km/s.

Moreover, the initial attitude quaternion and angular velocity are given as $\boldsymbol{q}_v = [0.1944 \ -0.2827 \ -0.7556 \ 0.5580]^{\mathrm{T}}$ and $\boldsymbol{\omega}_b = [-0.3194 \ \ -0.1780 \ \ 0.1728]$ rad/s. The spacecraft inertial matrix is

$$\boldsymbol{J}_b = \begin{bmatrix} 55.91 & 8.92 & 12.24 \\ 8.92 & 53.26 & 6.92 \\ 12.24 & 6.92 & 56.29 \end{bmatrix} \mathrm{kg}\cdot\mathrm{m}^2$$

The external disturbance is assumed to be

$$\boldsymbol{d} = 0.001\begin{bmatrix} 1+\sin(\pi t/125)+\cos(\pi t/200) \\ 2+\sin(\pi t/150)+\cos(\pi t/185) \\ -1+\sin(\pi t/225)+\cos(\pi t/175) \end{bmatrix} \mathrm{Nm}$$

In addition, the bound of the lumped external uncertainty is assumed to be $\bar{d} = 0.007$. To show the effectiveness of the proposed predefined-time stability controller, simulations with $T_c = 60$ and $T_c = 30$ are presented, respectively. Referring to [29–31], the largest attitude control torque of the spacecraft is chosen to be 5 Nm for $T_c = 60$ and 8 Nm for $T_c = 30$, respectively. To avoid the chattering phenomenon of the proposed controller, $\mathrm{sign}\left(\boldsymbol{\sigma}^{\mathrm{T}}\boldsymbol{J}_b^{-1}\right)_i$ in (9) is replaced by $\frac{(\sigma^{\mathrm{T}}\boldsymbol{J}_b^{-1})_i}{\|(\sigma^{\mathrm{T}}\boldsymbol{J}_b^{-1})_i\|+\epsilon}$ in the numerical simulations, where ϵ is a small positive constant decided by simulation accuracy.

When $T_c = 60$, the design parametes are set as $p_1 = 0.16$, $p_2 = 0.16$, $T_{c_1} = 30$ and $T_{c_2} = 30$. The simulation results with the proposed controller are presented in Figs. 1–3: with Fig. 1 and Fig. 2 showing the error quaternion and error angular velocity, respectively, and Fig/ 3 presenting the trajectory of the control torque.

It can be observed from Fig. 1 to Fig. 2 that the proposed controller can provide the system with predefined-time stability within 60 seconds.

When $T_c = 30$, the design parameters are set as $p_1 = 0.17$, $p_2 = 0.17$, $T_{c_1} = 15$ and $T_{c_2} = 15$. The simulation results are presented in Figs. 4–6.

Figs. 4–5 demonstrate that the system achieves practical predefined-time stability within 30 seconds. Therefore, the effectiveness of the proposed controller in terms of predefined-time stability is demonstrated by the simulations above. Moreover, it can be observed from Fig. 3 and Fig. 6 that a larger maximum control torque is required when the prescribed time is set to be smaller, which reflects the fact that the faster a spacecraft tracks its desired attitude, the more energy it needs.

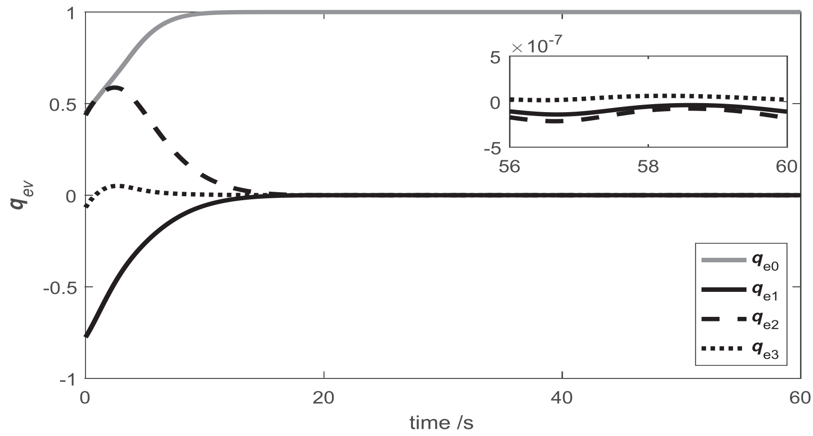

Fig. 1. Trajectories of the error quaternion.

