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An adaptive filter method for spacecraft using gravity assist

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ABSTRACT

Celestial navigation (CeleNav) has been successfully used during gravity assist (GA) flyby for orbit determination in many deep space missions. Due to spacecraft attitude errors, ephemeris errors, the camera center-finding bias, and the frequency of the images before and after the GA flyby, the statistics of measurement noise cannot be accurately determined, and yet have time-varying characteristics, which may introduce large estimation error and even cause filter divergence. In this paper, an unscented Kalman filter (UKF) with adaptive measurement noise covariance, called ARUKF, is proposed to deal with this problem. ARUKF scales the measurement noise covariance according to the changes in innovation and residual sequences. Simulations demonstrate that ARUKF is robust to the inaccurate initial measurement noise covariance matrix and time-varying measurement noise. The impact factors in the ARUKF are also investigated.

1. Introduction

Planetary gravity assist (GA) is believed to be one of the most promising techniques for outer planet exploration, which allows spacecraft to reach its outer solar system destination that would not be accessible with current technology, or with significantly reduced propulsion requirements or reduced travel time [1–3]. GA can transfer energy from a planet to a spacecraft or vice versa to change the direction and magnitude of the heliocentric velocity of the spacecraft when the spacecraft enters sphere of influence of the target planet [4–6]. GA has been successfully used in many deep space missions, such as Voyager [7], Galileo [8], Cassini [9], etc.

Although the use of GA can reduce the propulsion requirement of space missions, the challenge for a GA mission is to provide precise orbit of the spacecraft to meet stringent orbit control requirements [10]. To guarantee the success of GA mission, celestial navigation (CeleNav) is often used [11–13]. CeleNav enables rapid and precise orbit determination using the measurement obtained from the images taken by onboard spacecraft camera of flyby celestial body [14].

Because of the existence of process and measurement noises, the CeleNav usually uses an optimal filter combined with the celestial measurements and the spacecraft orbit dynamics to estimate the position of the spacecraft. By far the most widely applied estimation algorithm for orbit determination includes the extended Kalman filter (EKF), unscented Kalman filter (UKF), etc [4,15]. No matter EKF or UKF, their navigation performance significantly relies on the quality of *a priori* information about the process and measurement noise. However, there are many error sources in the CeleNav system during the GA, such as target body's ephemeris errors, camera center-finding bias, and frequency of the images. These errors lead to the time-varying measurement noise as demonstrated by linear covariance analysis in many literatures [16,17]. Because the inaccurate measurement noise

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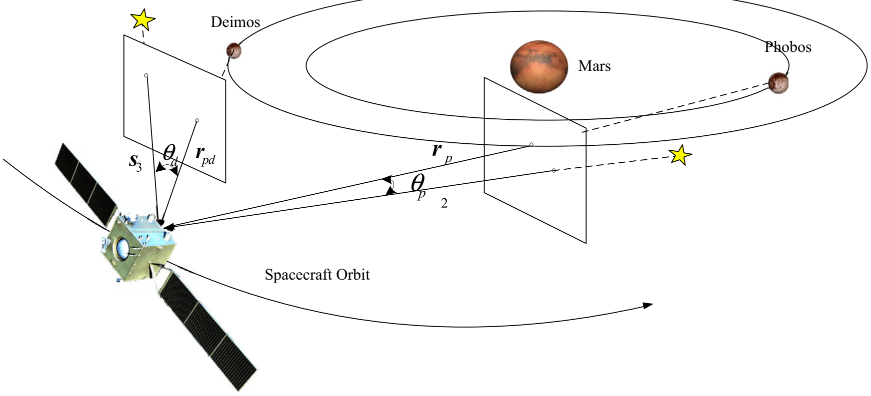

Fig. 1. Star angles between the stars and the navigation celestial body.

covariance matrix ($\boldsymbol{R}$) can degrade the CeleNav system performance, it is necessary to adapt $\boldsymbol{R}$ to accommodate for changes of measurement noise.

