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Stellar sensor based nonlinear model error filter for gyroscope drift extraction

Jing Jin [1], Haiting Tian *, Chunxi Zhang [2], Ningfang Song [3], Song Lin [3]

Institute of Opto-Electronic Technology, Beihang University, No. 37 Xueyuan Road, Haidian Block, Beijing 100191, PR China

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ABSTRACT

The conventional gyroscope based spacecraft attitude determination approaches are afflicted by the integration drift of gyroscope. To ameliorate the performance of gyroscope, many compensation algorithms have been developed. A novel spacecraft attitude determination algorithm, which is based on the stellar sensor and nonlinear model error filter algorithm, is discussed in this paper. This algorithm uses the attitude quaternion obtained from stellar sensor to generate the attitude information of spacecraft, which could compensate the drift error of the gyroscope unit. This approach could be utilized both as the gyroscope error compensation in high precision integrate navigation system and as an attitude determination unit individually in low precision mini-satellite task. Two verifications of different satellite orbits have been launched and the performance of our approach was proved.

1. Introduction

With the development of human technique in aerospace exploration, many kinds of spacecrafts have been constructed, which include satellites, space shuttles, space ships and space stations. The related technology is becoming the most attractive realm of the human science. In the voyage of a spacecraft, the attitude control system that includes the navigation sensor, the attitude sensor, the thruster, the attitude control component and the main processor works as the human cerebellum to control the behavior of the spacecraft. Given the importance of the attitude control system, the research of this technique is in a particular focus.

The most prevalent attitude determination and navigation approaches currently are the inertial rate sensor (IRS), celestial vector sensor (CVS) and the global position system (GPS). The IRS works by continuously tracking the position, orientation and velocity (direction and speed of movement) of a vehicle without the aid of external references [1]. The GPS, which belongs to the radio navigation mode, can be used to bind the position error of inertial navigators. However, the integrated hybrid radio-inertial navigation system is no longer self-contained and without radiating. The GPS also could not provide service for the inter-planetary spacecraft voyage [2]. Instead, the CVS could satisfy the requirement of self-contained and high navigation precision [3]. With the improvement of spacecraft attitude determination, the integrated attitude determination (IAD) system, which combines the advantages of IRS, CVS and GPS, has been proposed and proved to be much better than the conventional single mode attitude determination approach [4,3].

As an important angular rate sensor of IRS, the gyroscope plays an important role in the traditional attitude determination system [1]. For more than 40 years' development, it has evolved as a large family, which includes conventional mechanical gyroscope, optic gyroscope and MEMS gyroscope [5]. Although an extremely high precision had been achieved, it still suffers from integration drift, which is inherent to all types of inertial navigation systems [6]. The integration drift is due to the accumulation of small drifts in the rotation rate measurements with time, and produces a large drift at final counting [5].

To eliminate or compensate the negative effect on the performance of gyroscope caused by the integration drift, many approaches have been proposed. As the gyroscope is a self-contained and no radiating navigation system, its error propagation cannot be separated by merely improving the precision of the sensor itself [6]. The common approach is to compensate the integration drift of gyroscope by importing the external reference information obtained from other navigation or attitude determination sensors. In the proposed CVS/IRS or GPS/IRS joint attitude determination systems, the integration drift could be compensated by importing the output of CVS or GPS, as there is no drift accumulation in their sensors [7,8].

Amongst these joint attitude determination approaches, the attitude determination based on stellar sensor (ADSS) is enchanting, as the stellar sensor has obvious merits in inter-planetary spacecraft applications. Compared with the other CVS devices, the

* Corresponding author. Tel.: +86 15101092397; fax: +86 10 82316887 818.
E-mail addresses: jinj@buaa.edu.cn (J. Jin), tsatnt@gmail.com (H. Tian).
zhangcx@buaa.edu.cn (C. Zhang), songnf@buaa.edu.cn (N. Song).
Lins@buaa.edu.cn (S. Lin).
[1] Tel.: +86 10 82316887 821; fax: +86 10 82316887 818.
[2] Tel.: +86 10 82316904; fax: +86 10 82316887 818.
[3] Tel.: +86 10 82316887 820; fax: +86 10 82316887 818.

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stellar sensor is advantageous for its high precision, solid state, high stable and wide workable range, which makes it to be suitable for deep space exploration [9]. In early days, the stellar sensor was limited by the process speed. But with the development of microelectronic technology, the performance of processor has been meliorated a lot and the image sampling time has been shortened, which have brought the stellar sensor a quick response time. Nevertheless, limited by the pattern of stellar sensor, it should avoid the shadow of planet and the eclipse of sun or moon, so it could hardly afford the mission in near surface orbit individually [10].

In this paper, a novel spacecraft attitude determination (SAD) algorithm, which compensates the gyroscope drift by importing the stellar sensor's quaternion output, is proposed. Furthermore, this SAD algorithm is based on the specified nonlinear model error filter (NMEF). The simulation result is listed at the end of the paper, which can reveal the performance of this algorithm. The key contributions of this paper are as follows:

- An introduction to the SAD fundamentals.
- A description of the NMEF-SAD algorithm.
- A simulation of the proposed algorithm using the spacecraft simulation software.

