Micro Attitude Determination System Based on MEMS Inertial Sensors

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Abstract: In this paper, a micro attitude determination system with a novel attitude estimating algorithm used among the aerial field is proposed based on low cost MEMS inertial sensors, including MEMS rate gyros, MEMS accelerometer and MEMS magnetometers. This paper presents the attitude estimating algorithm based on the error quaternion formulation, whereby the error quaternion is selected to compose the state vector that is updated by output of MEMS rate gyros. In order to estimate the attitude during high dynamic maneuvers, the earth's magnetic field and the effective part of gravity along the motion direction as the two measured quantities are introduced as virtual measurements in the extended Kalman filter (EKF) which decreases short-term errors. Simulation and flight experimental results indicate that attitude quaternion is determined with standard deviations below 5° even in high dynamics and 0.7° in static state.

Keywords: low cost; MEMS inertial sensors; attitude determination; Kalman filter

基于 MEMS 惯性传感器的微型姿态测量系统

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摘 要: 提出了一种基于低成本 MEMS 惯性传感器的微型姿态测量系统,包括 MEMS 速率陀螺、MEMS 磁强计、单轴 MEMS 加速度传感器. 重点研究了基于扩展 Kalman 滤波(EKF)的姿态估计创新算法,通过速率陀螺更新误差状态四元数计算姿态角,并通过飞行方向的加速度传感器和三轴磁强计来补偿陀螺漂移和姿态角误差,利用扩展卡尔曼滤波方程消除瞬时干扰,实现高动态姿态测量. 系统的仿真和高动态实验表明,姿态测量动态精度低于 5°,静态精度低于 0.7°.

关键词: 低成本; MEMS 惯性传感器; 姿态测量; 卡尔曼滤波
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A large amount of effort has been directed at developing low cost attitude determination systems. For example, Refs. [1-2] introduce attitude heading reference system (AHRS) using gyros, accelerometers and magnetometers. A low pass filter is used to increase the accuracy. An inexpensive attitude determination system is discussed for aviation applications in Ref. [3]. This attitude determination system does not employ any inertial sensors, but

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relies on a kinematic model of the vehicle with GPS positioning and velocity measurements derived from a single GPS antenna. Some similar attitude algorithms are presented by constructing baseline utilizing different GPS and multi-GPS and aided by gyros etc. To improve the performance in high dynamic situation, an adaptive extended Kalman filter is adopted by Refs. [4-5] where the filter tunes its gain automatically based on the system dynamics to yield optimal performance. Ref. [6] presents a gyro-free attitude determination algorithm with low cost MEMS sensors, and the attitude could be estimated through constructing the quaternion state and measurement equation by measurement of earth magnetic field and gravity acceleration.

Practically vector matching in two coordinate frames is often referred to as Wahba's problem in 1965. Wahba proposed an attitude solution by matching two non-zero, non-colinear vectors that were measured in two different coordinate frames[7-8].

In this paper, a novel method to solve Wahba's problem is demonstrated, in which the attitude error equation can be cast into a standard form, and measurement equation of gravity acceleration and magnetic field can be constructed in linear form. To improve the performance in high dynamic maneuvers, the extended Kalman filter(EKF) is used. In section 1 of this paper, the configuration of the micro attitude determination system is illustrated. In section 2, the attitude determination algorithm is derived step by step. In section 3, the algorithm and the Kalman filter implementation of solution are verified by simulation and experiments in aerial flight. Finally, conclusions are summarized in section 4.

1 System configuration

The system configuration is very important in the design of micro attitude determination prototypes since characteristics of the onboard components, such as weight, size, and performance of MEMS inertial sensors, significantly affect the error of the attitude estimating. Fig. 1 shows the system configuration.

