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Research on Simulation Technology for Rapid Space Maneuver Control of Agile Satellites

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Abstract. To tackle the rapid maneuvering control issue of agile satellites with significant attitude changes in space, a satellite attitude dynamics and kinematics model is formulated using Euler quaternions. Drawing upon modern control theory, the control law is devised through the application of error quaternion feedback and PD control, thereby equipping the controller with anti-singularity capabilities and enabling swift control of large attitude maneuvers for agile satellites. Modeling and simulation are conducted using Simulink software, and the results demonstrate that the developed simulation model can faithfully mirror the satellite's in-orbit attitude variations, fulfilling the demands of rapid maneuvering control for large attitude changes.

Keywords: :Agile satellite, PD control, Attitude quaternion, simulation model

1. Introduction

The concept of agile satellites emerged in the early 21st century, with the main purpose of promoting fast multitasking through high-precision attitude control. Compared with traditional satellites, agile satellites have significantly improved maneuverability, with single attitude adjustment time reduced from a few minutes to a few seconds[1-2]. Recently, the progress of micro/nano satellite technology has promoted the miniaturization and intelligence of agile satellites, enabling some models to carry out autonomous mission planning and airborne edge computing[3]. In military reconnaissance and disaster monitoring scenarios, agile satellites significantly improve data collection efficiency through rapid follow-up and multi angle imaging.

The attitude control of agile satellites mainly includes the following key technologies[4-6]: (1) Hybrid actuator design. To strike a balance between the need for large-angle maneuverability and precise stability control, hybrid actuators have emerged as the predominant solution. A typical combination comprises a Control Moment Gyroscope (CMG) and a Reaction Flywheel, with the CMG providing high-torque maneuverability and the Flywheel tasked with fine-tuning the attitude. (2) Time-Optimal Control (TOC) algorithm. The TOC algorithm minimizes maneuver duration through multi-phase optimization. Simulations reveal that in a 90° attitude adjustment task, the TOC algorithm saves approximately 30% of the time and reduces energy consumption compared to traditional PID control. (3) Variable structure control approach. Addressing nonlinear dynamic characteristics, the variable structure controller achieves robust control through multi-mode

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switching. For instance, sliding mode control is employed to suppress disturbances during rapid maneuvering phases, whereas it switches to a Linear Quadratic Regulator (LQR) during stable attitude phases to enhance accuracy.

This article focuses on the problem of rapid maneuvering of agile satellites with large attitude in space. A mathematical model of the satellite attitude control system is established, and the current mainstream PD control algorithm is used for control simulation to verify the accuracy of the model algorithm.

2. Satellite attitude description and mathematical model

The composition of the satellite attitude control system is shown in Figure 1, mainly including attitude sensors, controllers, and actuators. The attitude sensor obtains the true attitude data of the satellite through sensor measurement. Based on the relevant data, the controller uses attitude determination algorithm to determine the satellite attitude and gives control instructions according to the control rate. The actuator outputs control torque according to the control instructions given by the controller [7].

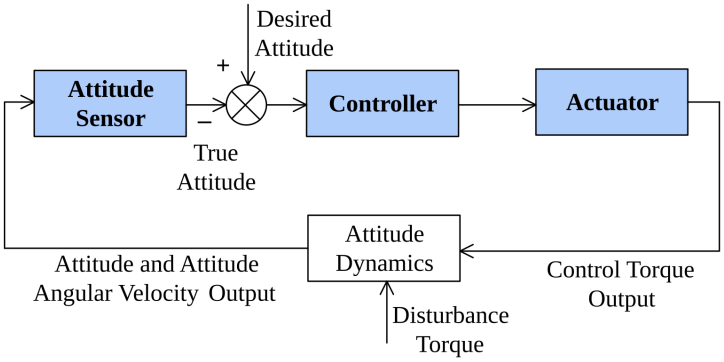

Figure 1: Satellite attitude control system composition

2.1. Coordinate system definition and transformation

The attitude maneuver of a satellite in space involves multiple different types of coordinate systems, such as the inertial coordinate system, orbital coordinate system, and satellite body coordinate system, as shown in Figure 2. The choice of coordinate system directly affects the equations and complexity of the satellite's mathematical model. Therefore, selecting an appropriate coordinate system is crucial for the study of the problem[8].

