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Differential X-ray pulsar aided celestial navigation for Mars exploration

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ABSTRACT

Celestial navigation (CeleNav) has been successfully used for position determination in many Mars exploration missions. Traditional CeleNav usually uses star angle as measurement, which is a function of spacecraft's position. Its accuracy is affected by the distance between the spacecraft and the Mars. X-ray pulsar-based navigation (XNAV) is a feasible autonomous method which can provide highly accurate distance measurements. However, single pulsar navigation system is unobservable and its navigational performance is influenced by the corresponding systematic biases caused by the pulsar directional error and the spacecraft-borne clock error. For the complementary of the CeleNav and the XNAV, the differential X-ray pulsar aided CeleNav method is proposed, which adopts the time-differenced X-ray pulsar measurement to eliminate the major part of systematic biases and improve the navigation accuracy. Simulations demonstrate the feasibility and effectiveness of this method.

1. Introduction

Mars exploration will attract more attention from the world in the near future [1,2]. Currently, the orbit determination of spacecraft primarily depends on Deep Space Network (DSN) [3,4], which is the biggest and most sensitive communication network for tracking and command system. Long communication delays between ground station and spacecraft still exist due to the large distance between the Mars and the Earth. Taking a Mars explorer as an example, the average distance between the Mars and the Earth is 225 million km and the corresponding two-way communication time is about 14 min [5]. Furthermore, the Sun transit caused outage is also a problem [6,7]. These problems make this navigation method hard to meet the requirement of stable and real-time navigation for Mars exploration. To improve the reliability, autonomous navigation needs to be researched.

Celestial navigation (CeleNav) is an autonomous navigation method for deep space spacecrafts [8–10]. It enables rapid and precise orbit determination by using celestial measurements such as star angles [11,12], line-of-sight [13], X-ray pulsars [14,15] and Doppler velocity [16,17].

Star angle is the angle subtended at the spacecraft between the line of sight to the star and that of the celestial body nearby such as the Mars, which is usually used as a traditional CeleNav measurement. It has following characteristics: (1) no additional equipment except the standard attitude sensors such as the star and earth sensors is needed; (2) not only orbit information but also attitude data can be provided. The navigation accuracy of CeleNav using star angle reduces with the increase of the distance between the spacecraft and the Mars [18].

X-ray pulsars are rapidly rotating neutron stars emitting pulsed radiation in the X band of electromagnetic spectrum [19]. The novel X-ray pulsar-based navigation (XNAV) can provide position and velocity for spacecraft autonomously in the solar system using observations of the X-ray emissions from pulsars [20–22]. The difference between the pulse time-of-arrival (TOA) at the spacecraft observed by X-ray sensor and its corresponding arrival time at solar system barycenter (SSB) predicted by the pulse timing model is usually used as a pulsar navigation measurement [23–25]. For the XNAV system, accurate pulse phase measurements from at least three noncoplanar pulsars are required for the determination of spacecraft's position. Reference [26] proved that single pulsar navigation system is completely unobservable. However, it is a great burden on expenses and weight to have three X-ray sensors. Furthermore, the systematic biases caused by the pulsar directional error and the spacecraft-borne clock error will influence the navigation performance.

Considering the complementarity of the traditional CeleNav and XNAV, reference [27] uses a federated Unscented Kalman filter (UKF) to fuse the CeleNav and XNAV for satellites, but it neglects the pulsar directional error and the spacecraft-borne clock error. Reference [23] proposes a pulse time difference of arrival (TDOA) technology to eliminate the effects of these systematic biases. However, this technology requires the TOA measurements of several spacecrafts simultaneously and cannot be applied for

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a single spacecraft. For the reason that these errors change little within a short time, reference [28] proposes a method adopting time-differenced measurement, which is the difference of the measurements at the neighbor epochs, instead of different spacecrafts. However, this method still needs to observe three pulsars simultaneously and is used for Earth-orbiting satellite.

Our study shows that a great improvement in navigation performance can be achieved through the X-ray pulsar aided autonomous CeleNav only observing one X-ray pulsar. To eliminate the systematic biases, a novel differential X-ray pulsar aided CeleNav method is proposed in this paper. The X-ray pulsar measurements of previous and current time are measured respectively, and their difference is adopted as the navigation measurement to assist the traditional CeleNav using star angle. Since the orbit dynamic model and measurement models of this method are nonlinear, and the measurements are related to the state vector of both current time and previous time, the UKF using previous estimation is adopted as the navigation filter. Simulations demonstrate that this method can achieve higher accuracy.

The remaining parts of the paper are organized as follows. After this introduction, the basic principle of traditional CeleNav method using star angle is introduced in section 2. Section 3 presents the measurement models of the XNAV and the time-differenced XNAV respectively. In section 4, the differential X-ray pulsar aided CeleNav method using UKF and previous state estimation is given. A comparison of navigation performance between the traditional CeleNav, single pulsar navigation, X-ray pulsar aided CeleNav, and differential X-ray pulsar aided CeleNav is shown in section 5. Finally, conclusions are summarized in section 6.

2. Traditional CeleNav using star angle measurement

On the premise that the position of a celestial body (the sun, earth, moon or stars) in an inertial frame at a certain time is known and its direction observed from the spacecraft in the spacecraft body frame is a function of the spacecraft's position, we can use a star angle measurement to estimate the position of the spacecraft. Since it contains measurement noise, unscented Kalman filter is used to obtain the optimal estimation. The state model is established from the orbital dynamics and the measurement model is established from the geometrical relationship of star angles.