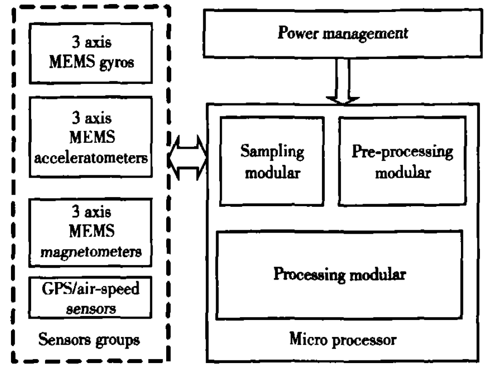

Fig. 1 Micro attitude determination system framework

2 Attitude determination algorithm

Wahba's attitude solution, attitude determination from non-zero, non-colinear vectors matching, requires two or more vectors to be known or measured in the respective coordinate frames. In short-term sampling period, the attitude quaternion could be calculated and updated by the measurement of gyros, and the error quaternion and gyro bias could be compensated by the measurement of magnetic field and gravity acceleration vector along the motion direction. With EKF, the instant disturbance in high dynamic maneuvering could be reduced. The initial attitude could be obtained by measuring earth magnetic field and gravity acceleration in static state[9].

2.1 Coordinate frames definition

Define the two orthogonal coordinate frames as the "body frame" (fixed to the vehicle's body and denoted by superscript b) and the "navigation frame" (attached to the local level plane and denoted by superscript e or n). The transformation between the vector $\boldsymbol{u}_b$ expressed in body frame and $\boldsymbol{u}_n$ expressed in the navigation frame is

$$\boldsymbol{u}_b = \boldsymbol{C}_n^b(\boldsymbol{q})\boldsymbol{u}_n \tag{1}$$

where $\boldsymbol{C}_n^b(\boldsymbol{q})$ is the direct cosine matrix(DCM) transforming from navigation frame to body frame. By the matrix difference equations, Eq. (1) could be written as

$$\dot{\boldsymbol{q}} = \frac{1}{2}\bar{\boldsymbol{\omega}}_b\boldsymbol{q} \tag{2}$$

where $\bar{\boldsymbol{\omega}}_b$ is the angle rate expressed in the body frame and measured by MEMS gyro equipped on the carrier. In terms of attitude quaternion, the DCM could be written as

$$C_n^b = \begin{bmatrix} q_0^2+q_1^2-q_2^2-q_3^2 & 2(q_1q_2+q_0q_3) & 2(q_1q_3-q_0q_2) \\ 2(q_1q_2-q_0q_3) & q_0^2-q_1^2+q_2^2-q_3^2 & 2(q_2q_3+q_0q_1) \\ 2(q_1q_3+q_0q_2) & 2(q_2q_3-q_0q_1) & q_0^2-q_1^2-q_2^2+q_3^2 \end{bmatrix} \tag{3}$$

By DCM with attitude quaternion, Euler angles can be calculated with Eq. (3).

2.2 Error quaternion, state equation and gyro model

In this paper, let $\hat{\bar{q}}$ be an estimate of the true attitude quaternion defined as $\bar{q}$. The attitude quaternion can be composed of two parts: one is defined as $\hat{\bar{q}}$, and the other is the error rotation from the estimated quaternion $\hat{\bar{q}}$ to $\bar{q}$, which is defined as $\bar{q}_e$. The relationship is expressed in terms of quaternion multiplication as follows:

$$\bar{q} = \hat{\bar{q}} \otimes \bar{q}_e \tag{4}$$

The error quaternion, $\bar{q}_e$, is assumed to represent a small rotation, thus $\bar{q}_e$ can be approximated as

$$\bar{q}_e = [1, q_e] \tag{5}$$

Let Eq. (4) be used in a different way from Eqs. (2) and (5), the relationship between $\bar{q}_e$ and $\hat{\omega}_b$ is obtained in the following equation which ignores the high level error and maintains the vector part:

$$\dot{q}_e = -\hat{\omega}_b q_e - \frac{1}{2}(\omega_b - \hat{\omega}_b) \tag{6}$$

where ω_b is the angle rate defined before, and $\hat{\omega}_b$ is the estimate of the ω_b expressed in body frame. The model of the output of rate gyro is defined as follows:

$$\hat{\omega}_b - \hat{b} = \omega_b - b + W_1 \tag{7}$$

where ω_b is denoted as the output of gyro; $\hat{\omega}_b$ is denoted as the estimate of gyro output; W_1 is stochastic noise of gyro assumed as Gaussian white noise with the zero mean.

