Orbit-Attitude Coupled Tracking and Landing Control for an Asteroid

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Abstract. Recent research on asteroid exploration has shown tremendous potential to provide humankind detailed views of unexplored worlds in the inner solar system. A potential market of asteroid mining has aroused interest from commercial companies gradually. Asteroid exploration has the potential to offer the possibility to revolutionize the supply of many resources which are vital but exorbitant for human. A typical asteroid exploration mission is composed of remote asteroid detecting, approaching, hovering, landing and sampling, take-off and return, re-entry stage. Among these stages the tracking and landing control of the asteroid is crucial for the exploration. Rendezvous, approaching, fly-around and landing problem of an asteroid is investigated in this paper. The expected motion of the tracking spacecraft determine by the asteroid’s orbital and attitude status is presented firstly. According to the design of the OSIRIS-REx project, several constant thrusters are used for spacecraft to track the asteroid. Different control strategies like pulse control, saturated nonlinear control, limit cycle control and PWM based low-thrust control are developed to fulfil the asteroid’s tracking mission of different stages. Numerical simulations demonstrate the effectiveness of the tracking control.

1. Introduction

Asteroid exploration and sampling have shown tremendous potential to provide humankind detailed views of the formation and evolution of planetary system and life origin and evolution in the solar system. With the continuous development of space technology and the increasing complexity of space science research mission, the asteroid exploration is developed from the early asteroid fly-by method, the impact attachment method, to the current sampling and return detection methods gradually.

The soft landing problem on the asteroid surface is a key problem for deep space exploration. According to the different instruments and equipment carried by the asteroid detector, a variety of researches can be carried out cover a large number of scientific research, such as appraising of asteroid mining and composition and characteristics analysis of asteroid soil. These research can deepen the understanding of asteroids, providing a scientific basis for the further research and utilization of asteroids, and promoting the development of space science and technology.

The close-range detection of small celestial bodies began in the 1990s. In recent years, small celestial exploration has become one of the key areas of deep space exploration, the most representative of these space exploration missions includes the Near Earth Asteroid Rendezvous (NEAR)\textsuperscript{[1]} mission of NASA, the Stardust mission\textsuperscript{[2]} of NASA’s comet sample return research, the
Deep Impact mission [3] for studying the interior composition of the comet Tempel 1, the Hayabusa mission [4] developed by the Japan Aerospace Exploration Agency (JAXA) to return a sample of material from a small near-Earth asteroid named 25143 Itokawa and the Rosetta mission [5] built by the European Space Agency to perform a detailed study of comet 67P. Among these space missions, the most representative missions are the NEAR and Hayabusa, the former was the first space mission to achieved flying around and landing on an asteroid, the latter was the first space probe to achieve attachment and sampling return. The Origins Spectral Interpretation Resource Identification Security Regolith Explorer (OSIRIS-REx) mission was a NASA new frontiers mission launching in 2016 to rendezvous with the near-Earth asteroid (101955) 1999 RQ36 in late 2018 [6-7]. On 21 September 2018, Hayabusa2 ejected the first two rovers, Rover-1A and Rover-1B, from about 55 meters altitude that fell independently to the surface of the asteroid Ryugu [8]. The necessity and key techniques of asteroid autonomous landing was elaborated by Cui [9]. Peng Z [10] designed an adaptive super-twisting controller without chartering problem for spacecraft soft landing on asteroids.

For most of the references, researchers tend to focus on the control research in the power decent phase of asteroid landing [11], in the meantime, the controller only contains a study of orbital or attitude control design. This paper will divide close proximity asteroid tracking into several stages based on the OSIRIS-REx mission planning [12], and different control methods are adopted in different stages to satisfy different task requirements.

In this paper, the close proximity control process after asteroid rendezvous and docking is divided into an approaching phase, a long distance fly-around phase, a close distance fly-around phase, a soft landing phase. At different phase, thrusters with different parameters are adopted, and the orbit-attitude coupled controllers are designed separately according to different task requirements. In the process of the spacecraft approaching the asteroids, different orbit and attitude control targets need to be met, the details are as follows:

(1). Approaching phase: The control objectives include keeping the distance between the spacecraft and the asteroid in the range of 20km, and keeping the spacecraft’s certain instrument pointing to the asteroid during the process. The actuator is a 5N constant small thruster;

