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Procedia Engineering 15 (2011) 959 – 963

Procedia Engineering

www.elsevier.com/locate/procedia

Advanced in Control Engineering and Information Science

Attitude Determination of Nano Satellite Based on Gyroscope, Sun Sensor and Magnetometer

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Abstract

With the fast development of nano-satellite applications, higher requirements on the method and accuracy for determining attitude of Nano-satellite are presented. Aiming at the accuracy requirements of attitude determination of sun-oriented mode of the nano-satellite, an integrated attitude sensor is built with combining of gyroscope, magnetometer and sun sensor. Furthermore, the extended Kalman filter is used to fuse measurement data from gyroscope, magnetometer and sun sensor. A simulation model of attitude determination is built, and a few simulations are performed. The simulation results show that the accuracy of the integrated attitude determination sensor proposed in this paper can meet the accuracy requirement (0.05. Deg.) of nano-satellite attitude determination.

Selection and/or peer-review under responsibility of [CEIS 2011]

Keywords: nano-satellite; Attitude determination; Optic gyroscope; Sun sensor; Magnetometer

1. Introduction

Nano-satellite is an outstanding satellite technology in astronautics field because of its light weight, short development cycle, low development cost, high functional density and flexible emission characteristics. Satellite attitude determination sensor is a core part in the nano-satellite, which directly

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doi:10.1016/j.proeng.2011.08.177

affects the performance of the satellite and satellite payloads. On one hand, the attitude determination sensor can provide information feedback for the attitude control system to control the attitude of the satellite. On the other hand, it can be provided for the payloads to use. At present, attitude determination sensors which were applied in launched satellites are star sensor, sun sensor, magnetometer, gyroscope, and so on, but all of those sensors are hard to meet the requirements of Nano-satellite's attitude determination. The measurement accuracy of gyroscope that is used to measure angular velocity is higher in short period, and could output continuous attitude information. But there have error accumulation because of gyroscope drift in long time. Angular-measurement sensors such as star sensor, sun sensor and magnetometer do not have the problem of error accumulation, but they cannot output attitude information and the data update frequency is slow. Therefore, to improve the attitude determination accuracy of Nano-satellite, an integrated attitude determination sensor combining angle sensors with angular velocity sensors was built. In this way, the angle sensors the angular velocity sensors can make up the advantages and disadvantages to each other. The sensor includes gyroscope, sun sensor and magnetometer. Furthermore, the extended Kalman filter was used to fuse measurement data from gyroscope, magnetometer and sun sensor. A simulation model of attitude determination was built, and a few simulations were performed.

2. Measurement Model of Sensor

In order to establish an integrated attitude determination model, the principle and mathematical model of the gyroscope, sun sensor and magnetometer is proposed in this part.

2.1. Gyroscope

The exporting angular velocity of gyroscope is the one of satellite body relative to inertial space when gyroscope measurement axis installs along the inertial axis of satellite body. Gyroscope systematical drift could be compensated by testing and analyzing to determine each drift coefficient in the mathematical model. The gyroscope exporting mathematical model established in this paper is

$$\hat{\omega}_{ib}^{b} = \omega_{ib}^{b} + b + v \tag{1}$$

Where ω_{ib}^{b} is the real angular velocity of satellite relative to inertial space, b is gyroscope drift and v is gyroscope white noise.

2.2. Magnetometer

Magnetic field is a vector field, which is the public resource of the earth. The Magnetic field strength vector of any point in geospace is different from that of another, and corresponds to the latitude and longitude of the point. Therefore, we can determine the attitude of a satellite by precisely determining the position of every point in geospace, then inquiring magnetic table so as to get the Magnetic field vector in the terrestrial coordinate system, which can be used to conduct the solution of filtering together with the magnetic field vector detected by the magnetometer in the satellite coordinate system.

According to the Transformation matrixes among geographic coordinate system, earth fixed coordinate system, geocentric inertial system, geocentric orbit system and satellite coordinate system, the Magnetic field under the coordinate system of the satellite is:

$$B_b = T_o^b T_i^o T_e^i T_t^e B_t \tag{2}$$

where T is the transformation matrixes among the coordinate systems, and the attitude angle of the system of three-axis stabilized satellite relative to the satellite orbit coordinate system is included in the transformation matrix T_o^b .

When the measurement axis of magnetometer is fixed along the principal axis of inertia of satellite, the output of magnetometer is the vector $\hat{B}_b$ of earth's magnetic field in the satellite coordinate system:

$$\hat{B}_b = B_b + \upsilon \tag{3}$$

where B_b is the true value of magnetic field in the satellite coordinate system, while ν is the measurement error of magnetometer.