This paper is consisted of five sections. In Section 2, the related works are introduced. In Section 3, some fundamentals of the SAD are described. The NMEF-SAD algorithm is discussed in Section 4. The verification and the simulation are listed in Section 5, and conclusion is drawn in Section 6.

2. Related works

The related works are introduced in this section. In the traditional attitude determination, the Kalman filter has been proven to be extremely useful for attitude estimation using vector attitude measurements and gyro measurements. The essential feature of the Kalman filter is the application of state-space formulations in the system model design. For spacecraft attitude estimation, the Kalman filter is most applicable to spacecraft attitude determination with three-axis gyroscopes [11]. In the application of Gyroless SAD, the analytical models of rate motion was used [12]. This approach has been successfully used in a real-time sequential filter (RTSF) algorithm which propagates state estimates and error covariance using a dynamic model [12,13]. However, the RTSF is essentially a Kalman filter in which the gyro bias is modeled as a Gaussian process with known covariance. Also, fairly accurate models of angular momentum are required in order to obtain accurate estimates. Subsequently, the design process for choosing the model error covariance becomes difficult [13,14].

A new method of performing optimal state estimation in the presence of significant model error has been proposed by Mook and Junkins [15]. This method, called the minimum model error (MME) estimator, different from most filter and smoother algorithms, does not presume that the model error is represented by a Gaussian process. Instead, the model error is determined during the MME estimation process [16]. Therefore, accurate state estimates can be determined without the use of precise system representations in the assumed model. This algorithm has been successfully used to estimate the attitude of an actual spacecraft without the utilization of gyro measurements [16]. However, the MME estimator is a batch (off-line) estimator which must utilize post experiment measurements [16].

Another approach in gyroless application SAD is the nonlinear predictive filter, which combines the good qualities of both the Kalman filter and the MME estimator [14]. The new algorithm is based on a predictive tracking scheme introduced by Prof. Lu in his publication [17]. The predictive filter algorithm developed in Prof. J. Crassidis's paper is reformulated as a filter and estimator with a stochastic measurement process [14]. Another interesting research of this predictive filter is presented by Dr. Yurong Lin in his paper [18].

The NMEF-SAD algorithm mentioned in this paper is based on the quaternion attitude determination algorithm, and blends the advantage of MME. This algorithm needs fewer parameters from the spacecraft kinematics equation than the predictive filter algorithm; nevertheless, a similar precision could be achieved.

3. SAD fundamental

To make a clearer understanding of the NMEF-SAD algorithm for readers, several fundamentals of SAD are introduced in this section. These concepts are the reference coordinates system (RCS) in space voyage and the space kinematics equations.

3.1. Reference coordinates system

As a fundamental, the RCS is crucial in space vehicle attitude determination. A precise Reference Coordinates can make the navigation or attitude calculation more accurate. The earth centered inertial coordinates system (ECICS) is commonly treated as the RCS in space trip. Furthermore, there are also many other RCS algorithms in different applications, such as the vehicle body coordinates system (VBCS), the trajectory coordinates system (TCS) and the geodetic coordinates system (GCS) [19]. The RCS mentioned in this paper is J2000 Reference Coordinates, which is a sub-type of ECICS and defined as the X and Z axes point toward the mean vernal equinox and mean rotation axes of the Earth on January 1, 2000 at 12:00:00.00 universal time coordinated (UTC). J2000.0 = 2000 January 1.5 = JD2451545.0 terrestrial dynamical time (TDT) [20].

The measurement output of Star Sensor shows the attitude angular from the stellar sensor body coordinates system (SSBCS) to the ECICS, and is always exported as the normalized quaternion vector q, and $q=[q_0,q_1,q_2,q_3]^T$; the conversion matrix between SSBCS and VBCS is C_{ss}. The gyroscope output shows the rotation angular rate, projected in FBCS, from the gyroscope body coordinates system (FBCS) to the ECICS, and is expressed as the rotation angular rate vector ω, and $\omega=[\omega_x,\omega_y,\omega_z]^T$; the conversion matrix between FBCS and VBCS is C_{gs}. For the simplification of description, it is assumed that SSBCS, FBCS and VBCS are the same, i.e. $C_{ss}=C_{gs}=I$. The related knowledge of Reference Frames transformation could be found in paper [21].

3.2. Spacecraft kinematics

The simplified three-axis stabilized spacecraft kinematics equation with attitude quaternion and rotation rate is expressed as follows:

$$\begin{cases} \dot{q}(t)=\dfrac{1}{2}\Omega\bullet q(t) \\ \Omega=\begin{bmatrix} 0 & \omega_z(t) & -\omega_y(t) & \omega_x(t) \\ -\omega_z(t) & 0 & \omega_x(t) & \omega_y(t) \\ \omega_y(t) & -\omega_x(t) & 0 & \omega_z(t) \\ -\omega_x(t) & -\omega_y(t) & -\omega_z(t) & 0 \end{bmatrix} \end{cases} \tag{1}$$

The vector $\dot{q}(t)=[\dot{q}_0(t),\dot{q}_1(t),\dot{q}_2(t),\dot{q}_3(t)]^T$ is the derivative of spacecraft kinematics quaternion, and Ω is the transition matrix, which consists of the vehicle angular rates. The spacecraft attitude