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INS/GPS Integrated Navigation Technology for Hypersonic UAV

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Abstract

INS/GPS integrated navigation system is studied in this paper for the hypersonic UAV in order to satisfy the precise guidance requirements of hypersonic UAV and in response to the defects while the inertial navigation system (INS) and the global positioning system (GPS) are being applied separately. The information of UAV including position, velocity and attitude can be obtained by using INS and GPS respectively after generating a reference trajectory. The corresponding errors of two navigation systems can be obtained through comparing the navigation information of the above two guidance systems. Kalman filter is designed to estimate the navigation errors and then the navigation information of INS are corrected. The non-equivalence relationship between the platform misalignment angle and attitude error angle are considered so that the navigation accuracy is further improved. The Simulink simulation results show that INS/GPS integrated navigation system can help to achieve higher accuracy and better anti-interference ability than INS navigation system and this system can also satisfy the navigation accuracy requirements of hypersonic UAV.

Keywords: *hypersonic UAV, INS/GPS integrated navigation, Kalman filter*

1. Introduction

Hypersonic vehicle has become the commanding heights of current military competition with its advantages of rapid global reach and high penetration probability. Because hypersonic vehicle will operate at the high velocity and with high dynamics, the navigation system of hypersonic vehicle is the critical factor to ensure the completion of precise strike missions.

The INS can obtain all navigation information at high accuracy independently within short period of time. However, if it works for long time, the error of the INS will be rapidly accumulated over time and the navigation precision will be decreased [1]. The GPS can obtain high-accuracy information of velocity and position. However, while the carrier is moving with high dynamics, it is difficult for GPS receiver to track the satellite carrier signal and GPS receiver will be out of work due to lock-lose, so it is difficult to guarantee its reliability also [2]. Therefore, in order to effectively solve the problems while using INS or GPS separately and to guarantee the high reliability and high precision navigation performance of hypersonic UAV under long time working condition, the best way is to integrate the INS with GPS to INS/GPS integrated navigation system. Then the optimum digital filter technology can be used to estimate the navigation error and correct the navigation information output from INS. INS/GPS integrated navigation system is superior to the individual system in terms of accuracy, performance and reliability [3,4]. In addition, this combination has the potential to detecting the accumulated error while the common integrated navigation system is unable to achieve [5]. In this study, the non-equivalence relationship between platform misalignment angle and attitude error angle is considered so as to further improve the precision of navigation.

2. Basic Principle of Kalman Filter

The equations of state and observation of one linear system are given as follows:

$$X_k = \Phi_{k,k-1} X_{k-1} + G_{k-1} w_{k-1} \tag{1}$$

$$z_k = H_k X_k + v_k \tag{2}$$

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where X_k is the state variables at time t_k; w_k is the system noise at time t_k; z_k is the observed variable at time t_k; v_k is the observed noise at time t_k; $\Phi_{k,k-1}$ is the state transition matrix from time t_{k-1} to time t_k; G_{k-1} is the coefficient matrix for system noise; H_k is the measurement matrix of observed equation; w_k and v_k are independent zero-mean Gaussian white noise sequence and Q_k and R_k are their variances respectively.

The estimated value of X_k is expressed as $\hat{X}_k$. According to the principle of Kalman filter, $\hat{X}_k$ can be obtained by the following equation:

The one-step prediction of state variable:

$$\hat{X}_{k/k-1} = \Phi_{k,k-1}\hat{X}_{k-1} \tag{3}$$

The one-step prediction mean-square error of state variable:

$$P_{k/k-1} = \Phi_{k,k-1}P_{k-1}\Phi^{\mathrm{T}}_{k,k-1} + G_{k-1}Q_{k-1}G^{\mathrm{T}}_{k-1} \tag{4}$$

The filter gain:

$$K_k = P_{k/k-1}H_k^{\mathrm{T}}(H_k P_{k/k-1}H_k^{\mathrm{T}} + R_k)^{-1} \tag{5}$$

The optimized state estimation:

$$\hat{X}_k = \hat{X}_{k/k-1} + K_k(z_k - H_k\hat{X}_{k/k-1}) \tag{6}$$

The mean-square error of estimator:

$$P_k = (I - K_k H_k)P_{k/k-1}(I - K_k H_k)^{\mathrm{T}} + K_k R_k K_k^{\mathrm{T}} \tag{7}$$

As long as the initial value of $\hat{X}_0$ and P_0 are given, the state estimation $\hat{X}_k$ at time k can be calculated according to the measured z_k at time k.

