International Journal of Engineering and Applied Sciences (IJEAS)
ISSN: 2394-3661, Volume-4, Issue-5, May 2017

Research of an Adaptive Cubature Kalman Filter for GPS/SINS Tightly Integrated Navigation System

Chen Zhao, Shuai Chen, Yiping Wang

Abstract— **In view of the characteristics of Ballistic missile, the GPS/SINS tightly integrated navigation algorithm based on pseudo range/pseudo range rate in the Launch inertial coordinates is studied in this paper. The coordinate transformation method from ECEF coordinates to the Launch inertial coordinates is derived. The strap-down inertial navigation error equation and GPS error equation were derived in this coordinates. The state equations and measurement equations for GPS/SINS tightly integrated navigation system are established. Then an adaptive cubature kalman filter (ACKF) is employed to improve the position performance. The numerical simulation shows the better accuracy performance of the integrated navigation system.e and control the quality of the vector tracking loop.**

Index Terms— **Adaptive cubature kalman filter,High dynamics,Launch inertial coordinates,Tightly integrated navigation.**

I. Introduction

Global positioning system (GPS) and inertial navigation system (INS) has an important role in the field of navigation. GPS can provide the global position, velocity and time information all-time and all-weather. And its precision does not change with time. INS is a completely autonomous navigation system. It has less susceptible to external electromagnetic interference, high data rate, posture information, etc. However, in practical applications, the respective shortcoming of GPS and INS also limits their application. Because of GPS and INS have complementary, combination of the two can achieve performance transcendence. The GPS/SINS tightly integrated navigation algorithm generally based on geographic coordinates [1-5]. In view of the characteristics of Ballistic missile, the GPS/SINS tightly integrated navigation algorithm based on pseudo range/ pseudo range rate in the Launch inertial coordinates is studied in this paper. An adaptive cubature kalman filter is employed as the integration filter[6-8]. And the numerical simulation indicates that this GPS/SINS tightly integrated navigation algorithm can improve navigation accuracy effectively.

Chen Zhao, College of Automation , National University of Science and Technology,Nanjing ,China.

Shuai Chen, College of Automation , National University of Science and Technology,Nanjing ,e-mail:China.

Yiping Wang, College of Automation , National University of Science and Technology,Nanjing ,China

This work was supported by the Fundamental Research Funds for the Central Universities (No. 30916011336), Jiangsu Planned Projects for Postdoctoral Research Funds (No. 1501050B), and the China Postdoctoral Science Foundation (No. 2015M580434), the Research Innovation Program for College Graduates of Jiangsu Province (No. KYLX16_0463). This work also got the special grade of the financial support from the China Postdoctoral Science Foundation (No. 2016T90461).

II. Error Model In The Launch Inertial Coordinates

A. State error model of the SINS

The error angle between SINS mathematical simulation platform and the navigation coordinate is attitude misalignment angle. It is

$$\varphi = [\varphi_x \quad \varphi_y \quad \varphi_z]^T \tag{1}$$

The symmetric matrix of the attitude misalignment angle is

$$(\varphi\times) = \begin{bmatrix} 0 & -\varphi_z & \varphi_y \\ \varphi_z & 0 & -\varphi_x \\ -\varphi_y & \varphi_x & 0 \end{bmatrix} \tag{2}$$

The update equations for the attitude transformation matrix is

$$\dot{C}_b^n = C_b^n(\omega_{nb}^b\times) \tag{3}$$

Where ω_{nb}^b is inertial navigation system sensitive to the angular velocity .

Because of the inertial device measurement and calculation process exist error. The following relationship exists in the inertial navigation system between the actual output attitude transformation matrix $\tilde{C}_b^n$ and the ideal attitude transformation matrix C_b^n .

$$\tilde{C}_b^n = C_b^n + \delta C_b^n = (I + \varphi\times)C_b^n \tag{4}$$

Therefore,

$$\delta C_b^n = (\varphi\times)C_b^n \tag{5}$$

Derivative (5) to obtain:

$$\begin{aligned} \delta\dot{C}_b^n &= (\dot{\varphi}\times)C_b^n + (\varphi\times)\dot{C}_b^n \\ &= (\dot{\varphi}\times)C_b^n + (\varphi\times)C_b^n(\omega_{nb}^b\times) \end{aligned} \tag{6}$$