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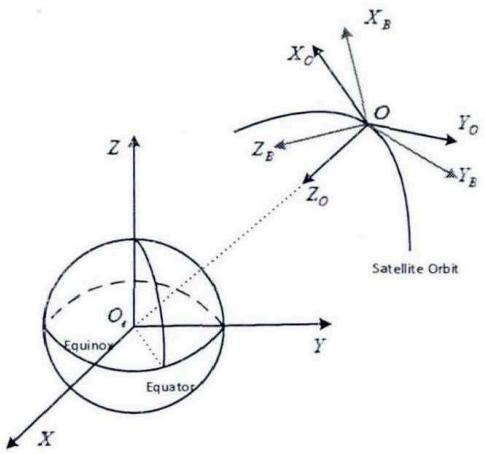

Figure 2: Satellite coordinate system diagram

2.1.1. Inertial coordinate system (i-frame)

It is a coordinate system that is stationary relative to inertial space. The origin is at the Earth's center, the Z_i-axis points to the northern celestial pole and coincides with the Earth's rotation axis, the X_i-axis lies in the Earth's equatorial plane and points towards the vernal equinox, and the Y_i-axis forms a right-hand rotating coordinate system with the X_i- and Z_i-axes within the equatorial plane.

2.1.2. Satellite orbital coordinate system (o-frame)

The origin of the orbital coordinate system is at the satellite's center of mass, and the satellite's orbital plane serves as the coordinate plane. The Z_o-axis points from the center of mass towards the Earth's center, the X_o-axis lies in the orbital plane, is perpendicular to the Z_o-axis, and points in the direction of the satellite's flight. The Y_o-axis, along with the X_o- and Z_o-axes, forms a right-hand orthogonal coordinate system and aligns with the negative normal to the orbital plane. This coordinate system is used as the nominal attitude reference for Earth-pointing control.

2.1.3. Satellite body coordinate system (b-frame)

It is a body-fixed coordinate system attached to the main body of the satellite. The origin of the satellite body coordinate system is at the satellite's center of mass, and the three coordinate axes align with the satellite's principal axes of inertia.

In the study of satellite attitude, the body coordinate system is usually chosen as the reference, with the coordinate origin typically taken as the satellite's center of mass. The ox-axis is the roll axis of the satellite, pointing in the direction of the satellite's motion, the oy-axis is the pitch axis, perpendicular to the ox-axis, and the oz-axis is the yaw axis, pointing towards the Earth's center. The three axes satisfy the right-hand rule.

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2.2. Satellite attitude description methods

The methods for describing satellite attitude include the Euler axis method and the quaternion method. The Euler method has intuitive physical concepts and clear geometric meaning, but it suffers from singularity issues when dealing with large angles. In the problem studied in this paper, where rapid maneuvering of a quick-response satellite in space involves large angles, the quaternion method is chosen to describe the satellite's attitude motion[9-10].

According to Euler's theorem, any angular displacement of a rigid body around a fixed point can be achieved by rotating around a certain axis passing through that point by a certain angle. That is, the rotation of a unit vector r to a new unit vector b can be considered as being accomplished by rotating around the axis e in space by an angle φ. Conventionally, the Euler quaternion is denoted as q, and it is defined as follows:

$$\mathrm{q}=\begin{pmatrix} q_0 \\ Q \end{pmatrix}=[q_0 \quad q_1 \quad q_2 \quad q_3]^{\mathrm{T}}=[e_x \cos\frac{\Phi}{2} \quad e_y \sin\frac{\Phi}{2} \quad e_z \sin\frac{\Phi}{2} \quad \cos\frac{\Phi}{2}] \tag{1}$$

