Attitude stability control of micro-nano satellite orbit maneuver based on bias momentum

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Abstract. In this paper, an attitude stability control strategy based on cold air micro propulsion microsatellite orbital maneuver is designed. Firstly, the influence of environmental disturbance moment, uncertainty of rotational moment of inertia and thrust eccentricity moment on the attitude stability of the micro-nano-satellite is considered, and the attitude dynamics model of the micro-nano-satellite based on biased momentum wheel and magnetic moment device is established. Then, the disturbance moments such as environmental disturbance moments, rotational inertia uncertainty and thrust eccentricity moments are analyzed. In order to suppress the influence of various internal and external disturbance factors on the stability of the micro-nano-satellite during the deorbiting process, a robust adaptive sliding mode variable structure controller is designed. The designed robust adaptive sliding mode controller is able to compensate for various disturbances adaptively. The robust adaptive sliding-mode variable structure controller is designed with a reasonable distribution of torque considering the characteristics of bias momentum wheel control and magnetic control. Finally, numerical simulations are performed, and the simulation results show that the system has good robustness.

Keywords: Micro-nano satellite, Cold air micro-thruster, offset momentum wheel, Magnetic torquer, Attitude control.

1. Introduction

Micro-nano-satellite is a new type of satellite that emerged in the 1990s with the development of delightful technologies in the fields of microelectronics, micromechanics, computers, new materials, etc. It is characterized by new technology, high functional density, low cost and flexible application. Entering the 21st century, with a variety of new technological breakthroughs, micro-nano-satellites have been rapidly developed, but also put forward new requirements for micro-nano-satellites. Rapid space response is a new concept in the modern space field, for sudden war, natural disasters and other emergencies, the rapid construction of emergency space systems through mobile means, to achieve rapid access to information, is a major initiative of modern space equipment from strategic support to tactical applications. Micro-nano-satellite with its unique advantages, in recent years in the military, civilian, commercial and other fields are widely used [1].
The control method using a combination of bias momentum wheel and magnetic moment device micro-nano-satellite before the orbital maneuver has the following three main stages of preparation: firstly, the rate damping phase, the rate damping phase, the main purpose is to reduce the impact of star-arrow separation, so that the angular velocity of the star quickly decay to a smaller range, this control phase is pure magnetic control; then the attitude capture phase, the satellite after the rate damping in the previous section In this control phase, the bias momentum wheel is rotated to the nominal speed, and the satellite is still coarsely oriented by pure magnetic control; finally, the attitude stabilization control phase, when the main task of the system is to eliminate the influence of various external environmental disturbances on the attitude stability of the satellite, and the bias momentum wheel and the magnetic momentum machine are jointly controlled in this phase [2].

In the literature [3], a magnetic moment energy control law based on attitude angle and attitude angular velocity feedback was proposed for the attitude capture problem of purely magnetically controlled microsatellites, but the control accuracy was not high. In [4], an LQR controller was designed based on pure magnetic control, but only the influence of environmental disturbance moments on attitude control was considered, and other disturbance moments were not considered. Literature 5] analyzed the effect of orbital control on flexural sail vibration and showed that the satellite orbital controller operates with a thrust force on the satellite, which couples with the sail vibration through the center-of-mass motion, causing sail excitation and thus affecting the attitude motion. The literature [6] analyzed the frictional moment of the micromotion wheel in the moment operation mode and used the control method of sliding mode variable structure to improve the control accuracy of the micromotion wheel. In the literature [7], a neural network variable structure control algorithm was designed for the flexible liquid-filled spacecraft, which suppressed sail vibration and liquid wobble, but the accuracy was not high.

In this paper, we propose to use a robust adaptive sliding-mode variable structure control approach to study the attitude stabilization control of micro-nano-satellite orbital maneuvers based on cold-air micro-propulsion. Based on the accurate modeling of the attitude dynamics and kinematics of the microsatellite with biased momentum wheel and magnetic momenter as actuators, the effects of environmental disturbance moments, rotational inertia uncertainties and thrust eccentricity moments on the attitude of the microsatellite during orbital maneuvers are analyzed. The design of robust adaptive sliding mode variable structure controller can compensate the disturbance adaptively and realize the attitude stability control of the micro-nano-satellite orbital maneuver under cold air micro-propulsion.