3. Design of INS/GPS Integrated Navigation System

In this study, firstly INS is used to obtain the position, velocity and attitude information of hypersonic UAV. Then the corresponding data obtained by GPS is compared with the INS output data and the difference is used as the inputs for Kalman filter. Kalman filter will output the navigation error estimate to correct the navigation information output from INS [6]. Figure 1 provides the schematic diagram of INS/GPS integrated navigation system for the hypersonic UAV.

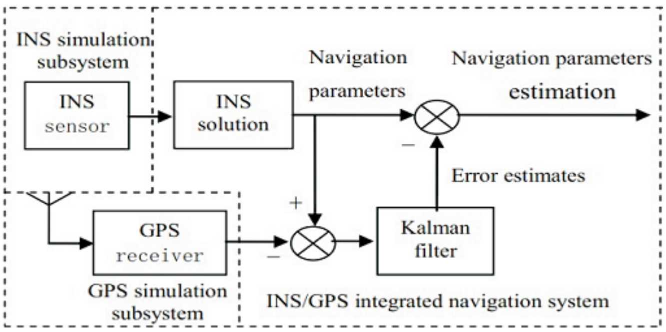

Figure 1. Schematic diagram of INS/GPS integrated navigation system

3.1. Mechanical Arrangement of Earth-Fixed Coordinate System

The task of INS mechanical arrangement is to use the appropriate mathematical model and measurement to calculate the navigation information. Figure 2 shows the flow chart for calculating the navigation information of UAV in earth-fixed coordinate system.

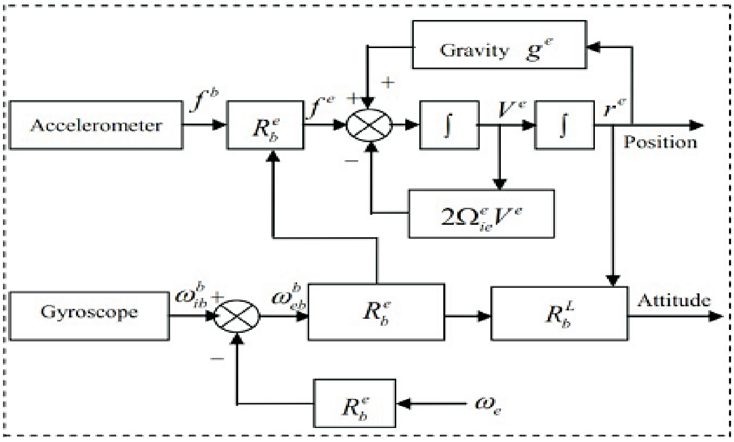

Figure 2. Mechanized arrangement in earth-fixed coordinate system

As it can be seen from Figure 2, the calculation of the navigation information is divided into two steps: At first, the angular velocity measured by the gyroscope (body coordinate system relative to the inertial coordinate system) is used to minus the angular velocity of the Earth rotation (earth-fixed coordinate system relative to the inertial coordinate system) so to obtain the desired angular velocity ω_{eb}^b (the body coordinate system relative to the earth-fixed coordinate system). The body coordinate system is corrected to the earth-fixed coordinate system by using ω_{eb}^b according to quaternion integral method to fix the coordinate transformation matrix. Secondly, the measured acceleration f^b in the body coordinate system is converted to acceleration f^e in the earth-fixed coordinate system through the coordinate transformation, and then the acceleration of gravity and Coriolis acceleration are compensated thereinto. The velocity and position of UAV can be obtained after integrating the acceleration. The position coordinates (latitude φ, longitude λ and altitude h) are used to calculate the matrix R_e^L. The transformation matrix R_b^L transformed from body coordinate system to the local level coordinate system can be obtained by $R_b^L = R_e^L R_b^e$ and then the information of the attitude angle can be achieved.

