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Constrained dual-loop attitude control design for spacecraft

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ABSTRACT

A constrained dual-loop controller is designed for the attitude control of rigid spacecraft with inertia uncertainties, assigned angular velocity and control constraints. The design is based on a modular approach that the design of the control law and the online estimator leads to an important "separation property". The control law is designed based on a two-loop structure in which the outer-loop control law implements by sliding mode control technique to stabilize the attitude under angular velocity constraints, and the inner loop control law is designed to track the limited command of outer loop under uncertainties and control input constraints. To improve the robustness of the attitude controller, an extended state observer is developed for online estimation of the lumped disturbance. The constraints on control input and angular velocity are implemented by second-order command filter. The stability of the subsystem during saturation is guaranteed by using a modified tracking error definition, in which the effect of the constraints has been filtered and removed. Simulation results are obtained to illustrate the effectiveness of the proposed attitude controller.

1. Introduction

The problem of attitude control of a rigid spacecraft has received considerable attention and is one of the most widely studied application topics in controls literature during past decade [1]. Most of the existing studies have ignored the saturation problem and assume that all the control torque of the spacecraft system is sufficient for control and without restriction on angular velocity (see [2–5]). However, due to the limitation, all the actuators have amplitude limitation and angular velocity should keep in the range of operation, for example, the reaction wheels of spacecraft can only provide a limited torques, the rate gyros have a limited measurement range. If the saturation problem is ignored in the controller design, it will cause damage to the system, and even make the system unstable. Taking the nonlinear saturation problem into account for spacecraft poses a huge challenge for attitude controller system designers.

The saturation problem for attitude control has attracted considerable interest in the existing literature. In practice, input saturation constraint is one of the most important saturation problems [6]. To eliminate the effect of input saturation, a feed-forward compensator is used to compensate for the nonlinear term arising from the input saturation in [7], which requires that the nonlinear term must be linearly parameterized. To relax the above restriction, a command filtered approach is presented in [8,9], and two saturated proportional-derivative attitude controllers are also proposed for rigid spacecraft with actuator constraints in [10,11]. However, the aforementioned control approaches did not consider the angular velocity saturation. In [12], the angular velocity constraints problem is taken into account, and a Barrier Lyapunov function is employed to prevent angular velocity from violating the rate constraint, but the control input saturation cannot be achieved due to the design of Barrier Lyapunov function (refer to Remark 3 in [13]). In [14], a novel robust nonlinear controller, which combines Barrier Lyapunov function and programming algorithms, is proposed for attitude stabilization of a rigid spacecraft under the constraints of assigned velocity and actuator torques, however, the actual control torque is continuous in simulation, not all the actuator torques reach the limitation. In [15], the nonlinear quaternion feedback control logic is proposed for the spacecraft eigenaxis rotations within the saturation limits of actuators and rate gyros, which not incorporate constraints on individual angular velocity but the minimum slow rate. Considering the constraints on individual angular velocity and control input, a constrained optimal PID (Proportional-Integral-Derivative)-like controller is presented in [16], which needs optimal control software MISER 3.3 to acquire the optimal parameters. Moreover, the control schemes in [15,16] require the nominal off-diagonal terms of the inertia matrix to be zeros, and none of them consider the model uncertainties. In order to solve the problem of constraints on the control surfaces and the virtual control states, a backstepping controller with command filtered approach is proposed for nonlinear flight control in [17–19],

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command filters are used to implement mechanical or operating constraints on the control surfaces and the virtual control states. This controller is designed to ensure the asymptotic convergence of the modified error system; it does not guarantee the system states in the range of constraints. As a result of the recursive design, correlation between the rings, it will cause fluctuations in the states and the control torque when the bandwidth of the command filters below a certain value.

In practical situations, the spacecraft system subjects to various uncertainties such as inertia uncertainties, external disturbances and on-board payload motion. Moreover, the upper bound of disturbance is difficult to obtain in practical situations [20]. To improve the disturbance rejection ability of the closed-loop system, feed-forward compensation control is one of the most effective disturbance rejection methods, which requires the measurement of the disturbance [21]. Adaptive radial basis function neural networks can be used to approximate unknown nonlinearities, however, it is not reasonable to require the estimate value converge to any particular equilibrium, but merely that it remains bounded [22]. The extended state observer (ESO) is a senseless process monitoring technique and also the most important part of the active disturbance rejection control [23,24], if the change rate of the lumped disturbance is negligible in comparison with disturbance estimation error dynamics, the disturbance estimation error will converge asymptotically to zero.