2.3. Sun sensor

Sun sensor is a kind of optical attitude sensor, which is used to get the orientation information of a spacecraft relative to the sun by determining the position of sun vector in the satellite coordinate system through sensing the position of sun vector. When the satellite is in a certain position, the vector of the sun in the satellite orbit coordinate system is S_o , and the output vector of the sun by the sun sensor is S_b , therefore

$$S_b = T_o^b S_o + \nu \tag{4}$$

where ν is the Measurement error of a sun sensor.

3. Model of Attitude Measurement

After the detection of the initial attitude, a satellite starts scientific detections during its in-orbit stage. In this mode the accuracy of the satellite attitude detection is required to be 0.05°, and the angular rate should be smaller than 0.2°/s. Angular rate is a standard used to judge whether a satellite is working normally, and we usually think it not when its angular rate is over 0.2°/s.

When the attitude of a nano-satellite has been determined by the combination of Gyroscope, Sun sensor, Magnetometer, State variables are the quaternion of the nano-satellite attitude and gyro drift, namely $X = \begin{bmatrix} q_0 & q_1 & q_2 & q_3 & b_x & b_y & b_z \end{bmatrix}^T$。

Suppose ω_{ob} is the angular rate of satellite coordinate system relative to orbit coordinate system, ω_{gyo} is the angular rate outputted by gyro drift, ω_{io} is the angular rate of satellite orbit coordinate system relative to Inertial coordinate system, b is gyro drift, ξ is gyro noise, and then we have $\omega_{ob} = \omega_{gyo} - \omega_{io} - b - \xi$. According to the equations about Satellite attitude kinematics and gyro drift features [6],

$$\dot{q} = \frac{1}{2} q \otimes \omega_{ob} \tag{5}$$

$$\dot{b} = 0 \tag{6}$$

So the system state equation is:

$$\begin{bmatrix} \dot{q} \\ \dot{b} \end{bmatrix} = f(q, b) \tag{7}$$

When measuring with sun sensor and magnetometer, we can get quaternion q which is the observed quantity by the means of q_method. The measurement equation is

$$q = \begin{bmatrix} 1 & 0 & 0 & 0 & 0 & 0 & 0 \\ 0 & 1 & 0 & 0 & 0 & 0 & 0 \\ 0 & 0 & 1 & 0 & 0 & 0 & 0 \\ 0 & 0 & 0 & 1 & 0 & 0 & 0 \end{bmatrix} X + \upsilon \tag{8}$$

where v is the measurement error caused by the sensor.

The nonlinear state equation and measurement equations of EKF are：

$$\dot{X} = F(X) + W \tag{9}$$

$$Z = h(X) + V \tag{10}$$

where X is state vector, Z is measurement vector, while W and V are white noise of system and measuring respectively.

Suppose $\Phi = I + \frac{\partial F(X)}{\partial X} + \partial[\frac{\partial F(X)}{\partial X} F(X)] \Big/ \partial X$, $H = \frac{\partial h(X)}{\partial X}$,

EKF formulas are

$$X_k = X_{k/k-1} + K[Z_k - h(X_{k/k-1})] \tag{11}$$

$$X_{k/k-1} = X_k + F(X_k)T + [\frac{\partial F(X_k)}{\partial (X_k)} F(X_k)]\frac{T^2}{2} \tag{12}$$

$$P_k = (I - KH) P_{k/k-1} (I - KH)^{\mathrm{T}} + KRK^{\mathrm{T}} \tag{13}$$

$$K = P_{k/k-1} H^T \left(H P_{k/k-1} H^T + R \right) \tag{14}$$

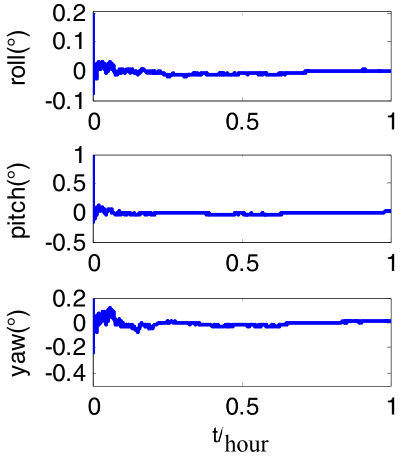
$$P_{k/k-1} = \Phi P_{k-1} \Phi^T + Q \tag{15}$$

State variables X of the system can be calculated by the equations listed above given the initial conditions X_0 and P_0, and therefore we can get the attitude information of the nano-satellite.