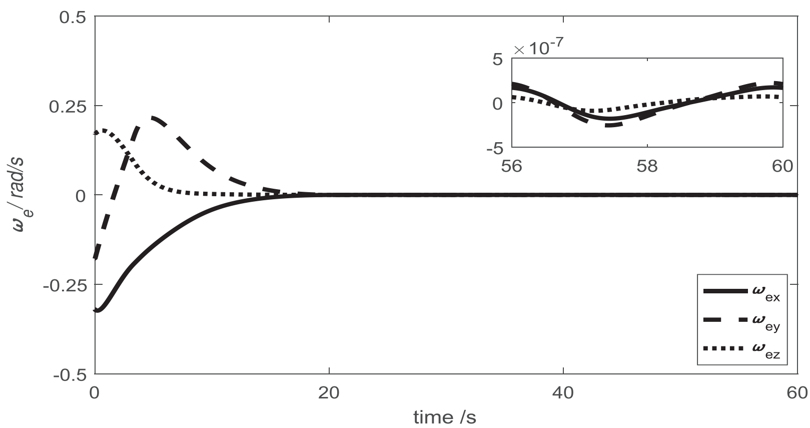

Fig. 2. Trajectories of the error angular velocity.

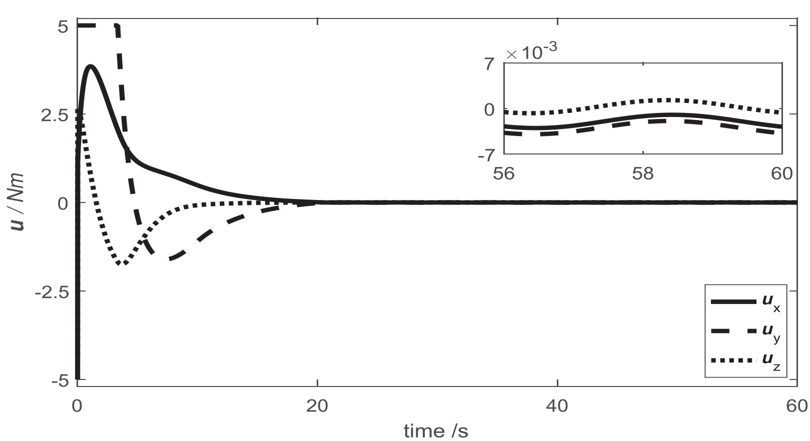

Fig. 3. Trajectories of the control torque.

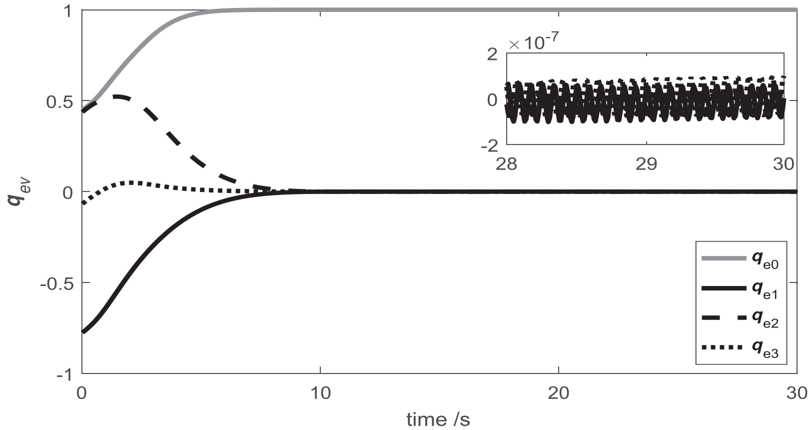

Fig. 4. Trajectories of the error quaternion.