Differentiating both sides of (3) results in:

$$\begin{aligned} \delta\dot{C}_b^n &= \delta C_b^n(\omega_{nb}^b\times) + C_b^n(\delta\omega_{nb}^b\times) \\ &= (\varphi\times)C_b^n(\omega_{nb}^b\times) + (\delta\omega_{nb}^b\times) \end{aligned} \tag{7}$$

Comparing (6) and (7) can be obtained that:

$$(\dot{\varphi}\times) = C_b^n(\delta\omega_{nb}^b\times)C_n^b \tag{8}$$

Combined with vector operation $(ab)\times = a(b\times)a^T$ results in:

$$\dot{\varphi} = C_b^n\delta\omega_{nb}^b \tag{9}$$

Where $\delta\omega_{nb}^b = [\varepsilon_x \quad \varepsilon_y \quad \varepsilon_z]^T$ are gyro drift error.

In the launch inertial coordinate, speed differential equation is:

$$\dot{V} = C_b^n f^b + g_e \tag{10}$$

Differentiating both sides of (10) results in:

$$\delta\dot{V} = \left(\delta C_b^n\right) f^b + C_b^n \delta f^b + \delta g_e \tag{11}$$

The position error equation

$$\begin{bmatrix} \delta\dot{x} \\ \delta\dot{y} \\ \delta\dot{z} \end{bmatrix} = \begin{bmatrix} \delta V_x \\ \delta V_y \\ \delta V_z \end{bmatrix} \tag{12}$$

Inertial device error equation

$$\dot{\varepsilon} = 0, \dot{\nabla} = 0 \tag{13}$$

The system state equation of SINS in the launch inertial coordinate system can be obtained from 1) to 4) results in

$$\dot{X}_I = F_I X_I + G_I w_I \tag{14}$$

In the equation above

,

$$\begin{aligned} X_I(t) = [&\varphi_x \quad \varphi_y \quad \varphi_z \quad \delta V_x \quad \delta V_y \quad \delta V_z \\ &\delta x \quad \delta y \quad \delta z \quad \varepsilon_x \quad \varepsilon_y \quad \varepsilon_z \quad \nabla_x \quad \nabla_y \quad \nabla_z] \end{aligned} \tag{15}$$

$$F_I = \begin{bmatrix} 0_{3\times3} & 0_{3\times3} & 0_{3\times3} & C_b^n & 0_{3\times3} \\ F_1 & 0_{3\times3} & F_2 & 0_{3\times3} & C_b^n \\ 0_{3\times3} & I & & 0_{3\times9} & \\ & & 0_{6\times15} & & \end{bmatrix}_{15\times15},$$

$$G_I = \begin{bmatrix} C_b^n & 0_{3\times3} \\ 0_{3\times3} & C_b^n \\ 0_{9\times3} & 0_{9\times3} \end{bmatrix}_{15\times6}$$

$$w_I = \begin{bmatrix} \omega_{gx} & \omega_{gy} & \omega_{gz} & \omega_{ax} & \omega_{ay} & \omega_{az} \end{bmatrix}^T$$

In the equation above, w is system noise variance, ω_{gx} 、 ω_{gy} 、 ω_{gz} are gyro drift noise variances of three axis; ω_{ax} 、 ω_{ay} 、 ω_{az} are accelerometers noise variances of three axis; I is identity matrix;

$$F_1 = \begin{bmatrix} 0 & f_Z & -f_Y \\ -f_Z & 0 & f_X \\ f_Y & -f_X & 0 \end{bmatrix},$$

$$F_2 = \begin{bmatrix} \frac{\partial g_x}{\partial x} & \frac{\partial g_x}{\partial y} & \frac{\partial g_x}{\partial z} \\ \frac{\partial g_y}{\partial x} & \frac{\partial g_y}{\partial y} & \frac{\partial g_y}{\partial z} \\ \frac{\partial g_z}{\partial x} & \frac{\partial g_z}{\partial y} & \frac{\partial g_z}{\partial z} \end{bmatrix}$$

B. *GPS error equations*

When Usually we takes two time-related errors to represent the GPS error condition: the distance error , which is equivalent to the clock error Δl_u; the distance error ,which is equivalent to clock frequency error Δl_{ru} , error model expression as follows:

$$\begin{cases} \delta\dot{l}_u = \delta l_{ru} + \omega_u \\ \delta\dot{l}_{ru} = -\dfrac{\delta l_{ru}}{T_{ru}} + \omega_{ru} \end{cases} \tag{16}$$

Where T_{ru} is the relevant time, ω_u is white noise of GPS clock error, ω_{ru} is GPS clock frequency white noise.