This study is motivated by constrained adaptive backstepping method [19], and seeks to reduce the influence of correlation between the rings and keeps the states within the limitations. In this paper, based on the previous work, a constrained dual-loop control scheme, which combines command filter and extended state observer, is proposed for rigid spacecraft attitude control. The objective of the control law design is to accomplish the attitude rotational motion within the limitations and parameters uncertainties. The main contributions of this study relative to other studies are as: (1) a constrained dual-loop attitude control scheme is proposed to accomplish the attitude rotational motion for rigid spacecraft under the actuator limitations and angular velocity constraints. In comparison with the aforementioned studies associated with actuator saturation and bound angular velocity, the proposed controller allows the angular velocity to reach the maximum value of the limitation, and relaxes the above restriction on the inertia matrix. Moreover, the proposed controller has a simple structure and is robust to disturbance. (2) Compared with the constrained backstepping control method [17–19], the constrained dual-loop control design eliminates the influence of the attitude error on the inner control design, which avoids the angular velocity exceeds the limitation when the initial attitude error is large. Moreover, the effect of control saturation on outer loop is eliminated, which improves the robustness of the closed-loop system to actuator bandwidths.

The rest of this paper is organized as follows. In Section 2, the attitude dynamics of the spacecraft and control problems formulation are summarized. In the presence of limitations, a constrained dual-loop attitude tracking control law based on extend state observer is derived in Section 3. The results of numerical simulation in Section 4 demonstrate the performance of the proposed control scheme. In Section 5, we give the summary of this paper.

2. Spacecraft attitude dynamics and control problems formulation

2.1. The dynamic equation

The spacecraft is modeled as a rigid body with reaction wheel as the control actuator; consider the inertia of the reaction wheel is far smaller than the spacecraft; the attitude dynamic model is governed by the following equation [25]

$$\boldsymbol{J}\dot{\boldsymbol{\omega}} = -\boldsymbol{\omega}^{\times}\boldsymbol{J}\boldsymbol{\omega} + \boldsymbol{u} + \boldsymbol{T}_d \tag{1}$$

where $\boldsymbol{J} \in R^{3\times 3}$ represents the positive definite inertia matrix of the spacecraft, $\boldsymbol{\omega} \in R^3$ denotes the angular velocity in a body-fixed reference frame, $\boldsymbol{u} \in R^3$ is the control input vector, $\boldsymbol{T}_d \in R^3$ is the disturbance torque. The Euler parameter kinematic differential equations are written as [1,4]

$$\dot{\bar{\boldsymbol{q}}}(t) = \boldsymbol{E}\big(\bar{\boldsymbol{q}}(t)\big)\boldsymbol{\omega}; \qquad \boldsymbol{E}\big(\bar{\boldsymbol{q}}(t)\big) = \frac{1}{2}\begin{bmatrix} -\boldsymbol{q}^{\mathrm{T}} \\ (\boldsymbol{q}^{\times} + q_0\boldsymbol{I}_3) \end{bmatrix} \tag{2}$$

where $\bar{\boldsymbol{q}}(t) = (q_0(t), \boldsymbol{q}(t))$ is the four-dimensional unit-norm constrained quaternion, which represents the attitude of body-fixed frame with respect to the inertial frame, $\boldsymbol{I}_n \in R^{n\times n}$ denotes the identity matrix, and for $\boldsymbol{x} = [\,x_1 \quad x_2 \quad x_3\,]^{\mathrm{T}} \in R^3$, the matrix operator $\boldsymbol{x}^{\times}$ denotes the skew-symmetric matrix that is equivalent to the cross product operation between any other vector as follows

$$\boldsymbol{x}^{\times}\boldsymbol{y} = \boldsymbol{x} \times \boldsymbol{y} \quad \forall \boldsymbol{y} \in R^3 \tag{3}$$

Further, let $\boldsymbol{C}(\bar{\boldsymbol{q}})$ be the corresponding direction cosine matrix in terms of the quaternion $\bar{\boldsymbol{q}}$ expressed as

$$\boldsymbol{C}(\bar{\boldsymbol{q}}) = \big(q_0^2 - \boldsymbol{q}^{\mathrm{T}}\boldsymbol{q}\big)\boldsymbol{I}_3 + 2\boldsymbol{q}\boldsymbol{q}^{\mathrm{T}} - 2q_0\boldsymbol{q}^{\times} \tag{4}$$