Where $e = [e_x, e_y, e_z]^T$ is the unit vector of the coordinate axis, Φ represents the rotation angle, q_0 is the scalar part of the quaternion, and Q is the vector part of the quaternion. The four parameters satisfy the following relationship:

$$q_0^{\,2}+q_1^{\,2}+q_2^{\,2}+q_3^{\,2}=1 \tag{2}$$

For a rotation sequence of 3-1-2 with rotation angles ψ, δ, and θ, the conversion relationship between the attitude quaternion and the Euler angles is as follows:

$$\begin{bmatrix} q_0 \\ q_1 \\ q_2 \\ q_3 \end{bmatrix}=\begin{bmatrix} \sin\frac{\delta}{2}\cos\frac{\theta}{2}\cos\frac{\psi}{2}-\cos\frac{\delta}{2}\sin\frac{\theta}{2}\sin\frac{\psi}{2} \\ \sin\frac{\delta}{2}\cos\frac{\theta}{2}\sin\frac{\psi}{2}+\cos\frac{\delta}{2}\sin\frac{\theta}{2}\cos\frac{\psi}{2} \\ \cos\frac{\delta}{2}\cos\frac{\theta}{2}\sin\frac{\psi}{2}+\sin\frac{\delta}{2}\sin\frac{\theta}{2}\cos\frac{\psi}{2} \\ \cos\frac{\delta}{2}\cos\frac{\theta}{2}\cos\frac{\psi}{2}-\sin\frac{\delta}{2}\sin\frac{\theta}{2}\sin\frac{\psi}{2} \end{bmatrix} \tag{3}$$

Based on the definition of quaternion multiplication, the kinematic equation of satellite attitude represented by quaternions can be expressed as:

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$$\dot{q} = \frac{1}{2} q \otimes \begin{pmatrix} 0 \\ \omega \end{pmatrix} = \begin{pmatrix} 0 & -\omega_x & -\omega_y & -\omega_z \\ \omega_x & 0 & \omega_z & -\omega_y \\ \omega_y & -\omega_z & 0 & \omega_x \\ \omega_z & \omega_y & -\omega_x & 0 \end{pmatrix} \begin{pmatrix} q_0 \\ q_1 \\ q_2 \\ q_3 \end{pmatrix} \tag{4}$$

2.3. Dynamic equations

Quick-response satellites are relatively small in mass and volume and generally do not have rotating solar panel mechanisms, so they can be approximated as rigid bodies for simplified analysis[11]. According to the momentum theorem, the satellite attitude dynamics equation is:

$$I_b \omega_b + \omega_b^{\times} I_b \omega_b = T_e \tag{5}$$

Where I_b is the inertia matrix of the satellite relative to its center of mass in the body coordinate system, and ω_b is the component array of the satellite's absolute angular velocity vector in the body coordinate system.

3. PID control algorithm design

In the field of satellite attitude control, commonly used algorithms include the classic PID control algorithm, robust control theory, variable structure sliding mode control algorithm, etc. However, in the current mainstream satellite design methods, the classic control algorithm is still widely used. For the quick-response satellite analyzed in this paper, which has a relatively simple structure, the PID control algorithm can meet the mission control requirements. The input-output relationship of the PID controller is [12]:

$$u(t) = K_P \left[e(t) + \frac{1}{T_i} \int_0^t e(t)dt + T_d \frac{de(t)}{dt} \right] = K_P e(t) + K_I \int_0^t e(t)dt + K_D \frac{de(t)}{dt} \tag{6}$$

The transfer function expression is:

$$G_C(s) = K_P + \frac{K_I}{s} + K_D s = K_P \left(1 + \frac{1}{T_i s} + T_d s \right) \tag{7}$$

Where K_P, K_I, and K_D are the proportional, integral, and derivative parameters corresponding to the three axes, which can be obtained through pole placement method.

4. Simulink modeling and control algorithm verification

4.1. Simulink modeling

The satellite attitude control system consists of attitude dynamics, kinematics, controller, and error modules, as shown in Figure 3.

