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Reliable spacecraft rendezvous without velocity measurement

Shaoming He *, Defu Lin

School of Aerospace Engineering, Beijing Institute of Technology, Beijing 100081, People's Republic of China

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ABSTRACT

This paper investigates the problem of finite-time velocity-free autonomous rendezvous for spacecraft in the presence of external disturbances during the terminal phase. First of all, to address the problem of lack of relative velocity measurement, a robust observer is proposed to estimate the unknown relative velocity information in a finite time. It is shown that the effect of external disturbances on the estimation precision can be suppressed to a relatively low level. With the reconstructed velocity information, a finite-time output feedback control law is then formulated to stabilize the rendezvous system. Theoretical analysis and rigorous proof show that the relative position and its rate can converge to a small compacted region in finite time. Numerical simulations are performed to evaluate the performance of the proposed approach in the presence of external disturbances and actuator faults.

1. Introduction

Owing to the stringent performance requirement, guidance law design for spacecraft rendezvous is still an active research area and a benchmark problem for engineers [1–6]. Spacecraft rendezvous mission refers to the trajectory maneuver command that guides a chaser spacecraft to arrive at the target spacecraft with the same velocity. Such mission is typically divided into many phases, including far-range closing, close-range closing, and final mating [7]. This paper mainly focuses on guidance scheme design for the close-range phase, where the main concern of this phase is the rendezvous precision instead of the energy consumption.

Since the chaser spacecraft is moving in the vicinity of the target vehicle during the close-range rendezvous, the relative motion between these two vehicles can be described by the well-known Clohessy-Wiltshire (C-W) differential equations [8] or Tschauner-Hempel (T-H) differential equations [9,10]. The difference between these two models lies in that the C-W equations are for circular target orbit while the T-H equations are for elliptical target orbit. This paper mainly focuses on the elliptical orbit rendezvous problem.

With the development of modern control theory, many elegant guidance schemes have been proposed in recent years to achieve autonomous rendezvous. To cite a few, the authors in Ref. [11] proposed a robust guidance law for circular orbital rendezvous based on multi-objective H_∞ control approach. This controller can be easily obtained by a convex optimization problem with linear matrix inequality constraints, but it only considers the state-feedback control scheme. To improve the overall guidance performance, the parametric Lyapunov differential equation approach was also a powerful tool that can be used to design rendezvous laws [12,13] for uncertainty-free rendezvous system. Compared to the rendezvous guidance laws based on the quadratic regulation theory, the parametric Lyapunov approach is easy to implement. To cope with uncertainties in the plant parameters and reduce the computational burden for onboard implementation, a passivity-based direct adaptive spacecraft proximity operational strategy was derived upon the simple adaptive control methodology in Ref. [14]. Under the framework of integral sliding mode control technique, the authors in Ref. [15] developed a robust control law for spacecraft formation flying, where the equivalent part was based on linear quadratic optimal control method. By combining backstepping methodology with sliding mode approach, the authors in Ref. [16] presented a new guidance law for uncertain spacecraft rendezvous system. Although the sliding mode rendezvous guidance law is robust to external perturbations, the inherent chattering problem limits its practical applications. To eliminate the reaching phase and guarantee global sliding manifold, an adaptive robust rendezvous guidance law was proposed in Ref. [17] based on the time-varying sliding mode control approach with a complicated sliding surface design. Integrating extended state observer with nonsingular terminal sliding mode, the authors in Ref. [18] proposed a finite-time convergence rendezvous law with full state feedback. The authors in Ref. [19] considered the application of State-Dependent Riccati Equation (SDRE) for coupled orbital and attitude relative motion of formation flying or rendezvous. The SDRE controller can provide optimal control performance in terms of a meaningful index for the spacecraft formation

* Corresponding author.
E-mail addresses: shaoming.he.cn@gmail.com (S. He), lindf@bit.edu.cn (D. Lin).

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system, but it is sensitive to external disturbances. As model predictive control (MPC) is one of the promising nonlinear techniques for regulating nonlinear systems while considering multiple constraints, the authors in Refs. [20,21] considers the application of MPC methodology to rendezvous control law design. Despite its advantages, the MPC method usually requires high computational load and thus is difficult for the implementation in the onboard embed system.

It should be noted that most of the above mentioned rendezvous laws are full state feedback control laws. However, the relative velocity measurement is always an issue for sensor-less low-cost space vehicles. Considering this, output feedback design using only position measurement is more desirable for spacecraft rendezvous from the viewpoint of real application. The typical solution to the output feedback for spacecraft applications includes the design of an ad hoc dynamic filter, i.e., Kalman filter for example, driven by the output only [22,23]. However, in these works, no external disturbances were considered for the filter design, although the perturbations are inevitable in real applications. To this end, the authors in Ref. [24] proposed an output feedback rendezvous law based on high-gain observer and input-to-state stability to suppress the effect of the external disturbances to a specified level. However, only asymptotical stability is considered in this reference, which means that the convergence of the relative range and its rate is exponential. Obviously, the control law with infinite settling time is not desirable for the critical high-value rendezvous missions, where the control precision, especially during the final close-in phase, is of great importance for many space operations. It is well known that finite-time convergence ensures better disturbance rejection performance and smaller steady-state tracking errors than asymptotically stable systems [25,26]. Therefore, it is more desirable to design finite-time velocity-free rendezvous control laws.