Thus Eq. (7) could be written as

$$\omega_b - \hat{\omega}_b = \Delta\hat{b} - W_1 \tag{8}$$

where $\Delta\hat{b}$ is defined as $\hat{b} - b$. A simple and realistic stochastic model of rate gyro can be written as

$$\Delta\dot{\hat{b}} = W_2 \tag{9}$$

where W_2 is the Gaussian white noise and independent on W_1. Define a state variable X and noise variable W that represent the error attitude quaternion as follows:

$$X = \begin{bmatrix} q_e \\ \Delta b \end{bmatrix} \qquad W = \begin{bmatrix} W_1 \\ W_2 \end{bmatrix} \tag{10}$$

In terms of Eq. (6) and gyro model, the state equation could be expressed as

$$\dot{X}(t) = F(t)X(t) + G(t)W(t) \tag{11}$$

where $F(t)$ and $G(t)$ are given by

$$F(t) = \begin{bmatrix} -\hat{\omega}_b & \frac{1}{2}I_{3\times3} \\ 0_{3\times3} & 0_{3\times3} \end{bmatrix}$$

$$G(t) = \begin{bmatrix} -\frac{1}{2}I_{3\times3} & 0_{3\times3} \\ 0_{3\times3} & I_{3\times3} \end{bmatrix}$$

2.3 Measurement equation

In this paper, magnetic field and gravity acceleration vector along the motion direction are measured to update the attitude quaternion.

Since the vector components of $\bar{q}_e$ are small, the perturbation to the DCM in Eq. (3) can be written as

$$C_n^b(\bar{q}_e) = \begin{bmatrix} 1 & 2q_{e3} & -2q_{e2} \\ -2q_{e3} & 1 & 2q_{e1} \\ 2q_{e2} & -2q_{e1} & 1 \end{bmatrix} \tag{12}$$

In terms of Eq. (1) and Eq. (12), the relationship of magnetic vector between the body frame and the navigation frame is written as

$$m_b^0 = (I - 2q_e)C_n^b(\hat{\bar{q}})m_n^0 + V_m \tag{13}$$

where V_m is the measurement error assumed to be zero mean Gaussian noises with variance V_1.

In the navigation frame, the magnetic vector and gravity acceleration could be expressed as $m_n^0 = [\cos\beta, 0, \sin\beta]^T$ and $a_n^0 = [0,0,1]^T$, where β is a local geomagnetic inclination.

Let the following be made: $\delta m_b = m_b - C_n^b(\hat{\bar{q}})m_n^0$, and $\hat{m}_b = C_n^b(\hat{\bar{q}})m_n^0$

Then the measurement equation can be obtained as:

$$\delta\hat{m}^b = 2\hat{m}_b q_e + V_m(t) \tag{14}$$

And a similar equation of gravity acceleration is expressed as follows:

$$\delta\hat{a}^b_{1\times1} = 2\hat{a}_{b1\times3} q_e + V_a(t)_{1\times1} \tag{15}$$

where $V_a(t)$ is the measurement error assumed to be zero mean Gaussian noises with variance V_2.

From Eqs. (14) and (15), the measurement equation can be represented as follows:

$$\begin{bmatrix} \delta\hat{a}^b_{1\times1} \\ \delta\hat{m}^b_{3\times1} \end{bmatrix} = \begin{bmatrix} 2\hat{a}_{b1\times3} \\ 2\hat{m}_{b3\times3} \end{bmatrix} q_e + \begin{bmatrix} V_{a1\times1} \\ V_{m3\times1} \end{bmatrix} \tag{16}$$

2.4 EKF

Eqs. (11) and (16) construct an EKF[10] to estimate the elements of the error quaternion, which can be used to derive the attitude angle. Fig. 2 shows how the two filters provide the estimation of attitude angles and gyro bias.

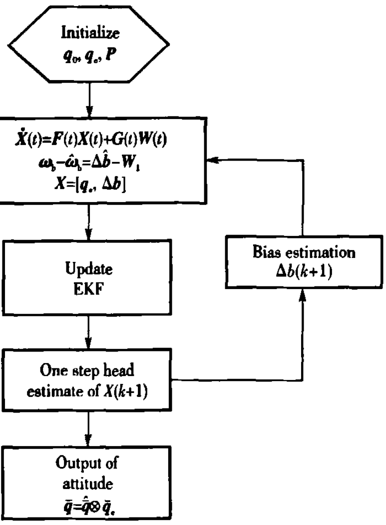

Fig. 2 Flow chart of the EKF

3 Simulation and experimental results

Random initial conditions are given to confirm convergence of iterations: $q_0=[1,0,0,0]$, $\bar{q}_{e0}=[1,0,0,0]$, and P_0 is the matrix full of 1.

