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Global stabilization of the linearized three-axis axisymmetric spacecraft attitude control system by bounded linear feedback ☆

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ABSTRACT

In this paper, the three-axis attitude stabilization of the axisymmetric spacecraft with bounded inputs is studied. By constructing some novel state transformations, saturated linear state feedback controllers are constructed for the considered attitude control system. By constructing suitable quadratic plus integral Lyapunov functions, globally asymptotic stability of the closed-loop systems is proved if the feedback gain parameters satisfy some explicit conditions. By solving some min–max optimization problems, a global optimal feedback gain for the underactuated attitude stabilization system is proposed such that the convergence rate of the linearized closed-loop system is maximized. Numerical simulations show the effectiveness of the proposed approaches.

1. Introduction

Attitude control systems play fundamental roles in the operation of satellite mission since they constitute a mandatory characteristic for both the survival of satellites and the satisfactory achievement of mission goals. Typical attitude stabilization systems include passive attitude stabilized systems [5,15,35] and active three-axis controlled systems [13,20,32,36,39]. It has been realized that the most accurate designs normally include momentum wheels and/or reaction wheels because they allow continuous and smooth control with the lowest possible parasitic disturbing torques [23].

The attitude stabilization problem of axisymmetric spacecraft has been investigated in the literature in the past years (see, [1, 27,28,41] and the references therein). The angular velocity equations of an axisymmetric spacecraft were globally asymptotically stabilized in [1] by means of a linear feedback when two control torques act on the body. Optimal control laws for axisymmetric spacecraft with restrictions on initial velocities were presented in [28]. Nonlinear H_∞ control designs with axisymmetric spacecraft control were proposed in [41]. It is noted that the gravity gradient moment and/or actuator saturation were not considered in the aforementioned references.

Saturation nonlinearity exists in every practical control system and makes the overall system inherently nonlinear (see [12,14, 17,31,38,40]). Take the spacecraft attitude control system for example. Typical actuators (such as magnetorquers, reaction wheels, or control moment gyros) are subject to saturation due to the physical limitation and energy consumption. Therefore, conventional attitude control schemes may result in control signals beyond the saturation level which can lead to serious differences between the commanded input signal and the actual control effort, and is thus a source of performance degradation or, even worse, instability of the closed-loop system. Therefore, the problem of spacecraft attitude control with actuator saturation has been addressed in the literature [6–8,11,19,21,29,39]. For example, two simple PD controllers were proposed in [24] to address the global asymptotic regulation of rigid spacecraft subject to actuator saturation, the problem of satellite attitude control with actuator saturation was addressed in [12,6,8], a novel observer-controller control scheme was proposed in [12] to solve the output feedback attitude control of a rigid body with bounded input, a simple nonlinear proportional-derivative-type (PD-type) saturated finite-time controller was designed in [6], the attitude tracking of a rigid body by using a quaternion description and global finite time attitude controllers were designed with three types of measurements in [8], discontinuous feedback controllers were designed in [29] to stabilize the attitude of an axisymmetric spacecraft with bounded control, and explicit saturated linear feedback controllers for attitude stabilization were constructed in [39] and global stability for the linearized system was guaranteed.

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Due to the scarce onboard resources and the complex working environment, particularly, for CubeSat-class nanosatellites, the attitude stabilizing controllers must be as simple as possible. Then linear feedback should be the best choice. Noting that in general a linear global stabilizing controller exists if the open-loop system is neutral stable, which is not always satisfied in practice. It is thus challenging to design linear globally stabilizing controllers for the axisymmetric spacecraft since the corresponding open-loop system is not Lyapunov stable [39].

Motivated by the existing results on this topic, in this paper we consider the problem of three-axis attitude control of axisymmetric spacecraft with bounded inputs. By taking the gravity-gradient torques effect into account, a novel state variable transformation on the linearized attitude equation is used to construct saturated linear state feedback controllers for the considered attitude control system. It is shown by constructing explicit Lyapunov function that the proposed controllers guarantee the global stability of the linearized closed-loop systems when the parameters in the feedback gain satisfy some explicit conditions. The optimal linear feedback gain (such that the convergence rate of the linearized closed-loop system is maximized) for the attitude control system is also obtained by studying some min–max optimization problems. We mention that the proposed control approaches are also applicable to nanosatellites since the proposed linear controllers are easy to implement and robust with respect to uncertainties. Numerical simulations show the effectiveness of the proposed approaches.

The remainder of this paper is organized as follows. In Section 2, the model of the attitude control system is introduced. The main results regarding global stabilization of the roll-yaw loop and the pitch loop are then respectively proposed in Sections 3 and 4. A numerical simulation is given to demonstrate the effectiveness of the proposed control law in Section 5. Finally, Section 6 concludes the paper.

2. Model of the attitude control

The attitude motion of a rigid spacecraft can be described in the following reference frames [33,34] (see Fig. 1):

1. Geocentric Equatorial Frame $F_{\rm i}$, where the X axis points in the vernal equinox direction, the X–Y plane is the Earth's equatorial plane, and the Z axis coincides with the Earth's axis of rotation and points northward.
2. Orbital Frame $F_{\rm o}$, where the origin at the center of mass of the spacecraft, $x_{\rm o}$ being along the orbit direction, $y_{\rm o}$ being perpendicular to the orbit plane and $z_{\rm o}$ being in the nadir direction.
3. Spacecraft-fixed Body Frame $F_{\rm b}$, where the origin at the center of mass of the spacecraft.