2. Dynamical model and problem description

2.1. Satellite attitude dynamics and kinematic modeling
A rigid body model of the micro-nano-satellite with momentum wheels is established, and the satellite body coordinate system is defined as the B-system, the satellite orbit coordinate system as the O-system, and the inertial coordinate system as the I-system. The dynamics of the micro-nano-satellite can be expressed as:

\[ \dot{w} + W^{T}(I\dot{w} + h_{w}) = u + u_{d} \]  

(1)

Where \( I \in R^{3 \times 3} \) is the inertia matrix of the micro-nano-satellite; \( w \in R^{3 \times 1} \) is the angular velocity of the body coordinate system with respect to the inertial coordinate system; \( u_{d} \in R^{3} \) is the environmental disturbance moment to the satellite; \( u \in R^{3} \) is the control moment of the satellite; \( h_{w} \in R^{3} \) is the angular momentum of the momentum wheel. \( W^{T} \in R^{3 \times 3} \) is the fork product matrix, for vector \( \gamma^{T} = [\gamma_{x}, \gamma_{y}, \gamma_{z}]^{T} \) can be expressed as
The component of $W_b = [W_{bx} \ W_{by} \ W_{bz}]^T$ in the b-system is $W$, which is the angular velocity vector of the satellite body coordinate system B with respect to the inertial coordinate system I. The relationship between $W$ and $W_b$ can be expressed as

$$W = W_b + c_b W_0$$

Where, $W_0$ denotes the angular velocity vector of the orbital coordinate system O with respect to the inertial coordinate system I.

Assume that the desired attitude of the satellite is $Q_d = [q_{od} \ q_d^T]^T$ and the current attitude is $Q_m = [q_{om} \ q_m^T]^T$.

The angular velocity error is

$$\tilde{W} = W - W_d$$

$W_d$ is the desired angular velocity

Due to:

$$q_{eo} = q_{om} q_{od} + q_d^T q_m$$

$$q_e = q_{od} q_m - q_{om} q_d - q_m^* q_d$$

$$q_e^2 + q_e^T q_e = 1$$

Then the quaternion error attitude kinematic equation of the micro-nano-satellite is

$$\begin{cases} 
\dot{q}_{eo} = -\frac{1}{2} q_e^T \tilde{W} \\
\dot{q}_e = \frac{1}{2} (q_e^* \tilde{W} + q_{eo} \tilde{W}) 
\end{cases}$$

2.2. Interference moment analysis

(1) Environmental disturbance moment

The orbit of the micro-nano satellite studied in this paper is around 500 km, and the main environmental disturbance moments are the aerodynamic moment $u_p$, the remanent magnetic moment $u_s$, the solar pressure moment $u_m$, and the gravitational gradient moment $u_n$. The total environmental disturbance moment to the satellite is expressed as

$$u_d = u_p + u_s + u_m + u_n$$
(2) Uncertainty of rotational inertia
The vibration of the battery array and antenna of the micro-nano-satellite will cause the change of rotational inertia, assuming that $I_o$ is the determined part of the rotational inertia of the satellite and $\Delta I$ the change part of the inertia, then the rotational inertia of the micro-nano-satellite is

$$I = I_o + \Delta I$$  \hspace{2cm} (10)

Bringing it into the micro-nano-satellite dynamics equation of equation (1) yields

$$I_o \ddot{w} + w^* I_o w + w^* \dot{h}_w + \Delta I w + w^* \Delta I \dot{w} = u + u_d$$  \hspace{2cm} (11)

From (11), the part of the rotational inertia change caused by $\Delta I \dot{w} + W^* \Delta I \dot{w}$, can be considered as the disturbance moment.

(3) Thrust eccentric moment
When the attitude maneuver of the micro-nano-satellite is stabilized to the three-axis state, the micro-thruster starts to work. Ideally, the thrust of the micro-thruster is through the satellite center of mass, and no thrust eccentric moment will be generated, so there will be no interference to the attitude control of the satellite, but in the actual working process of the thruster, due to the installation deviation of the thruster, the resulting thrust cannot be through the satellite center of mass. The thrust of the thruster is larger during the orbital maneuver of the micro-nano-satellite, and the thrust eccentricity moment has a great influence on the attitude of the satellite at this time. The X-axis of the satellite body coordinate system is the roll axis, the Y-axis is the pitch axis, and the Z-axis is the yaw axis. The cold gas micro-thruster installation diagram of the micro-nano-satellite is shown in Fig1.

![Installation diagram of the cold gas micro-thruster of the micro-nano satellite](image)

**Figure 1.** Installation diagram of the cold gas micro-thruster of the micro-nano satellite

The following analysis of the eccentric moments on the micro-nano-satellite begins.
In Fig. 2, $F_x$ is the orbital thrust during the deorbiting of a micro-nano satellite that normally passes through the satellite's center of mass, $O$ is the satellite's center of mass, and the disturbance moment generated by the thrust eccentricity can be introduced through Fig. 2 and expressed as

\[
\begin{cases}
T_z &= |ob| F_x' \\
T_x &= 0 \\
T_y &= -|OC| F_x'
\end{cases}
\]

(12)

2.3. Problem Description

Substituting the three main disturbance moments analyzed above into the dynamical equations of the micro-nano-satellite yields.

\[
I_0 \ddot{w} + w^* I_0 w + w^* h_w + \Delta I \dot{w} + w^* \Delta I w = u + u_d + M_c + Tm
\]

(13)

$M_c = \begin{vmatrix} M_{c_x} & M_{c_y} & M_{c_z} \end{vmatrix}$ is the control moment and $Tm = \begin{vmatrix} T_x & T_y & T_z \end{vmatrix}$ is the thrust eccentricity moment.