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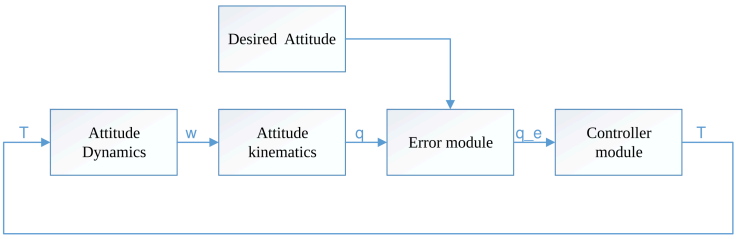

Figure 3: Composition of the satellite attitude control system

The attitude dynamics module takes control torque and satellite disturbance torque as inputs and outputs the satellite's attitude angular velocity. The kinematics module takes the satellite's attitude angular velocity as input and outputs the satellite's attitude angles, represented by the Euler quaternion q in this model. The error module takes the current measured attitude of the satellite and the desired attitude of the satellite as inputs and outputs the satellite's attitude error. The controller module takes the satellite's attitude error as input and outputs the satellite's control torque. The specific design methods for the attitude dynamics, kinematics, and controller modules have been described in the previous sections. Below, we will focus on introducing the modeling of the quaternion error module [8].

Assuming the desired attitude quaternion of the target is q_t and the current measured attitude quaternion is q_m, the error quaternion q_e can be represented as:

$$q_e = q_t^* \otimes q_m \tag{8}$$

4.2. Simulation analysis

The simulation conditions selected are as follows:

Satellite's inertia matrix: $I = \text{diag}\{2.27, 3.39, 4.42\}$ kg·m²;

Initial attitude angles: $\varphi_0 = 30°$, $\theta_0 = 20°$, $\psi_0 = 50°$;

Initial attitude angular velocity: $\omega_0 = 0$ rad/s;

Desired attitude angles: $\varphi_0 = 0°$, $\theta_0 = 0°$, $\psi_0 = 0°$;

Disturbance torque: $T_d = [0.005\sin(t)\ 0.005\sin(t)\ 0.005\sin(t)]^T$ N·m;

Other parameters: $K_P = \text{diag}\{3.71, 5.56, 6.63\}$, $K_d = \text{diag}\{22.6, 26.3, 28.63\}$.

The simulation results are shown in Figures 4~6.

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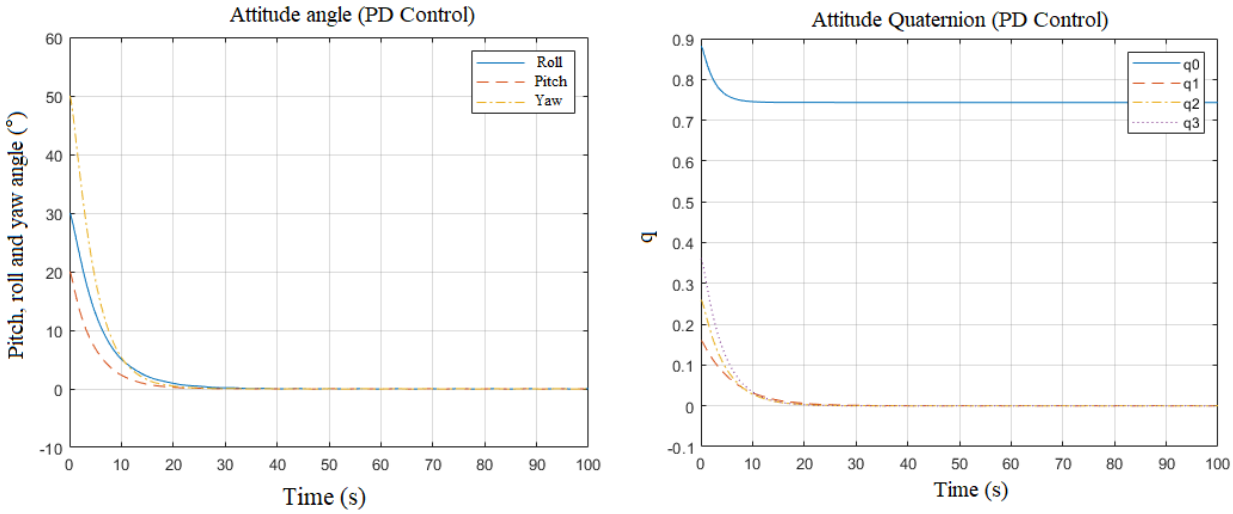