Let $\boldsymbol{\omega}_r \in R^3$ and $\bar{\boldsymbol{q}}_r$ denote the bounded and smooth reference angular velocity vector and reference quaternion states defined in reference frame, respectively. Then the attitude and angular velocity errors can be defined as follows:

$$\boldsymbol{C}(\bar{\boldsymbol{q}}_e) = \boldsymbol{C}(\bar{\boldsymbol{q}})\boldsymbol{C}^{\mathrm{T}}(\bar{\boldsymbol{q}}_r); \qquad \delta\boldsymbol{\omega} = \boldsymbol{\omega} - \boldsymbol{C}(\delta\bar{\boldsymbol{q}})\boldsymbol{\omega}_r \tag{5}$$

Using Eqs. (1), (2), and (5), along with a few identities, leads to the attitude error dynamics and kinematics [26]:

$$\begin{aligned} &\dot{\bar{\boldsymbol{q}}}_e = \boldsymbol{E}\big(\bar{\boldsymbol{q}}_e(t)\big)\delta\boldsymbol{\omega} \\ &\boldsymbol{J}\delta\dot{\boldsymbol{\omega}} = -\boldsymbol{\omega}^{\times}\boldsymbol{J}\boldsymbol{\omega} + \boldsymbol{T}_d + \boldsymbol{u} + \boldsymbol{J}\boldsymbol{\eta} \end{aligned} \tag{6}$$

where $\boldsymbol{\eta}$ is defined for notational convenience to represent the following

$$\boldsymbol{\eta} = \delta\boldsymbol{\omega}^{\times}\boldsymbol{C}(\bar{\boldsymbol{q}}_e)\boldsymbol{\omega}_r - \boldsymbol{C}(\bar{\boldsymbol{q}}_e)\dot{\boldsymbol{\omega}}_r \tag{7}$$

2.2. Control problem statements

The spacecraft inertia can be written as the sum of both nominal part and uncertainties part, i.e., $\boldsymbol{J} = \boldsymbol{J}_0 + \boldsymbol{\Delta J}$. Then the error dynamic of the rigid spacecraft can be simplified as

$$\begin{aligned} &\dot{\boldsymbol{q}}_e = \boldsymbol{G}_1\delta\boldsymbol{\omega} \\ &\delta\dot{\boldsymbol{\omega}} = \boldsymbol{F} + \boldsymbol{G}_2\boldsymbol{u} + \boldsymbol{d} \end{aligned} \tag{8}$$

where $\boldsymbol{d}$ is the lumped disturbance termed and

$$\begin{aligned} &\boldsymbol{G}_1 = \frac{1}{2}\big(q_{e0}\boldsymbol{I}_3 + \boldsymbol{q}_e^{\times}\big), \qquad \boldsymbol{F} = -\boldsymbol{G}_2\boldsymbol{\omega}^{\times}\boldsymbol{J}_0\boldsymbol{\omega} + \boldsymbol{\eta} \\ &\boldsymbol{G}_2 = \boldsymbol{J}_0^{-1}, \qquad \boldsymbol{d} = -\boldsymbol{G}_2\big\{\boldsymbol{\omega}^{\times}\Delta\boldsymbol{J}\boldsymbol{\omega} + \Delta\boldsymbol{J}(\dot{\boldsymbol{\omega}} - \boldsymbol{\eta}) - \boldsymbol{T}_d\big\} \end{aligned}$$

Furthermore, the constraints on the control input $\boldsymbol{u}(t)$ and the angular velocity $\boldsymbol{\omega}(t)$ are given by

$$\begin{aligned} &\boldsymbol{u} \in D_{\boldsymbol{u}} = \big\{\boldsymbol{u} : \big|u_i(t)\big| \le u_{\max}, \quad i = 1,2,3\big\} \\ &\boldsymbol{\omega} \in D_{\boldsymbol{\omega}} = \big\{\boldsymbol{\omega} : \big|\omega_i(t)\big| \le \omega_{\max}, \quad i = 1,2,3\big\} \end{aligned} \tag{9}$$

where $u_{\max}$ and $\omega_{\max}$ are the maximum torque value of reaction wheels and the limited operate angular velocity, respectively.

Assumption 1. The rate of the lumped disturbance is bounded but unknown.