4. Experiment

According to the current development level of gyroscope, magnetometer and sun sensor, we assume that the magnetometer can achieve a measurement accuracy of 500nT, while the gyroscope possesses a drift of 0.5°/h and the sun sensor's accuracy is 0.05°. The initial parameters of filter solving are as follows: ① the initial quaternion is *[1 0.01 0.01 0.01]*^{T}, and the initial value of the gyroscope drift is *[1 ° /h 1 ° /h 1 ° /h]*^{T}; ② the initial covariance matrix is *diag [0.022 0.022 0.022 0.022 (1 ° /h) 2 (1 ° /h) 2 (1 ° /h) 2]*; ③ the output frequency of gyroscope is 100Hz, while that of the magnetometer is 10Hz; ④ the system covariance matrix are related to the gyroscope noise and the attitude quaternion, and will update in real-time filtering; ⑤ the measurement covariance matrix is *diag [(1°) 2 (1°) 2 (1°) 2 (1°) 2]*, and the simulation time is 3600s. Using EKF filtering method on the attitude determination of nano-satellites, the simulation results are shown in Figure 1 and Figure 2.

Figure 1 shows the estimation error of integrated attitude determination using gyroscope, sun sensor and magnetometer. In the simulation it is found that the estimated attitude angle error of the integrated attitude determination can converge to 0.05° within 20 minutes, while the roll angle and the yaw angle will achieve a final accuracy better than 0.02° and the pitch angle will achieve a final accuracy better than 0.05°. Figure 2 shows the estimated gyroscope drift under the integrated attitude determination and the constant drift is 0.5°/h, while after 15 minutes the estimated drift error will be less than 0.05°/h and the final drift accuracy will be better than 0.05°/h.



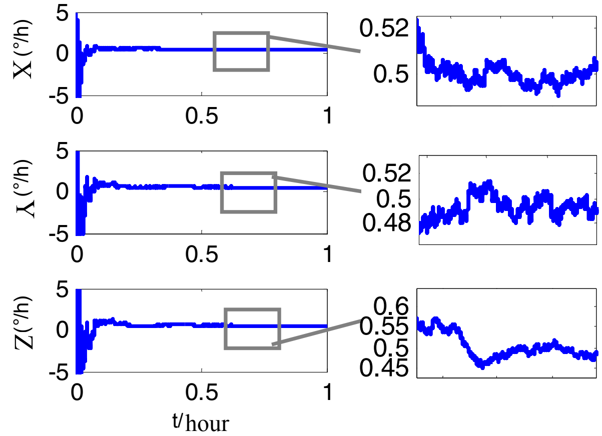


Figure1 gyroscope(0.5°/h)/ magnetometer(500nT) / sun sensor(0.05°)estimated attitude error

Figure2 gyroscope(0.5°/h)/ / magnetometer(500nT) / sun sensor(0.05°) estimated gyroscope drift

The results of the above simulation show that with sensors of such accuracy and the EKF integrated attitude determination algorithm, the accuracy of 0.05° in the three-axis attitude determination on nano-satellite can be achieved.

5. Conclusion

According to the requirements of three-axis stabilization on nano-satellite and that of attitude determination accuracy on sun-oriented mode of in-orbit nano-satellites, this paper designs an integrated attitude determination method for the sensor under sun-oriented mode and proceeds related stimulation under different configuration of gyroscope, magnetometer and star sensor, each with different accuracy. The results show that the attitude determination method we propose is feasible, which can meet the accuracy requirements of 0.05° for the three-axis stabilization of nano-satellite attitude determination.

References

[1] Fan Xu. The Attitude Control System Design and Simulation Study of Microsatellite. Harbin: Harbin Institute of Technology, 2006.

[2] Cui Zhang. Attitude Determination and Experimental Research Based on Magnetometer, Celestial Sensor and Gyro. Beijing: Beijing University of Aeronautics and Astronautics, 2010.

[3] Dongqing Gu, Yan Wang, Wenjun Zhou, Yongyuan Qing. Federated Filter Design for Multi-sensors Satellite Attitude Determination. Chinese Space Science and Technology, 2004, 6(3):7-13.

[4] Yong Qian. Studies on the Attitude Determination and Control System of High Precision Three-Axis Stabilized Satellite [D]. Xian: Northwestern Polytechnical University, 2002.

[5] Deut schmann (Thienel now) and R.R. Harman. Results of the Magnetometer Navigation (MAGNAV) in Flight Experiment[C]. AIAA/AAS Astro-dynamics Specialist Conference and Exhibit, Providence, Rhode Island, 16-19 August 2004:1-17.

[6] Tingguo Pei, Shilu Chen, Xinggang Li. Nonlinear Estimation Algorithm for Attitude Determination of Micro Satellite [J]. Journal of Projectiles, Rockets, Missiles and Guidance, 2006, 26(3):67-70.