(2). Fly-around phase: The control objective is to achieve a full-scale observation of the asteroid, the optical camera should be able to fully cover the visible area of the asteroid. According to the requirements of optical navigation system, this phase is divided into two parts: long-distance flying and short-distance flying. The long-distance fly-around phase is to achieve relative optical navigation data acquisition. The actuator is a 5N constant small thruster, the radius of the relative circular orbit is about 20km. The close- distance fly-around phase mainly relies on relative optical navigation to check the asteroid’s topography, preparing for the landing site selection and sampling; the actuator is a 0.2N constant-value micro-thruster, the radius of the relative circular orbit is about 5km;

(3). Landing phase: The control objective is to achieve soft landing within the tolerance. In the landing and sampling phase, the 0.2N constant value small thruster is used to control the spacecraft to hover over the top of the landing site about 100m, and then the 0.09N variable micro-thruster is used for soft landing, the position error of the landing horizontal direction is less than 10m, and the vertical relative speed is less than 10cm/s.

2. Lander orbital and attitude dynamics

There are three kinds of coordinate frames used to represent the spacecraft and asteroid orbit and attitude dynamics: asteroid body-fixed frame $B_a$, spacecraft body-fixed frame $B_d$, and inertial-reference frame $N$. The inertial translational EOMs of the asteroid and spacecraft are:

$$\ddot{\mathbf{R}}_a = -\mu_a \mathbf{R}_a / |\mathbf{R}_a|^3, \quad \ddot{\mathbf{R}}_d = -\mu_a \mathbf{R}_d / |\mathbf{R}_d|^3 + \mathbf{U}/m_d + G$$

(1)

where $\mu_a$ is sun’s gravitational constant, $\mathbf{R}_a$ and $\mathbf{R}_d$ are the inertial position vectors of the asteroid and spacecraft respectively with respect to (w.r.t.) the inertial-reference frame $N$. $m_d$ is the mass of the spacecraft, $\mathbf{U}$ is the control forces of the spacecraft, $G$ is microgravity of the asteroid. The attitude dynamics of a rigid spacecraft can be described by the following nonlinear equation:
\[ J \dot{\omega} + \omega \times J \omega + t_d = t_c \]  

(2)

where \( \omega \) is the angular velocity of the spacecraft with respect to the inertial-reference frame \( \mathcal{N} \) and is expressed in the body-fixed frame \( \mathcal{B}_d \), \( J \) is the inertia matrix of the spacecraft, \( t_c \) is the torque control and \( t_d \) is the bounded external disturbance. The nonlinear differential equations governing the attitude kinematics of the spacecraft in terms of unit quaternion can be expressed as:

\[ \dot{q} = \frac{1}{2} \cdot Q(q) \omega \]  

(3)

2.1. Asteroid gravitational field model

There are three methods for calculating the irregular gravitational field of asteroids [13]: spherical harmonic function/ellipsoid harmonic function method[14,15], particle group method[16], and polyhedral method[17-20]. Itokawa is selected as a hypothetic asteroid for landing control in this paper, the gravitational field model of asteroid is calculated based on polyhedral model. Specifically, the asteroid is decomposed into a polyhedron composed of 1846 vertices and 3688 faces, the gravitational potential on a field point due to an asteroid modelled as a polyhedron can be defined as

\[ U = \frac{1}{2} G \sigma \sum_{e \in \text{edges}} r_e \cdot E_e \cdot r_e \cdot L_e - \frac{1}{2} G \sigma \sum_{f \in \text{faces}} r_f \cdot F_f \cdot r_f \cdot \omega_f \]  

(4)

where \( G \) is the universal gravitational constant, \( \sigma \) is the polyhedron density, \( r_f \) is a vector from the field point to any point on the facet, \( r_e \) is a vector from the field point to any point on the edge common to two facets, \( F_f \) and \( E_e \) are the second-order tensors corresponding to the facets and their edges, \( L_e \) is a dimensionless factor that represents the potential energy of the each edge, \( \omega_f \) is an angle defined for each facet.