If the environmental disturbance moment, the uncertainty of rotational inertia, and the thrust eccentricity moment are unified as disturbance moments, the disturbance $u_d$ can be redefined as $u_{d1}$ below, and $u_{d1}$ can be expressed as

\[
u_{d1} = u_d + M_c + Tm - \Delta IW - W^* \Delta IW
\]

(14)

Then the satellite dynamics equation can be rewritten as:

\[
I_0 \ddot{w} + w^* I_0 w + w^* h_w = u + u_{d1}
\]

(15)

The interfering moments $u_{d1}$, $u_d$ are bounded functions satisfying
\[
\begin{align*}
|u_d| & \leq \delta_1 \\
|u_d| & \leq \delta_2
\end{align*}
\]  \hspace{1cm} (16)

\(\delta_1, \ \delta_2 > 0\), are constants.

The problem of attitude control of a micro-nano satellite during orbital maneuvers can be described as, considering the environmental disturbance moment, the uncertainty of rotational inertia and the influence of thrust eccentricity moment on attitude, designing the control moment \(u\), so that the attitude angular error of the satellite reaches 0° in finite time and the attitude angular velocity error reaches 0°/S, and then maintaining the attitude stability.

\[
\lim_{t \to \infty} q_e = 0 \quad \lim_{t \to \infty} \dot{W} = 0
\]  \hspace{1cm} (17)

3. Controller design

3.1. Sliding mode controller design and analysis

Define the slipform surface as:

\[
S = \dot{W} + Cq_e
\]  \hspace{1cm} (18)

Where, \(C > 0\) is a constant.

The design integral sliding die surface \(S_I, S_I\) can be expressed as

\[
S_I = S + \alpha \int_0^t S(\tau)d(\tau) - s(0)
\]  \hspace{1cm} (19)

\[
S_I(0) = 0
\]  \hspace{1cm} (20)

The proof is that the system is on the slipform surface from the beginning. When \(S_I = 0\), get

\[
\dot{S}_I = S + \alpha S = 0
\]  \hspace{1cm} (21)

Solve for:

\(S = Ae^{-\alpha t}\), when \(t \to \infty\),

\[
S = \lim_{t \to \infty} Ae^{-\alpha t} = 0
\]  \hspace{1cm} (22)

The Lyapunov function is chosen as

\[
V = \frac{1}{2} S_I^T L_o S_I > 0
\]  \hspace{1cm} (23)
According to the equivalence control principle.

\[ \dot{V} = S_f^T I_o S_f = S_f^T I_o (u - u_{eq}) \]  \hspace{1cm} (25)

\[ u_{eq} = W^* I_o W + W^* h_w - \frac{1}{2} C I_o q_e \dot{W} - \frac{1}{2} c I_o q_e \dot{W} - \alpha S \]  \hspace{1cm} (26)

Then the design controller \( u \) is

\[ u = W^* I_o W + W^* h_w - \alpha S - \frac{1}{2} C I_o q_e \dot{W} - \frac{1}{2} c I_o q_e \dot{W} - \sigma \text{sgn}(S_f) \]  \hspace{1cm} (27)

Substituting (27) into (25) yields.

\[ \dot{V} = S_f^T I_o \dot{S}_f = S_f^T (u - u_{eq}) < - \sigma S \text{sgn}(S_f) + |u_{eq}| |S_f| < (- \sigma + \delta_2) |S_f| < 0 \]  \hspace{1cm} (28)

Where \( \alpha > \delta_2 > 0 \) and is a constant.

It is known that \( \lim_{t \to \infty} S_f = 0 \) according to Lyapunov's stability theorem. Combining the conclusion derived from equation (21), it can be proved according to equation (18) that

\[ \lim_{t \to \infty} q_e = 0 \quad \text{and} \quad \lim_{t \to \infty} \dot{W} = 0 \]  \hspace{1cm} (29)

Thus, the attitude angle error and attitude angular velocity error are proved to be asymptotically convergent.

3.2. Control torque distribution

The pitch axis of the micro-nano-satellite is controlled by a bias momentum wheel. During the control of the bias momentum wheel, the drive voltage is limited and cannot achieve an arbitrarily large input, so the actual control input needs to be controlled by a saturation input. In order to eliminate the chattering caused by the sign function, the salt(s) function is therefore introduced. So, the controller \( u \) is modified as

\[ u = W^* I_o W + W^* h_w - \alpha S - \frac{1}{2} c I_o q_e \dot{W} - \frac{1}{2} C I_o q_e \dot{W} - \sigma \text{sat}(S_f) \]  \hspace{1cm} (30)
Where $\xi > 0$ is a constant.