GPS error equation can be expressed as follows:

$$\dot{X}_G(t) = F_G(t) X_G(t) + G_G(t) W_G(t) \tag{17}$$

In the equation above,

$$X_G(t) = [\delta l_u \quad \delta l_{ru}]^T \text{,} \quad W_G(t) = [\omega_u \quad \omega_{ru}]^T$$

$$F_G(t) = \begin{bmatrix} 0 & 1 \\ 0 & -\dfrac{1}{T_{ru}} \end{bmatrix} \text{,} \quad G_G(t) = \begin{bmatrix} 1 & 0 \\ 0 & 1 \end{bmatrix}$$

Therefore, tightly integrated navigation system error equation can be get from integrating the SINS and GPS error equations:

$$\begin{aligned} \begin{bmatrix} \dot{X}_I(t) \\ \dot{X}_G(t) \end{bmatrix} &= \begin{bmatrix} F_I(t) & 0 \\ 0 & F_G(t) \end{bmatrix} \begin{bmatrix} X_I(t) \\ X_G(t) \end{bmatrix} \\ &+ \begin{bmatrix} G_I(t) & 0 \\ 0 & G_G(t) \end{bmatrix} \begin{bmatrix} W_I(t) \\ W_G(t) \end{bmatrix} \end{aligned} \tag{18}$$

C. *Measurement equation*

The pseudo-range from GPS receiver to the GPS satellite numbered j can be expressed as follows:

$$\rho_G^j = \rho^j + \delta l_u + v_{\rho j} \tag{19}$$

In the equation above, ρ^j is the ideal pseudo range. δl_u is the equivalent distance corresponding to clock error. δl_u is main error in pseudo range measurement. Therefore, when building the pseudo range measurement model, the impact of the error is the primary consideration; $v_{\rho j}$ is pseudo range measurement noise, caused by multipath effect, tropospheric delay, ionospheric error. And it can be regarded as white noise。

Satellite position get from the ephemeris is in ECEF coordinate system, assuming that the position of the Satellite numbered j in the ECEF coordinate system is $\left(x_e^j \quad y_e^j \quad z_e^j\right)$. To compare with the INS information, we need to convert it to the launch inertial coordinate system:

(1) Convert Satellite position in ECEF (e coordinate) coordinate to launch coordinate (l coordinate), the conversion formula is as follows:

$$\begin{bmatrix} X_l \\ Y_l \\ Z_l \end{bmatrix} = C_e^l \cdot \begin{bmatrix} X_e \\ Y_e \\ Z_e \end{bmatrix} - \begin{bmatrix} r_{0x} \\ r_{0y} \\ r_{0z} \end{bmatrix} \tag{20}$$

In the equation above, r_0 is the geocentric vector emission points, C_e^l is the transpose matrix of C_l^e, C_l^e is as follows:

$$C_l^e = \begin{bmatrix} -\sin A_0 \sin\lambda_0 - \cos A_0 \sin B_0 \cos\lambda_0 & \cos B_0 \cos\lambda_0 & -\cos A_0 \sin\lambda_0 + \sin A_0 \sin B_0 \cos\lambda_0 \\ \sin A_0 \cos\lambda_0 - \cos A_0 \sin B_0 \sin\lambda_0 & \cos B_0 \sin\lambda_0 & \cos A_0 \cos\lambda_0 + \sin A_0 \sin B_0 \sin\lambda_0 \\ \cos A_0 \cos B_0 & \sin B_0 & -\sin A_0 \cos B_0 \end{bmatrix} \quad (21)$$

(2) Convert the position of the Satellite in Launch coordinate system (l coordinate) to launch inertial coordinate (g coordinate), the conversion formula is as follows:

$$\begin{bmatrix} X_g + r_{0x} \\ Y_g + r_{0y} \\ Z_g + r_{0z} \end{bmatrix} = C_l^g \cdot \begin{bmatrix} X_l + r_{0x} \\ Y_l + r_{0y} \\ Z_l + r_{0z} \end{bmatrix} \quad (22)$$