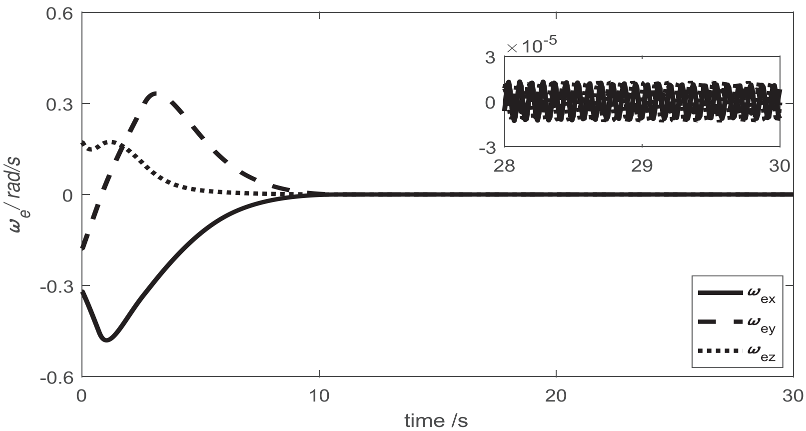

Fig. 5. Trajectories of the error angular velocity.

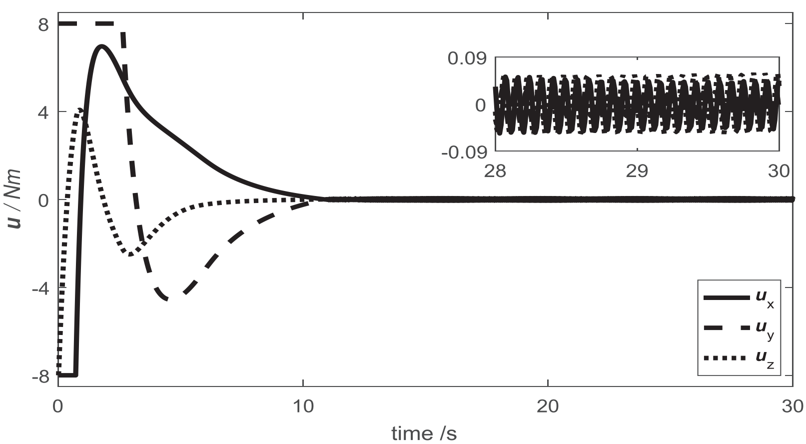

Fig. 6. Trajectories of the control torque.

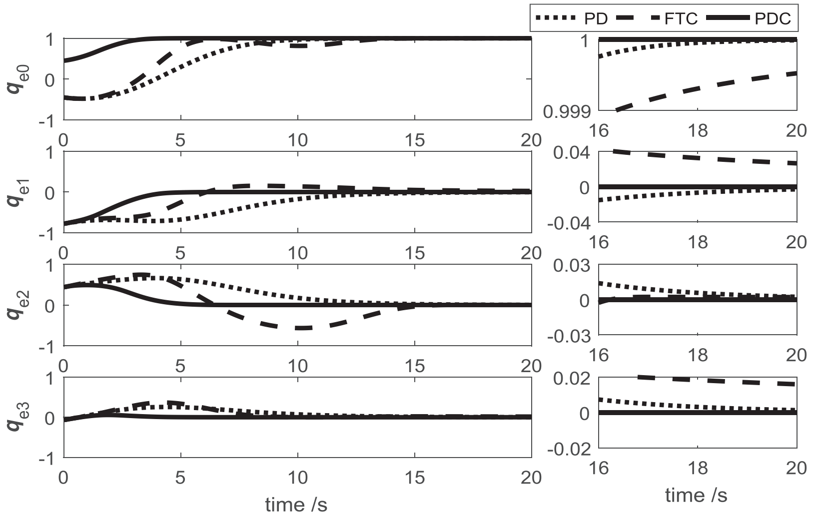

Fig. 7. Comparison of the error quaternion.