2.2. Expected orbital and attitude motion

This section is going to obtain the expected translational and rotational motion of the spacecraft, such that the position of the spacecraft tracks the landing site of the asteroid and the attitude of the spacecraft pointing towards the landing site. Therefore the landing site in asteroid body-fixed frame \( \mathcal{B}_c \) is \( \rho_{land}^r \), due to soft landing requirements, the velocity in landing site is \( \rho_0 \). The expected landing position, velocity and acceleration vector projected in inertial-reference frame \( \mathcal{N} \) is

\[
\begin{align*}
\mathbf{R}_{md} &= \mathbf{R}_{c} + C_{c} \rho_{d} \\
\mathbf{R}_{dd} &= \mathbf{R}_{c} + C_{c} \left( \mathbf{b}_{B/N} \times \mathbf{b}_{d} \right) \\
\mathbf{R}_{dd} &= \mathbf{R}_{c} + C_{c} \left( \mathbf{b}_{B/N} \times \mathbf{b}_{d} \right)
\end{align*}
\]

(5)

where \( C_{c} \) is the rotation matrix from asteroid body-fixed frame \( \mathcal{B}_c \) to inertial-reference frame \( \mathcal{N} \), \( \mathbf{b}_{B/N} \) is the angular velocity of the \( \{ \mathcal{B}_c \} \) frame w.r.t. the \( \{ \mathcal{N} \} \) frame.

The objective of the attitude control is to make a certain instrument point towards the asteroid. The expected vector pointing from the spacecraft to the asteroid is given by \( \mathbf{d} = \mathbf{R}_c - \mathbf{R}_d \). Without loss of generality, this paper defines that the z axis of the spacecraft’s body frame should point to the center of mass of the asteroid. The direction of \( \mathbf{d} \) is \( \hat{e}_z \). The expected attitude maneuver can be expressed in quaternion as:

\[ q_e = \left[ \begin{array}{c} \cos \alpha/2 \\
\hat{e}_z^T \sin \alpha/2 \end{array} \right] \]  

(6)

where \( \hat{e}_z \) the rotating axis during the attitude maneuver, \( \alpha = \cos^{-1}(z^T \hat{e}_z) \) is the angle between two vectors \( \hat{e}_z \) and \( z \). The expected angular velocity and the expected angular acceleration of the spacecraft are:

\[
\begin{align*}
\mathbf{\omega}_d &= \mathbf{\omega} - \mathbf{\omega}_r = Q_{dB} \mathbf{\omega} - \mathbf{\omega}_r \\
\mathbf{\dot{\omega}}_d &= \mathbf{\dot{\omega}} - \mathbf{\dot{\omega}}_r = Q_{dB} \mathbf{\dot{\omega}} - \mathbf{\dot{\omega}}_r
\end{align*}
\]  

(7)
where $\dot{Q}_{DB} = -\omega_r \times Q_{DB}$ and $Q_{DB}$ is the rotation matrix from the spacecraft body fixed frame $\{B_d\}$ to target spacecraft body frame, $\omega_r$ is the manoeuvre angular velocity.

3. Controller design

Given the expected trajectories and attitude of the spacecraft with respect to the asteroid, a control input $U$ is expected to make the spacecraft land smoothly. Define the position tracking error as $e = \Delta r$. In approaching phase, limit cycle control with a 5N constant small thruster is adopted in approaching phase, the control objectives are to approach the asteroid within 20km, and keeping the spacecraft’s observation axis pointing to the asteroid during the process. The analog phase plane controllers include lead correction controller and pseudo rate controller. The lead correction controller is generally composed of a lead correction network and a Schmitt trigger, therefore the limit cycle controller is designed as:

$$u_c = \begin{cases} 
1 & u_i \geq u_o \\
1 & u_i > (1-h)u_o, u_i < 0 \\
0 & -(1-h)u_o \leq u_i \leq u_o, u_i > 0 \\
0 & -u_o < u_i \leq (1-h)u_o, u_i < 0 \\
-1 & u_i \leq -(1-h)u_o, u_i > 0 \\
-1 & u_i \leq -u_o 
\end{cases}$$

(8)

where $u_c$ is the output signal of controller, $h$ is the hysteresis coefficient, $u_i$ is input signal of the Schmitt trigger which is designed as $u_i = -K_1 \Delta r - K_2 \dot{r}$. In fly-around phase, the control objective is to achieve a detailed observation of the asteroid. According to the previous design, this phase is divided into long-distance fly-around and close-distance fly-around. For long-distance fly-around phase, limit cycle control with a 5N constant small thruster is adopted again, the radius of the relative circular orbit is about 20km. The controllers are same with the controllers in the approaching phase.