In the equation above, C_l^g is the transpose matrix of C_g^l, C_g^l is as follows:

$$C_g^l = \begin{bmatrix} (1-b_x^2)(1-\cos\omega_e t) & b_x b_y (1-\cos\omega_e t) + b_z \sin\omega_e t & b_x b_z (1-\cos\omega_e t) - b_y \sin\omega_e t \\ b_y b_x (1-\cos\omega_e t) - b_z \sin\omega_e t & 1-(1-b_y^2)(1-\cos\omega_e t) & b_y b_z (1-\cos\omega_e t) + b_x \sin\omega_e t \\ b_z b_x (1-\cos\omega_e t) + b_y \sin\omega_e t & b_z b_y (1-\cos\omega_e t) - b_x \sin\omega_e t & 1-(1-b_z^2)(1-\cos\omega_e t) \end{bmatrix} \quad (23)$$

In the equation above, $b_x = \cos B_0 \cos A_0$, $b_y = \sin B_0$, $b_z = -\cos B_0 \sin A_0$; B_0 is the latitude of the emission point, λ_0 is the longitude of the emission point, A_0 is the azimuth , ω_e is the angular velocity of Earth rotation.

Finally, the position of the Satellite in launch inertial coordinate is $\begin{pmatrix} x_g^j & y_g^j & z_g^j \end{pmatrix}$.

So the pseudo-range from the carrier to the GPS satellite numbered j can be get from INS:

$$\rho_g^j = \sqrt{(x_g - x_g^j)^2 + (y_g - y_g^j)^2 + (z_g - z_g^j)^2} \quad (24)$$

Assumed that the real value of the carrier position is $\begin{pmatrix} x & y & z \end{pmatrix}$, expand the equation above at $\begin{pmatrix} x & y & z \end{pmatrix}$ in taylor formula and discard second and higher order terms:

$$\rho_g^j = \rho^j + \frac{\partial\rho^j}{\partial x}\delta x + \frac{\partial\rho^j}{\partial y}\delta y + \frac{\partial\rho^j}{\partial z}\delta z \quad (25)$$

In the equation above,

$$\frac{\partial\rho^j}{\partial x} = \frac{x_g - x_g^j}{\rho_g^j} = e_{jx}, \quad \frac{\partial\rho^j}{\partial y} = \frac{y_g - y_g^j}{\rho_g^j} = e_{jy}, \quad \frac{\partial\rho^j}{\partial z} = \frac{z_g - z_g^j}{\rho_g^j} = e_{jz}.$$

The pseudo range measurement equations of tightly integrated navigation system can be expressed as follows:

$$\delta\rho^j = e_{jx}\delta x + e_{jy}\delta y + e_{jz}\delta z - \delta l_u - v_{\rho j} \quad (26)$$

Therefore,

$$Z_\rho(t) = H_\rho(t)X(t) + V_\rho(t) \quad (27)$$

Suppose that the number of effective GPS satellites is m, so:

$$Z_\rho(t) = \begin{bmatrix} \delta\rho^1 & \delta\rho^2 & \cdots & \delta\rho^m \end{bmatrix}^T$$

$$H_\rho(t) = \begin{bmatrix} 0_{1\times6} & e_x^1 & e_y^1 & e_z^1 & 0_{1\times6} & -1 & 0 \\ 0_{1\times6} & e_x^2 & e_y^2 & e_z^2 & 0_{1\times6} & -1 & 0 \\ \vdots & \vdots & \vdots & \vdots & \vdots & \vdots & \vdots \\ 0_{1\times6} & e_x^m & e_y^m & e_z^m & 0_{1\times6} & -1 & 0 \end{bmatrix}$$

$$V_\rho(t) = \begin{bmatrix} -v_{\rho1} & -v_{\rho2} & \cdots & -v_{\rho m} \end{bmatrix}^T$$

The pseudo range rate from GPS receiver to the GPS satellite numbered j can be expressed as follows:

$$\dot\rho_G^j = \dot\rho^j + \delta l_{ru} + v_{\rho j} \quad (28)$$

Suppose that the velocity of Satellite numbered j in ECEF coordinates is $\begin{pmatrix} v_{ex}^j & v_{ey}^j & v_{ez}^j \end{pmatrix}$, convert it to launch inertial coordinate, the velocity of Satellite in launch inertial coordinate can be obtained as follows : $\begin{pmatrix} v_{ex}^j & v_{ey}^j & v_{ez}^j \end{pmatrix}$.