The yaw and roll axes of the micro-nano-satellite are controlled by a magnetic torque converter. Considering the characteristics of the magnetic control torque, the actual generated control torque must be different from the desired torque, the switching process of the system is non-ideal, and the system will inevitably have the chattering problem, so for this situation, the form of the switching-free robust control rate is constructed as

$$u_{nl} = -KS$$  \hspace{1cm} (31)

Where $K > \delta_2 > 0$.

The corrected magnetic moments are

$$u_{exp} = u_{eq} + u_{nl}$$  \hspace{1cm} (32)

Construct the switchless robust control rate in the form of

$$u_{exp} = W^*I_0W + W^*h_\omega - \frac{1}{2}Cl_0q_\omega \dot{W} - \frac{1}{2}cI_0q^*_\omega \dot{W} - KS$$  \hspace{1cm} (33)

According to Krasovskii-LaSalle theorem [14], it is known that $\lim_{t \to \infty} S_t = 0$.

In order to make full use of the limited magnetic moment available from the satellite magnetic moment generator and to improve the control efficiency, the desired moment $u_{exp}$ is investigated by decomposing it in s-space into two parts: the part $u_{exp}^\perp$ perpendicular to the switching manifold $s$. and the part $u_{exp}^\parallel$ parallel to the switching manifold $s$. Only $u_{exp}^\parallel$ is compensated in the design of the control moment.

![Figure 3. Decomposition schematic diagram of expected torque in s space](image)

$$u_{exp}^\parallel = \|u_{exp}\| \cos \theta \frac{S}{\|S\|} = \frac{u_{exp} \cdot s}{\|S\|^2} s$$  \hspace{1cm} (34)

The final control torque of the system is
4. Simulation calibration

To verify in order to verify the effectiveness of the controller, this subsection takes a micro-nano-satellite with cold air micro-propulsion as an example to verify the adaptive sliding mode variable structure control performance proposed in this paper, using the following micro-nano-satellite parameters.

The micro-nano-satellite orbit is 500 Km, the satellite mass is 2.5 Kg, and the inertia matrix is [0.004 0.0148 0.0138] Kg.m². The dominant disturbance moments in the environmental disturbance moments are the remanent magnetic moment and the aerodynamic moment, and the solar pressure moment and the gravitational gradient moment are one order of magnitude lower than the aerodynamic moment and can be neglected. The remanent magnetic disturbance moment $[1.1 0.12 1.2] \times 10^{-7}$Nm, the pneumatic disturbance moment $[3\cos(\omega_0 t) 1.5\sin(\omega_0 t) + 3\cos(\omega_0 t) 3\sin(\omega_0 t)] \times 10^{-8}$Nm, assuming that the maximum inertial uncertainty of the microsatellite is diag $(7.95 \times 10^{-4}, 2.97 \times 10^{-3})$ Assuming that the maximum thrust of the micropropeller is 10 mN and the thrust eccentricity is 1 mm, the maximum thrust eccentricity moment is $[0 1 1] \times 10^{-5}$Nm.

The initial value of the three-axis attitude of the satellite is assumed to be $[0.5° 0.4° -0.3°] ^T$, and the initial value of the attitude angular velocity is $[0.01°/S 0.01°/S 0.01°/S] ^T$. The target attitude angle is $[0° 0° 0°] ^T$, and the target value of the attitude angular velocity is $[0°/S 0°/S 0°/S] ^T$. The controller parameters in Table 1 are used, and noise is added to the simulation interference, the attitude control system of the micro-nano satellite was simulated. The simulation parameters are shown in Fig. 4.

| Table 1. Parameters of sliding mode controller |
|-----------------------------------------------|
| Parameter | c | α | σ | K |
|----------|---|---|---|---|
| SMC      | 1 | 0.4 | 0.2 | 0.1 |

Simulation results of the three-axis attitude angle and attitude angular velocity variation are given below.

Figure 4. Three-dimensional angle change curve and angular velocity change curve
5. Conclusion
In this paper, we analyze the influence of internal and external disturbance factors on the attitude stability of micro-nano-satellite during orbital maneuvers, and design an adaptive sliding mode control law.

For the uncertainties of environmental disturbance moment, uncertainty of rotational inertia and thrust eccentricity moment, the adaptive variable structure sliding mode controller effectively suppresses the disturbance and uncertainty factors and weakens the jitter phenomenon, so that the three-axis angular error of the satellite converges to about 0.02° and the angular velocity error converges to about 0.05°/s. The simulation results show that the designed control system effectively improves the attitude stability of the micro-nano-satellite during orbital maneuvers.

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