Over the past few decades, there have been many investigations in the area of adaptive filter. Popular adaptive filter methods include covariance estimation [18–20], Multiple Model Estimation (MME) [21–24], and covariance scaling [25,26]. The covariance estimation method calculates $\boldsymbol{R}$ directly from the innovation or residual sequence. The MME method uses multiple Kalman filters that run simultaneously with different $\boldsymbol{R}$. The covariance scaling method adjusts $\boldsymbol{R}$ by introducing a scaled factor. In this paper, an adaptive UKF with $\boldsymbol{R}$ scaled (ARUKF) is developed. The scaled factor is introduced to scale $\boldsymbol{R}$ based on the innovation and residual sequences. Improvement in the CeleNav system's accuracy and robustness with regard to time-varying measurement noise during the GA flyby has been achieved by the proposed method.

The remaining parts of the paper are organized as follows. In Section 2, the state model and measurement model of spacecraft during Mars flyby and mathematical formulation of the CeleNav are described. Section 3 presents the development of the proposed ARUKF method in detail. Simulation results about navigation performance comparison between ARUKF and UKF and impact factors analysis in ARUKF are given in Section 4. Finally, conclusions are summarized in Section 5.

2. Mathematical model of spacecraft CeleNav system

To evaluate the adaptive method of ARUKF used in CeleNav system during a GA flyby, the scenario chosen for this study is a spacecraft on a Mars GA flyby. The state and measurement models of the spacecraft on such a swing-by trajectory are presented in following part.

2.1. State model

When a spacecraft is on a Mars GA flyby trajectory, its orbital motion can be described as a perturbed three-body problem with Mars as the central body. The dynamical model in the Mars-centered inertial frame (J2000.0) is written as

$$\begin{cases} \dot{x} = v_x + w_x \\ \dot{y} = v_y + w_y \\ \dot{z} = v_z + w_z \\ \dot{v}_x = -\mu_m x/r_{pm}^3 - \mu_s[(x-x_1)/r_{ps}^3 + x_1/r_{ms}^3] + w_{v_x} \\ \dot{v}_y = -\mu_m y/r_{pm}^3 - \mu_s[(y-y_1)/r_{ps}^3 + y_1/r_{ms}^3] + w_{v_y} \\ \dot{v}_z = -\mu_m z/r_{pm}^3 - \mu_s[(z-z_1)/r_{ps}^3 + z_1/r_{ms}^3] + w_{v_z} \end{cases} \tag{1}$$

where (x_1, y_1, z_1) and (x, y, z) are the position of the Sun and the spacecraft. μ_m and μ_s are the gravitational constants of Mars and the Sun. $\mathbf{r}_{ms}$ is the position vectors of the Sun relative to Mars. $\mathbf{r}_{pm}$ and $\mathbf{r}_{ps}$ are the position vectors of the spacecraft relative to Mars and the Sun, respectively. w_x, w_y, w_z, w_{v_x}, w_{v_y}, and w_{v_z} is the process noise, which comes from the perturbations including solar radiation pressure, spacecraft propulsion and other perturbations [27].

Assuming state variables $\mathbf{x} = [x, y, z, v_x, v_y, v_z]^T$ and process noise $\mathbf{w} = \left[w_x, w_y, w_z, w_{v_x}, w_{v_y}, w_{v_z}\right]^T$, the state model can be simply presented as

$$\dot{\boldsymbol{x}}(t) = f(\mathbf{x}, t) + \boldsymbol{w}(t) \tag{2}$$

2.2. Measurement model

The angles between Mars' moons (Deimos and Phobos) and the background stars are selected as measurements, which are shown in Fig. 1. The measurement model of θ_p and θ_d is given as

$$\begin{bmatrix} \theta_p \\ \theta_d \end{bmatrix} = \begin{bmatrix} \arccos\left(-\frac{\mathbf{r}_{pp} \times \mathbf{s}_1}{r_{pp}}\right) \\ \arccos\left(-\frac{\mathbf{r}_{pd} \times \mathbf{s}_2}{r_{pd}}\right) \end{bmatrix} + \begin{bmatrix} v_{\theta_p} \\ v_{\theta_d} \end{bmatrix} \tag{3}$$

where $\mathbf{s}_1$ and $\mathbf{s}_2$ are the position vector of the navigation stars in the Mars-centered inertial frame, which can be obtained from the star catalog by star identification and coordinate transformation. $\boldsymbol{r}_{pp}$ and $\boldsymbol{r}_{pd}$ are the position vectors of the spacecraft relative to Phobos and Deimos, respectively.