(a) Euler angle attitude (b) Quaternion attitude

Figure 4: Attitude angle simulation curves

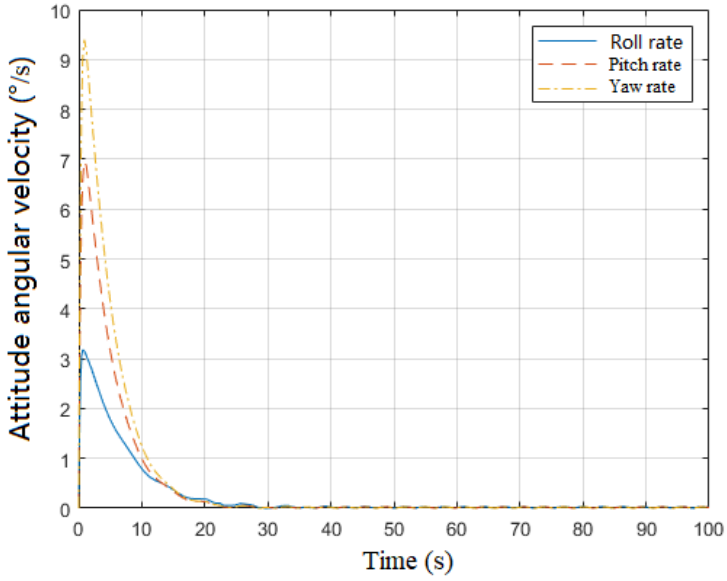

Figure 5: Attitude angular velocity curve

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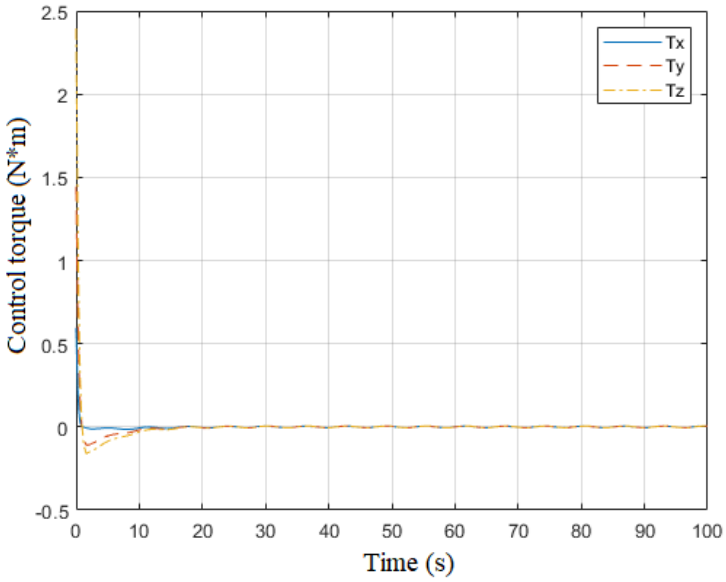

Figure 6: Control torque curve

Figure 4 shows the satellite attitude angle simulation curves, presented in two ways: Euler angles and quaternions. As can be seen from the figure, using PID control, the three-axis attitude angles reach a stable control state within 28 seconds. The roll angle has an overshoot of no more than 13%, while the pitch and yaw angles have no overshoot. Figures 5 and 6 show the satellite attitude angular velocity simulation curve and the control torque simulation curve, respectively. They are consistent with the trend of the attitude angle changes and remain generally stable, meeting the requirements for attitude control.

5. Summary

This article delves into the design and simulation of satellite attitude control systems against the backdrop of rapid applications in agile satellite space. Utilizing the quaternion method, we constructed computational models for agile satellite attitude kinematics and dynamics. Furthermore, we employed the PD control algorithm to design the crucial parameters of the control system. A simulation model was then established using Matlab Simulink, followed by numerical simulation analysis. The simulation results indicate that the designed control algorithm fulfills the requirements for rapid response control of large attitude maneuvers in agile satellite space. The pertinent theories laid out herein can serve as theoretical support for subsequent agile satellite space applications.

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