Motivated by the above analysis, this paper investigates the problem of designing a robust finite-time convergence guidance law for elliptical orbital spacecraft rendezvous using only relative position measurements. Comparison results show that the proposed law has better disturbance rejection performance and smaller steady-state tracking errors than the asymptotical convergence law. The key features and contributions of the proposed approach are summarized as follows.

(1) Using homogeneous and Lyapunov theories, a robust observer is proposed to precisely estimate the relative velocity between two neighboring spacecraft despite external disturbances in finite time.
(2) With the reconstructed velocity information, a novel output feedback rendezvous law is then synthesized to regulate the relative distance and its rate to a small compact region in finite time. Moreover, analysis also proves that the proposed rendezvous law is robust against actuator faults and thus is a reliable control approach.

The rest of the paper is organized as follows. In Sec. 2, some backgrounds and preliminaries are stated, followed by the proposed output feedback rendezvous law derived in Sec. 3. Finally, some simulation results and conclusion remarks are offered.

2. Backgrounds and preliminaries

2.1. Problem formulation

This work assumes the chaser spacecraft is equipped with a high-performance low-level attitude control system that provides attitude stabilization and maneuver tracking, i.e., there is no time delay in tracking the attitude command provided by the guidance loop. This study aims to design the guidance input to this low-level controller to achieve autonomous rendezvous.

Consider the target-orbital rotating coordinate system $x-y-z$, as shown in Fig. 1, where the origin is fixed at the center of mass of the target spacecraft. $\boldsymbol{R}$ and $\boldsymbol{r}$ represent the vector from the center planet to the target spacecraft and the vector from the target spacecraft to the chaser spacecraft, respectively.

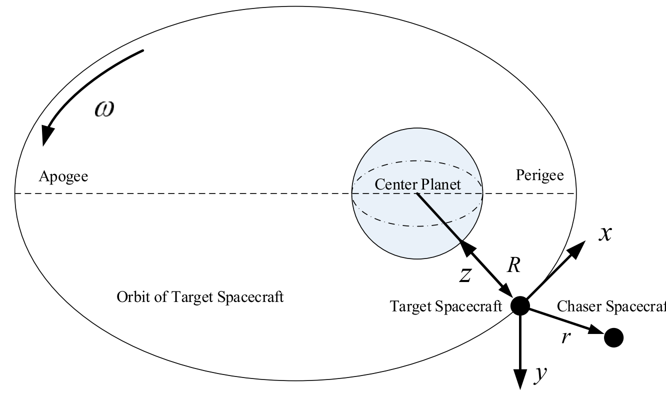

Fig. 1. Target-orbital rotating coordinate system.

As this paper considers elliptical orbit rendezvous problem, the relative motion kinematics of the chaser with reference to the target spacecraft can be formulated as [27].

$$\frac{d^2\boldsymbol{r}}{d^2t} = -\mu\left(\frac{\boldsymbol{R}+\boldsymbol{r}}{||\boldsymbol{R}+\boldsymbol{r}||^3} - \frac{\boldsymbol{R}}{||\boldsymbol{R}||^3}\right) + \boldsymbol{u} + \boldsymbol{w} \tag{1}$$

where μ represents the gravity constant, $\boldsymbol{u}$ the controlled acceleration input vector and $\boldsymbol{w}$ the lumped disturbance.

Assumption 1. The unknown external disturbances, in space, including environmental disturbance, solar radiation and magnetic effect, are all bounded in reality. With this mind, the norm of the lumped disturbance satisfies $||\boldsymbol{w}|| \le \delta$, where $\delta > 0$.

For simplicity, defining the notations as $\boldsymbol{r} = [x, y, z]^T$, $\boldsymbol{u} = [u_x, u_y, u_z]^T$ and $\boldsymbol{w} = [w_x, w_y, w_z]^T$, then, system (1) can be written as

$$\begin{cases} \ddot{x} = \omega^2 x + 2\omega\dot{z} + \dot{\omega}z - \dfrac{\mu x}{||\boldsymbol{R}+\boldsymbol{r}||^3} + u_x + w_x \\ \ddot{y} = -\dfrac{\mu y}{||\boldsymbol{R}+\boldsymbol{r}||^3} + u_y + w_y \\ \ddot{z} = \omega^2 z - 2\omega\dot{x} - \dot{\omega}x - \mu\left(\dfrac{z-R}{||\boldsymbol{R}+\boldsymbol{r}||^3} + \dfrac{1}{R^2}\right) + u_z + w_z \end{cases} \tag{2}$$

where ω denotes the angular rate of the target orbit, which is governed by

$$\omega = \sqrt{\frac{\mu(1+e\cos\theta)}{R^3}} \tag{3}$$

where $e \in [0, 1)$ represents the eccentricity of the orbit, and θ stands for the true anomaly.

During the close-range closing phase, the distance between the chaser and the target spacecraft r is much smaller than the distance between the planet center and the target R, i.e. $r \ll R$. With this fact in mind, system (2) can be further simplified as [27].

$$\begin{cases} \ddot{x} = \omega^2 x + 2\omega\dot{z} + \dot{\omega}z - \dfrac{\mu x}{R^3} + u_x + w_x \\ \ddot{y} = -\dfrac{\mu y}{R^3} + u_y + w_y \\ \ddot{z} = \omega^2 z - 2\omega\dot{x} - \dot{\omega}x + \dfrac{2\mu z}{R^3} + u_z + w_z \end{cases} \tag{4}$$

In order to determine the target orbit, the following two augmented equations are required [28].