It follows that if the attitude of the spacecraft is the identity, the body coordinates $x_{\rm b}$–$y_{\rm b}$–$z_{\rm b}$ coincide exactly with the orbital coordinates $x_{\rm o}$–$y_{\rm o}$–$z_{\rm o}$ [23].

The attitude matrix and attitude kinematics and dynamics can be described respectively as [33]

$$C = \left(q_4^2 - q_v^{\rm T} q_v\right) I_3 + 2 q_v q_v^{\rm T} - 2 q_4 q_v^{\times} \triangleq \begin{bmatrix} c_x & c_y & c_z \end{bmatrix},$$

$$\dot{q} = \frac{1}{2}\begin{bmatrix} 0 & \omega_{rz} & -\omega_{ry} & \omega_{rx} \\ -\omega_{rz} & 0 & \omega_{rx} & \omega_{ry} \\ \omega_{ry} & -\omega_{rx} & 0 & \omega_{rz} \\ -\omega_{rx} & -\omega_{ry} & -\omega_{rz} & 0 \end{bmatrix} q, \tag{1}$$

and

$$J\dot{\omega} + \omega \times J\omega = T_g + T_c, \tag{2}$$

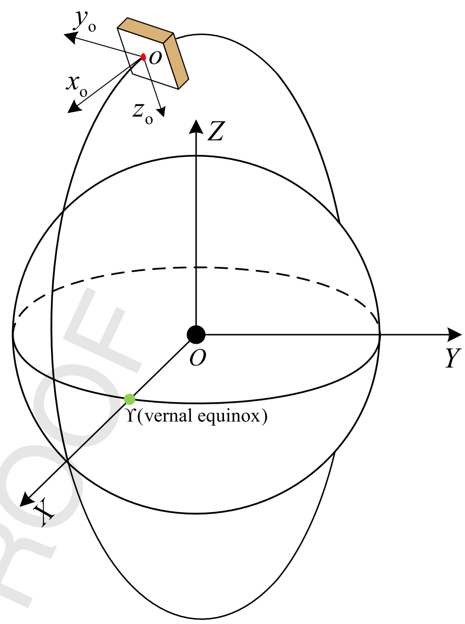

Fig. 1. The geocentric equatorial frame X–Y–Z and the orbital reference frame $x_{\rm o}$–$y_{\rm o}$–$z_{\rm o}$.

where I_3 is the 3×3 identity matrix, $q = [q_v^{\rm T}, q_4]^{\rm T}$ is the attitude quaternion, $q_v = [q_1, q_2, q_3]^{\rm T}$ is its vector part, $\omega_r = \left[\omega_{rx}, \omega_{ry}, \omega_{rz}\right]^{\rm T}$ is the (relative) angular velocity of the body frame $F_{\rm b}$ relative to the orbital frame $F_{\rm o}$, [33,34]

$$T_g = \left[T_{gx}, T_{gy}, T_{gz}\right]^{\rm T} = 3\omega_0^2 c_z \times J c_z$$

is the gravity-gradient torque vector, in which ω_0 is the orbital rate, the vector $T_c = \left[T_x, T_y, T_z\right]^{\rm T}$ denotes the control torque, $J = {\rm diag}\left\{J_x, J_y, J_z\right\}$ is the inertia matrix of the spacecraft, and $\omega = \left[\omega_x, \omega_y, \omega_z\right]^{\rm T}$ is the angular velocity of the body frame $F_{\rm b}$ relative to the geocentric inertial frame $F_{\rm i}$, and $q_v^{\times}$ is the corresponding cross-product operation defined as [33,34]

$$q_v^{\times} = \begin{bmatrix} 0 & -q_3 & q_2 \\ q_3 & 0 & -q_1 \\ -q_2 & q_1 & 0 \end{bmatrix}.$$

As mentioned in Section 1, this paper focuses on the three-axis attitude control for an axisymmetric spacecraft in a circular orbit. We thus assume that the inertia matrix is symmetric and the axis of symmetric is the minor principal axis, namely,

$$J_x = J_y > J_z. \tag{3}$$

On the other hand, the control input of the attitude control system is subject to saturation, namely,

$$|T_{ci}| \le \varpi_i, \; i = x, y, z,$$

in which $\varpi_i > 0, i \in \{x, y, z\}$, denote the maximal allowable value of controls in the i-axis.

Linearizing the attitude equations (1) and (2) including the gravity-gradient torque at the equilibrium $q^* = [0, 0, 0, 1]^{\rm T}$ and $\omega^* = [0, -\omega_0, 0]^{\rm T}$ gives [33,34]

$$\begin{bmatrix} \dot{q}_1 \\ \dot{q}_2 \\ \dot{q}_3 \end{bmatrix} = \frac{1}{2}\begin{bmatrix} \omega_x + 2\omega_0 q_3 \\ \omega_y + \omega_0 \\ \omega_z - 2\omega_0 q_1 \end{bmatrix},$$

$$\begin{bmatrix} \dot{\omega}_x \\ \dot{\omega}_y \\ \dot{\omega}_z \end{bmatrix} = \begin{bmatrix} -8\omega_0^2 q_1 \sigma_1 - 2\omega_0 \sigma_1 \dot{q}_3 \\ -6\omega_0^2 \sigma_1 q_2 \\ 0 \end{bmatrix} + J^{-1} {\rm sat}_{\varpi}\left(T_c\right),$$

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