According to carrier speed obtained by INS in launch inertial coordinate $\begin{pmatrix} v_{gx} & v_{gy} & v_{gz} \end{pmatrix}$ the pseudo-range rate can be get as follows:

$$\dot\rho_g^j = e_{jx}(v_{gx} - v_{gx}^j) + e_{jy}(v_{gy} - v_{gy}^j) + e_{jz}(v_{gz} - v_{gz}^j) \quad (29)$$

Assumed that the real value of the carrier speed is $\begin{pmatrix} v_x & v_y & v_z \end{pmatrix}$, expand the equation above at $\begin{pmatrix} v_x & v_y & v_z \end{pmatrix}$ in taylor formula and rounding second and higher order terms:

$$\dot\rho_g^j = \dot\rho^j + e_{jx}\delta v_x + e_{jy}\delta v_y + e_{jz}\delta v_z \quad (30)$$

So the pseudo range rate measurement equation is

$$\delta\dot\rho^j = e_{jx}\delta v_x + e_{jy}\delta v_y + e_{jz}\delta v_z - \delta l_{ru} - v_{\rho j} \quad (31)$$

Therefore,

$$Z_{\dot\rho}(t) = H_{\dot\rho}(t)X(t) + V_{\dot\rho}(t) \quad (32)$$

Similarly, suppose that the number of effective GPS satellites is m, so:

$$Z_{\dot\rho}(t) = \begin{bmatrix} \delta\dot\rho^1 & \delta\dot\rho^2 & \cdots & \delta\dot\rho^m \end{bmatrix}^T$$

$$H_{\dot\rho}(t) = \begin{bmatrix} 0_{1\times3} & e_x^1 & e_y^1 & e_z^1 & 0_{1\times9} & 0 & -1 \\ 0_{1\times3} & e_x^2 & e_y^2 & e_z^2 & 0_{1\times9} & 0 & -1 \\ \vdots & \vdots & \vdots & \vdots & \vdots & \vdots & \vdots \\ 0_{1\times3} & e_x^m & e_y^m & e_z^m & 0_{1\times9} & 0 & -1 \end{bmatrix}$$

$$V_{\dot\rho}(t) = \begin{bmatrix} -v_{\dot\rho1} & -v_{\dot\rho2} & \cdots & -v_{\dot\rho m} \end{bmatrix}^T$$

The measurement equation of tightly integrated navigation system can be get as follows:

$$Z(t)=\begin{bmatrix}H_\rho(t)\\H_{\dot\rho}(t)\end{bmatrix}X(t)+\begin{bmatrix}V_\rho(t)\\V_{\dot\rho}(t)\end{bmatrix}=H(t)X(t)+V(t) \quad (33)$$

III. The Adaptive Cubature Kalman Filter

Considering a discrete nonlinear system as

$$\mathrm{x}_{k+1}=f(x_k,k)+w_k$$
$$z_k=h(x_k,k)+v_k$$

Where the state vector $x_k\in\Re^n$, the process noise vector is $w_k\in\Re^n$, the measurement noise vector is $v_k\in\Re^m$, the measurement vector is $z_k\in\Re^m$. The vectors w_k and v_k are zero mean Gaussian white sequences with zero cross-correlation with each other:

$$E[w_kw_i^T]=\begin{cases}Q_k, i=k\\0, i\neq k\end{cases}$$

$$E[v_kv_i^T]=\begin{cases}R_k, i=k\\0, i\neq k\end{cases};E[w_kv_i^T]$$

Where Q_k is the process noise covariance matrix, and R_k is the measurement noise covariance matrix.