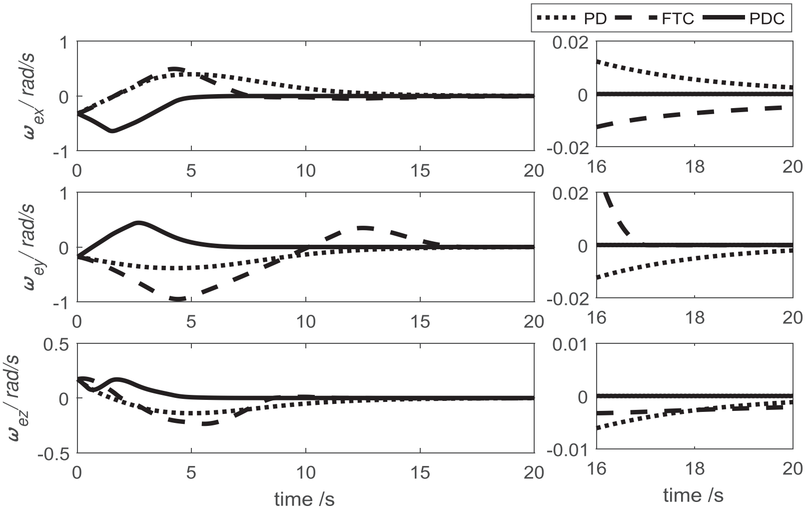

Fig. 8. Comparison of the error attitude angular velocity.

In addition, the comparisons between the proposed predefined-time controller (denoted as PDC) and two existing controllers, the traditional PD controller in [6] (denoted as PD) and the finite-time controller in [14] (denoted as FTC), are conducted with $T_c = 20$. Numerical simulation results are presented in Figs. 7–9.

From Figs. 7 to 9, it can be observed that with similar control torque requirements, the system is practically stable within 20 s, which is faster than that with PD or FTC. This comparison demonstrates that the proposed control scheme has advantage in prescribed-time convergence over the two existing control schemes.

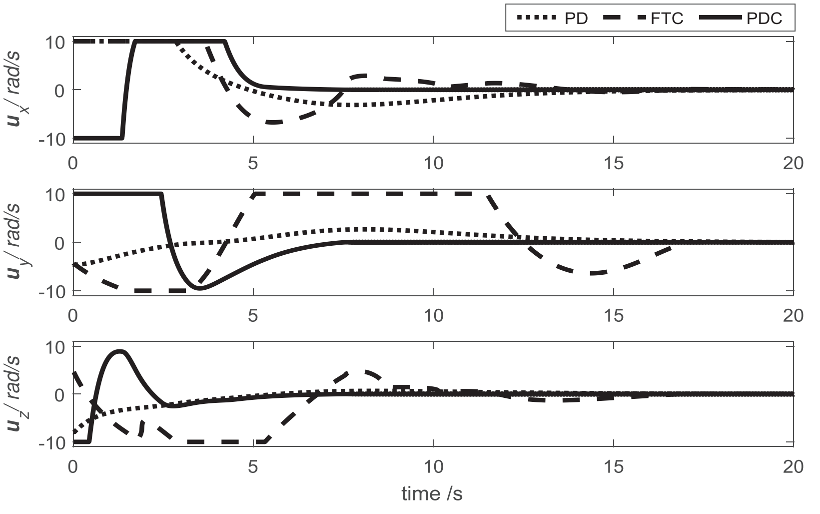

Fig. 9. Comparison of the control torque.

5. Conclusion

In this paper, the attitude tracking problem of a spacecraft has been solved with guaranteed performance. We have proved rigorously that the predefined-time stability is feasible and resilient for the underlying problem, and the salient feature of the proposed strategy is the settling time of the overall system can be prescribed. Future work on this venue would focus on how to enforce the predefined time stability in attitude synchronization of a group of spacecraft.

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