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Pico-satellite Autonomous Navigation with Magnetometer and Sun Sensor Data

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Abstract

This article presents a near-Earth satellite orbit estimation method for pico-satellite applications with light-weight and low-power requirements. The method provides orbit information autonomously from magnetometer and sun sensor, with an extended Kalman filter (EKF). Real-time position/velocity parameters are estimated with attitude independently from two quantities: the measured magnitude of the Earth's magnetic field, and the measured dot product of the magnetic field vector and the sun vector. To guarantee the filter's effectiveness, it is recommended that the sun sensor should at least have the same level of accuracy as magnetometer. Furthermore, to reduce filter's computation expense, simplification methods in EKF's Jacobian calculations are introduced and testified, and a polynomial model for fast magnetic field calculation is developed. With these methods, 50% of the computation expense in dynamic model propagation and 80% of the computation burden in measurement model calculation can be reduced. Tested with simulation data and compared with original magnetometer-only methods, filter achieves faster convergence and higher accuracy by 75% and 30% respectively, and the suggested simplification methods are proved to be harmless to filter's estimation performance.

Keywords: pico-satellite; autonomous navigation; orbit estimation; magnetometer; Kalman filter

1. Introduction

Recently, considerable effort has been invested in research and development programs for micro-technology for space applications[1]. Pico-satellite research is one of these programs. The development of a satellite weighing only about 1 kg may significantly reduce space vehicle launch costs, as well as the manufacturing time. Until January 2008, more than 30 different pico-satellites had been launched into space[2], and after that, more pico-satellites are developing. By far, there are over 100 research institutions participating in the CubeSat pico-satellite project[3], thus it is very clear that there will be more and more pico-satellites launched into space in the near future. This trend, however, may lead to some problems in the traditional ground station-based orbit determination systems. To most satellites, knowledge of orbit and position is essential in operating in-orbit missions. To get this information, traditional ways rely on ground station-based range and range-rate data[4]. Because the number of satellite ground stations is limited, these traditional ground station-based ways may not satisfy the incoming requirements of excessive pico-satellites launches, not to mention many of them are only available for higher priority space missions. Compared with those, the global positioning system (GPS) is one of the most popular ways in determining spacecraft orbit. However, to get the best possible accuracy, differential GPS is needed, which still relies on ground-based measurements[5]. Furthermore, to pico-satellite, as its power budget is always austere, the installation of GPS

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equipment may cause competition with other design functions.

Magnetometer is one of the common sensors used in pico-satellites[6]. It is lightweight, reliable, and has low-power requirement, with no moving parts. As it provides both magnitude and direction of magnetic field, which is relevant to planet's geographic position, magnetometer can be used for orbit determinations. This is a completely autonomous way, as it can work without any intervention from a ground station or space systems. Autonomous orbit determination of spacecraft using magnetometers has been studied and verified using real-flight data in many previous works. Methods including batch filter[7-9], extended Kalman filter (EKF) [10-15], unscented Kalman filter (UKF)[16-18] are developed and discussed. These methods, however, still need improvements. Batch filter can estimate satellite orbit along with some other model coefficients[7-9], but it is not a proper method for real-time navigation, as the data used in filters should be collected for days. Kalman filter-based methods can be divided into two categories: the attitude dependent methods[13-15,18] and the attitude independent methods[10-12,16-17]. The former methods can be used to estimate both satellite orbit and attitude, and it is reported with quick convergence and good accuracy when tested with real-flight data[13]. The computation expense, however, might be unbearable. During an in-flight experiment, it is reported that the onboard processor operated frequently at its full capacity[14]. The latter methods, i.e., the attitude independent methods, estimate orbit without knowledge of satellite attitude. Good accuracy can be achieved; also the computation time might be acceptable[10]. The convergent rate, however, is very slow. Tested with real-flight data, it takes about 12 h for filter to complete convergence[12].

Nevertheless, for pico-satellite, magnetometer based estimation are still effective methods, as they meets the requirements of being low-power, light-weighted as well as autonomous. In this article, several further improvements have been discussed and introduced. To achieve faster convergence and higher accuracy, the possibility of including sun sensor data in measurements is discussed, and a newly designed EKF has been introduced. The filter is attitude independent, and results in better performance in simulation compared with original magnetometer-only algorithms. Additionally, to reduce computing burden, several techniques are discussed and suggested. Simplification methods in EKF's Jacobian calculations are introduced and testified, and a polynomial model for fast magnetic field calculation is developed, which can be used as a full replacement of IGRF model for several months.

2. Filter Design

In this section, an EKF has been implemented. Different from current EKF algorithms, this filter estimates satellite orbit with one additional sensor, the sun sensor. Besides the measured magnitude of the Earth's magnetic field, the measured dot product of the magnetic field vector and the sun direction vector is also used as filter measurement.

2.1. State vector

Generally, the orbit element set used in orbit estimation filter is usually defined in three ways: the Cartesian coordinates[12,16], the Keplerian elements[11] and the geographic form[10]. Each element set has its own advantages. The Cartesian form is often used in numerical integration, the Keplerian form is better in orbit description, and the geographic form is convenient in magnetic field calculation. For low altitude orbit, Earth's atmosphere density cannot be ignored. In order to achieve higher accuracy, the effect of atmospheric drag needs to be modeled, thus the ballistic coefficient of satellite needs to be estimated. Furthermore, calibration elements, such as measurement bias, can also be estimated[11]. In this design, all sensors are considered to be pre-calibrated, and satellite operates in a low earth orbit, therefore the filter's state vector is defined as

$$\boldsymbol{x} = [\boldsymbol{r}^{\mathrm{T}} \quad \boldsymbol{v}^{\mathrm{T}} \quad B^*] \tag{1}$$

where $\boldsymbol{r}$ and $\boldsymbol{v}$ are the position and velocity vectors in inertial frame, and B^* is the inverse value of satellite's ballistic coefficient, which is the multiplication of the drag coefficient and the area-to-mass ratio.

2.2. Dynamic model

For real-time applications, the dynamic model of satellite orbital motion should be defined carefully. A precise dynamic model may increase filter accuracy with unacceptable computational burden, while an over simplified model would be just the opposite. Based on conclusions of Roh's work[16], the orbital dynamic equations adopted in this work are

$$\dot{\boldsymbol{r}} = \boldsymbol{v} \tag{2}$$

$$\dot{\boldsymbol{v}} = \boldsymbol{a}_{\mathrm{geo}} + \boldsymbol{a}_{\mathrm{drag}} + \boldsymbol{w}_1 \tag{3}$$

$$\dot{B}^* = w_2 \tag{4}$$

where $\boldsymbol{a}_{\mathrm{geo}}$ is the geopotential acceleration, $\boldsymbol{a}_{\mathrm{drag}}$ the acceleration due to atmospheric drag. $\boldsymbol{w}_1$ and w_2 are system process errors, which can be approximated as zero-mean Gaussian noise.

For state propagation, Eqs.(2)-(4) can also be written as

$$\dot{\boldsymbol{x}} = f(\boldsymbol{x}) + \boldsymbol{w} \tag{5}$$

where $\boldsymbol{w}$ is a combination of $\boldsymbol{w}_1$ and w_2.

$$E(\boldsymbol{w}\boldsymbol{w}^{\mathrm{T}}) = \boldsymbol{Q} \tag{6}$$

where $\boldsymbol{Q}$ is the system noise covariance matrix.

The Jacobian calculation of Eq.(5) can be written as