2.1. State model

When a spacecraft is on an Earth–Mars transfer orbit, its accurate orbital dynamical model in the Sun-centered inertial frame (J2000.0) can be written as follows, which takes celestial bodies' gravity accelerations and solar radiation pressure acceleration into account.

$$\begin{cases} \dot{\boldsymbol{r}} = \boldsymbol{v} \\ \dot{\boldsymbol{v}} = \boldsymbol{a}_s + \boldsymbol{a}_m + \boldsymbol{a}_c + \boldsymbol{a}_p + \boldsymbol{w}_v \end{cases} \tag{1}$$

where $\boldsymbol{r}$ and $\boldsymbol{v}$ are the position and velocity of the spacecraft with respect to the Sun respectively. $\boldsymbol{a}_s$ is the gravity perturbation caused by the Sun. $\boldsymbol{a}_m$ is the gravity perturbation caused by Mars. $\boldsymbol{a}_c$ is the gravity perturbation caused by the other celestial bodies except Mars and the Sun. $\boldsymbol{a}_p$ is solar radiation pressure perturbation. $\boldsymbol{w}_v$ is the process noise that comes from propulsion and miscellaneous perturbations [29].

$\boldsymbol{a}_s$ can be expressed as

$$\boldsymbol{a}_s = -\mu_s \frac{\boldsymbol{r}}{r^3} \tag{2}$$

where μ_s is the gravitational constants of the Sun. r represents the scalar of $\boldsymbol{r}$.

$\boldsymbol{a}_m$ can be expressed as

$$\boldsymbol{a}_m = -\mu_m [\frac{\boldsymbol{r}_{sm}}{r_{sm}^3} + \frac{\boldsymbol{r}_m}{r_m^3}] \tag{3}$$

where μ_m is the gravitational constants of Mars. $\boldsymbol{r}_{sm} = \boldsymbol{r} - \boldsymbol{r}_m$ is the position vectors of the spacecraft relative to Mars, and $\boldsymbol{r}_m$ is the position vectors of Mars relative to the Sun.

$\boldsymbol{a}_c$ can be expressed as [30]

$$\boldsymbol{a}_c = -\sum_i^N \mu_i [\frac{\boldsymbol{r}_{si}}{r_{si}^3} + \frac{\boldsymbol{r}_i}{r_i^3}] \tag{4}$$

where i varies from 1 to N, representing Jupiter, Venus, Earth, Mercury, Saturn, Neptune, Uranus, and Pluto respectively. μ_i is the gravitational constants of the ith celestial body. $\boldsymbol{r}_{si} = \boldsymbol{r} - \boldsymbol{r}_i$ is the position vectors of the spacecraft relative to the ith celestial body, and $\boldsymbol{r}_i$ is the position vectors of the ith celestial body relative to the Sun.

$\boldsymbol{a}_p$ is solar radiation pressure perturbation which can be expressed as [31]

$$\boldsymbol{a}_p = -c_R p_{SR} (\frac{A}{m}) \frac{\boldsymbol{r}}{r} \tag{5}$$

where c_R is reflection coefficient that is related to the material and shape of spacecraft surface. The value of c_R equals 1 when the radiation is totally absorbed, and it equals 2 when totally reflected. m is the spacecraft's mass, and A is the spacecraft's sectional surface area. p_{SR} is the solar radiation pressure given to first order as [31]

$$p_{SR} = \frac{G_1}{r^2} \tag{6}$$

where G_1 is the solar radiation force constant at 1 AU (astronomical unit) and is approximately equal to 1×10^{14} kg·km/s^2.

In order to quantitatively analyze each acceleration, we exploit STK (Systems Tool Kit) to export planetary masses and create the ideal trajectory of the spacecraft using the orbit parameters of Mars Pathfinder. The planetary ephemerides data is generated by STK using DE421 ephemeris, and the simulation period was from July 1, 1997, at 00:00:00.000 UTCG to July 6, 1997, at 00:00:00.000 UTCG. During this period, r varies from 2.31×10^8 km to 2.33×10^8 km. a_s varies from 2.49×10^{-3} m/s^2 to 2.47×10^{-3} m/s^2. a_m varies from 6.15×10^{-6} m/s^2 to 3.89×10^{-4} m/s^2. a_c varies from 1.17×10^{-7} m/s^2 to 1.22×10^{-7} m/s^2. From Eq. (6), p_{SR} is about 1.9×10^{-6} kg/m·s^2. Assuming the area to mass ratio of the spacecraft $A/m = 0.01$ m^2/kg, the magnitude of a_p is nearly 10^{-8} m/s^2.

Through comparing the magnitudes of each perturbation, we can find that a_c and a_p are far less than a_s and a_m over the navigation period. We regard them as process noise and consider the orbital motion as a perturbed two-body problem with the Sun as the central body. The simplified dynamical model is written as follows.

$$\begin{cases} \dot{\boldsymbol{r}} = \boldsymbol{v} \\ \dot{\boldsymbol{v}} = \boldsymbol{a}_s + \boldsymbol{a}_m + \boldsymbol{w}'_v \end{cases} \tag{7}$$

where $\boldsymbol{w}'_v = \boldsymbol{a}_c + \boldsymbol{a}_p + \boldsymbol{w}_v$. Eq. (7) can be written in the general form as

$$\dot{\boldsymbol{X}}(t) = f(\boldsymbol{X}(t), t) + \boldsymbol{W}(t) \tag{8}$$

where the state variables $\boldsymbol{X} = [\boldsymbol{r}, \boldsymbol{v}]^T$. $\boldsymbol{W} = [\boldsymbol{0}, \boldsymbol{w}'_v]^T$ is process noise with covariance $\boldsymbol{Q}$.