The details of the implementation of ACKF are as follows:

Step One: Initialize state vector $\hat{x}_{0|0}$ and state covariance matrix $P_{0|0}$

Step Two: Time update

Factorize the covariance:

$$P_{k-1|k-1}=S_{k-1|k-1}S_{k-1|k-1}^T \quad (34)$$

Evaluate the cubature points through the process model

$$X_{i,k-1|k-1}=S_{k-1|k-1}\xi_i+\hat{X}_{k-1|k-1} \quad (35)$$

Estimate the propagated cubature points through the process model

$$X_{i,k,k-1}^*=f(X_{i,k-1|k-1}) \quad (36)$$

Estimate the predicted mean

$$\hat{x}_{k|k-1}=\sum_{i=1}^{2n}\omega_iX_{i,k|k-1}^* \quad (37)$$

Estimate the predicted error covariance

$$P_{k|k-1}=\sum_{i=1}^{2n}\omega_iX_{i,k|k-1}^*X_{i,k|k-1}^{*\,T}-\hat{x}_{k|k-1}\hat{x}_{k|k-1}^T+Q_{k-1} \quad (38)$$

Step Three: Measurement update

Factorize the covariance

$$P_{k|k-1}=S_{k|k-1}S_{k|k-1}^T \quad (39)$$

Evaluate the cubature points

$$X_{i,k|k-1}=S_{k|k-1}\xi_i+\hat{x}_{k|k-1} \quad (40)$$

Evaluate the propagated cubature points through observation model

$$Z_{i,k|k-1}=h(X_{i,k|k-1}) \quad (41)$$

Evaluate the propagated observation

$$\hat{z}_{k|k-1}=\sum_{i=1}^{2n}\omega_iZ_{i,k|k-1} \quad (42)$$

(5) Evaluate the innovation covariance

$$P_{zz}=\frac{1}{a_k}\sum_{i=1}^{2n}\omega_iZ_{i,k|k-1}Z_{i,k|k-1}^T-\hat{z}_{k|k-1}\hat{z}_{k|k-1}^T+R_k \quad (43)$$

(6) Estimate the cross-covariance

$$P_{xz}=\frac{1}{a_k}\sum_{i=1}^{2n}\omega_iX_{i,k|k-1}Z_{i,k|k-1}^T-\hat{x}_{k|k-1}\hat{z}_{k|k-1}^T \quad (44)$$

$$K_k=P_{xz}P_{zz}^{-1} \quad (45)$$

$$x_{k|k}=\hat{x}_{k|k-1}+K_k(z_k-\hat{z}_{k|k-1}) \quad (46)$$

$$P_{k|k}=\frac{1}{a_k}P_{k|k-1}-K_kP_{zz}K_k^T \quad (47)$$

Where:

$$a_k=\frac{tr(P)}{tr(e(k)e(k)^T)} \quad (48)$$

$$P=\sum_{i=1}^{2n}\omega_iZ_{i,k|k-1}Z_{i,k|k-1}^T-\hat{z}_{k|k-1}\hat{z}_{k|k-1}^T \quad (49)$$

IV. The Simulation And Results

In order to verify the correctness of the algorithm, ballistic trajectory generator platform and numerical simulation are designed in this paper. The missile is fired vertically, the total flight time is 200s, the time of powered phase is 60s, the thrust acceleration in powered phase is 40 m/s^2; the constant drift of gyro is 0.5°/h, the white noise of gyro is 0.05/h; the constant drift of accelerometer is 1mg, the white noise of accelerometer is 0.5mg; The start point: latitude: 31.98°N, longitude:118.8°E, height: 0m;

Initial speed: forward speed: 394.8917m/s, upward speed 0m/s, lateral speed 0m/s. White noise of pseudo-range is 10m, the white noise of pseudo-range rate is 0.1m/s.

Initial attitude: pitch angle: 90°, roll angle: 0°, yaw angle: 0°, The initial attitude error is 0.1°. The numerical simulation results are as follows:

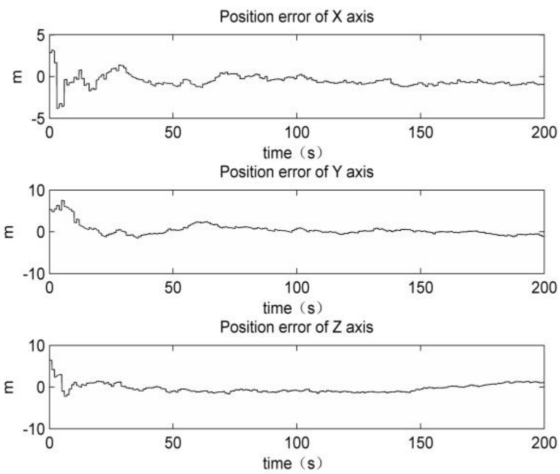

Figure 1. The position error

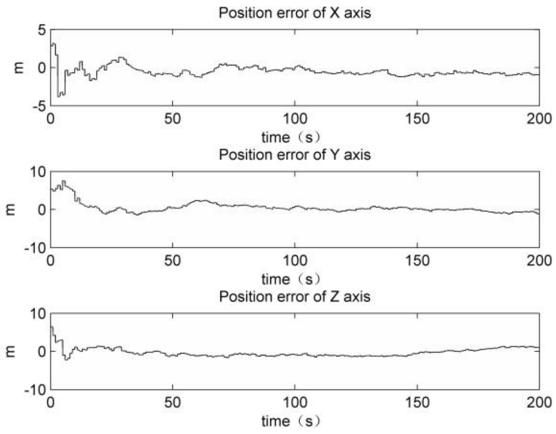

Figure 2. The velocity error

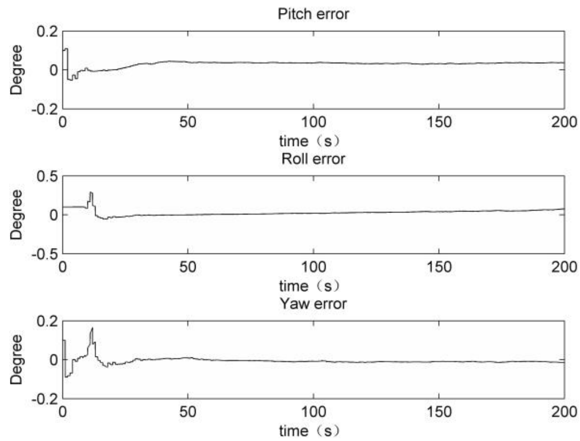

Figure 3. The attitude error

According to figure 1 to figure 3, when received eight stars, position error can converge to within 2m, speed error can converge to 0.1m/s or less, attitude error can converge to within 0.05 °.

V. Conclusions

According to the characteristics of ballistic missiles, a kind of GPS/SINS tightly integrated navigation algorithm in Launch inertial coordinate is designed. And the state equations and measurement equations for GPS/SINS tightly integrated navigation system are established. An adaptive cubature kalman filter was applied to the system. The numerical simulation indicates that this GPS/SINS tightly integrated navigation algorithm can improve navigation accuracy effectively.

Acknowledgment

Chen Zhao thanks the team he belongs to,it is the team that provides a good academic atmosphere.

References

[1] Wu Ruixiang. Research On Key Technology In GPS/MIMU Tightly Integrated System [D]. Nanjing: Southeast University, 2010.
[2] Ji Feng. Research on INS/GPS Tightly Integrated Navigation System Simulation and Key Technology [D]. Nanjing: Nanjing University of Aeronautics and Astronautics, 2009.
[3] Ye Ping. Research on mems IMU/GNSS ultratight integration navigation Technology [D]. Shanghai:Shanghai Jiao Tong University, 2011.
[4] Yuan Jungang. The Study of Tightly GPS/INS Integrated Navigation System [D]. Nanjing: Nanjing University of Aeronautics and Astronautics, 2011.
[5] Lei Haoran. Study on Missile-Borne SINS/GNS Integrated Navigation System [D]. Nanjing: Nanjing University of Science & Technology, 2014.
[6] Fang Jiancheng, Ning Xiaolin, Tian Yulong. Spacecraft Autonomous Celestial Navigation Principles and Methods [M]. Beijing: National Defense Industry Press, 2006.
[7] Zhang Libin. Study on navigation, midcourse correction and attitude control of launch vehicle upper stage [D]. Harbin: Harbin Institute of Technology.
[8] Jiang Jinlon. Research on SINS/SAR/GPS Integrated Navigation System [D]. Harbin: Harbin Institute of Technology.

Chen Zhao ,a master study in College of Automation , National University of Science and Technology,Nanjing ,China. Research on the integrated navigation,the inertial navigation.

Shuai Chen, a associate professor of College of Automation , National University of Science and Technology,Nanjing ,China. Research on the integrated navigation, the inertial navigation.

Yiping Wang ,a master study in College of Automation , National University of Science and Technology,Nanjing ,China. Research on the integrated navigation, the inertial navigation.