Gaussian white noise with variance 0.3 $(\mathrm{m}\cdot\mathrm{s}^{-2})^2/\mathrm{Hz}$ was assigned to the true measurement of heading axis of gravity acceleration. Other axes of gravity could be any values with standard deviation 10 $(\mathrm{m}\cdot\mathrm{s}^{-2})^2/\mathrm{Hz}$ because they could not affect the filter at all. Random noises with standard deviation 0.006 $\mathrm{gauss}^2/\mathrm{Hz}$ were added to the outputs of magnetometers. Random noises and large drifts were appended to the angular rate signals. The sampling and calculating interval was 0.04 s. Tab. 1 shows that the attitude estimation errors were considerable low. Fig. 3 shows the estimated attitude angle.

Tab. 1 Attitude error statistics

Eular	Pitch/(°)	Roll/(°)	Yaw/(°)
Standard deviation	2.29	1.89	4.86

To verify the attitude algorithm in high dynamic situation, the micro attitude determination system is equipped in unmanned aerial vehicle (UAV) for flight tests. Fig. 4 shows the attitude estimate in flight.

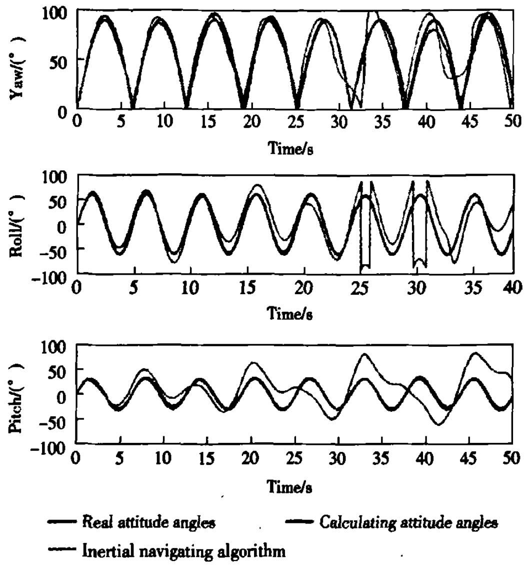

Fig. 3 Attitude and gyro drift estimation

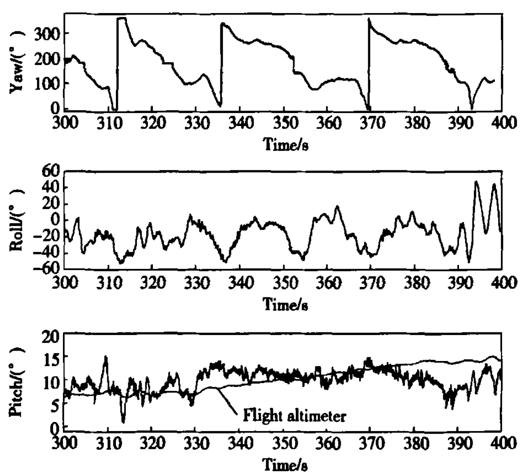

Fig. 4 Attitude estimate in flight

4 Conclusions

In classical MEMS based attitude determination systems, the kinetic accelerations and bias of gyros have limited their applications strictly. To cope with this problem, a novel attitude estimating algorithm is proposed in this paper, which is constructed with error quaternion state equation and measurement equation by MEMS inertial sensor in terms of high dynamic situation. It has eliminated the effect of kinetic accelerations except the motion axis in body frame. An EKF is used to estimate the attitude error quaternion and reduce the errors. Simultaneity gyros' bias is estimated and it reduces the measurement error of gyros. A simulation test with enough random noises has been conducted, whose performance has shown that the method is effective and feasible. Results indicate that quaternion is determined with standard deviations below 5° even in high dynamics and 0.7° in static state. Flight experiments based on MEMS sensors will be taken soon and the ultimate goal is to implement the attitude determination in UAV.

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