For close-distance fly-around phase, the actuator is a 0.2N constant-value micro-thruster, the radius of the relative circular orbit is about 5km. A nonlinear orbit control law is designed as:

$$u_i = \dot{R}_{dd} + \mu_i \dot{R}_d / |R_d| - \left[ K_1 \Delta r - [K_2 \Delta \dot{r} \right]$$

(9)

For a large number of thrusters, it is convenient to achieve on-and-off switch without adjusting the magnitude of the thrust. Meanwhile, constant thruster reduces the requirements on the thruster and improves the reliability of the controller. Pulse width modulation (PWM) can be utilized to simulate the control signal calculated by the designed algorithm for micro thruster. Amplitude modulation ratio is an important parameter of PWM, which is defined as $K_a = V_m / V_c$, where $V_m$ is peak of modulated wave, $V_c$ is peak of carrier wave. Control law in landing phase has the same form with different control parameters as the previous control law in fly-around phase.

A nonlinear attitude control law can be designed as:

$$T_c = \omega \times (J \omega) + \lambda \dot{\omega}_d - D(\omega - \omega_d) - Kq_m$$

(10)

where $D$ and $K$ are diagonal matrix with positive diagonal elements, $\lambda$ is constant feedback coefficient, $\dot{\omega}_d$ is the expected angular acceleration, $T_c$ is the control torque.

4. Numerical simulations

The scope and effect of the previous control law will be analysed through numerical simulations. The initial orbit parameters $R_{e0}$ and $V_{e0}$ of asteroid are calculated by six Keplerian elements of asteroid Eros. The asteroid is 1500 kg with the moment of inertia $J_c = [500 700 1000] \text{kg} \cdot \text{m}^2$. The initial orbit parameters of spacecraft $R_{d0} = R_{e0} + [20; 30; -30]$ and $V_{d0} = V_{e0}$. Ass of landing spacecraft is 200 kg with the moment of inertia $J_d = [3 3.3] \text{kg} \cdot \text{m}^2$. The initial Euler Angle and $\omega$ of asteroid and spacecraft are zeros. The latitude and longitude of the landing site are 45 and 90 degrees. In approaching phase and long-distance fly-around phase, limit cycle control is adopted, $h=0.2$ is the hysteresis coefficient, $u_m = 5N$ is the constant thrust, $u_D = 4.5N$ is control parameter in limit cycle.
control method. PWM based constant thrust $u_{th} = 0.2N$ is used in short-distance fly-around phase and landing phase, the sawtooth amplitude is 0.1 and the carrier frequency is 0.001 Hz.

Due to paper length limitations, attitude control effect will be demonstrated by table only. Table 1 shows pointing angle mean error and maximum torques in different phase.

|     | approaching | fly-around | landing |
|-----|-------------|------------|---------|
| PA error (Deg) | 0.1         | 0.0012     | 0.01    |
| Torque (Nm)    | 0.4         | $1.4 \times 10^{-5}$ | $3 \times 10^{-4}$ |

Tracking and landing simulation of tracking spacecraft relative to asteroid are shown as follows.

![Figure 1](image1) trace in approaching phase

![Figure 2](image2) control force in approaching phase

![Figure 3](image3) trace and force in long-distance (20 km) and short-distance (5 km) fly-around phase

![Figure 4](image4) trace in landing phase

![Figure 5](image5) vertical soft landing and control force

Figure 1-Figure 5 show the trajectories of spacecraft relative to asteroid in different phase. As we can see from these graphs, the spacecraft completed the asteroid landing mission well with pre-set thruster parameters. The pulse used in each phase are shown in the table 2.

|     | approaching | long-fly | long-fly | hover | landing |
|-----|-------------|----------|----------|-------|---------|
|     | 19583.35    | 105584.42| 18114.31 | 2514.81| 384.96  |

5. Conclusion

In this study, a detailed landing detection project is proposed for asteroid exploration, we divide close proximity asteroid tracking into several stages. Different control methods are adopted in different stages to meet the limits of different thrusters. The expected motion of the tracking spacecraft...
determined by the asteroid’s orbital and attitude status is presented firstly and several controllers is
designed for the landing detection. The gravitational acceleration of asteroid is calculated by the
polyhedral model. Terminal landing accuracy and attitude pointing accuracy have meet the demand by
proposed control algorithm. Simulations results of asteroid detection problems are given to
substantiate the effectiveness of landing control and illustrate the reliability of the multi-controllers
cooperation control strategy.

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