3.2. Relationship between Platform Misalignment Angle and Attitude Error Angle

Misalignment angle is considered as the attitude angle error in many studies while using the Kalman filter to estimate the navigation error [7-9]. However, this approximation will increase the navigation error due to inaccurate mathematical model [10]. In order to improve the navigation accuracy, the relationship between platform misalignment angle ϕ_e, ϕ_n, ϕ_u (subscripts represent east, north and heaven directions in local coordinate) and attitude error angle $\delta\psi, \delta\theta, \delta\gamma$ is achieved in this paper through analyzing the physical meaning of misalignment angle and attitude angle error. The transformation matrix among navigation coordinate system (n), platform coordinate system (L) and body coordinate system (b) is with following relationship [10]:

$$R_b^L = R_n^L R_b^n \tag{8}$$

where R_b^L is the measured attitude matrix, R_n^L is the platform misalignment angle matrix and R_b^n is the ideal attitude matrix.

$$R_b^L = \begin{bmatrix} T_{11}^{'} & T_{12}^{'} & T_{13}^{'} \\ T_{21}^{'} & T_{22}^{'} & T_{23}^{'} \\ T_{31}^{'} & T_{32}^{'} & T_{33}^{'} \end{bmatrix}, R_b^n = \begin{bmatrix} T_{11} & T_{12} & T_{13} \\ T_{21} & T_{22} & T_{23} \\ T_{31} & T_{32} & T_{33} \end{bmatrix}$$

The platform misalignment angle can be expressed as follows:

$$\phi = \begin{bmatrix} \phi_e \\ \phi_n \\ \phi_u \end{bmatrix}$$

As the platform misalignment angle is very small, the following relationship can be obtained by omitting high order small quantity $R_n^L = I - [\phi\times]$.

The attitude angle obtained from the inertial navigation system should be the sum of ideal attitude angle and attitude angle error, that is:

$$\begin{cases} \psi = \psi_t + \delta\psi \\ \theta = \theta_t + \delta\theta \\ \gamma = \gamma_t + \delta\gamma \end{cases} \tag{9}$$

where ψ, θ, γ are the measured attitude angles of INS, $\psi_t, \theta_t, \gamma_t$ are the ideal attitude angles, $\delta\psi, \delta\theta, \delta\gamma$ are the corresponding attitude angle errors.

The following formula can be obtained from (9)

$$\tan\psi = \tan(\psi_t + \delta\psi) = \frac{\tan\psi_t + \tan\delta\psi}{1 - \tan\psi_t \tan\delta\psi} \tag{10}$$

As $\delta\psi$ is very small, make Taylor extension for $\tan\psi$ and ignore the high-order terms. The following formula can be obtained.

$$\tan\psi = \tan\psi_t + (1 + \tan^2\psi_t)\delta\psi \tag{11}$$

According to (8), the following formula can be obtained.

$$\tan\psi = \tan(\psi_t + \delta\psi) = \frac{T_{12}^{'}}{T_{22}^{'}} = \frac{T_{12} + T_{22}\phi_u - T_{32}\phi_n}{T_{22} - T_{12}\phi_u + T_{32}\phi_e} \tag{12}$$

Make Taylor extension of above formula by ignoring the high-order terms,

$$\tan\psi_t + (1 + \tan^2\psi_t)\delta\psi = \frac{T_{12}}{T_{22}} + (1 + \frac{T_{12}^2}{T_{22}^2})\phi_u - \frac{T_{32}}{T_{22}}\phi_n - \frac{T_{12}T_{32}}{T_{22}^2}\phi_e \tag{13}$$

Combine (13) with (11) and simplify,

$$\delta\psi = \phi_u - \frac{T_{32}T_{22}}{T_{12}^2 + T_{22}^2}\phi_n - \frac{T_{12}T_{32}}{T_{12}^2 + T_{22}^2}\phi_e \tag{14}$$

The following formula can be obtained by similar method.

$$\delta\theta = \frac{T_{12}}{\sqrt{1-T_{32}^2}}\phi_n - \frac{T_{22}}{\sqrt{1-T_{32}^2}}\phi_e \tag{15}$$

$$\delta\gamma = \frac{T_{21}T_{33} - T_{23}T_{31}}{T_{31}^2 + T_{33}^2}\phi_e + \frac{T_{13}T_{31} - T_{11}T_{33}}{T_{31}^2 + T_{33}^2}\phi_n \tag{16}$$

3.3. State Equation of INS/GPS Integrated Navigation System

The navigation system is non-linear system in general and the equations describing the system are very complicated. Although the error equation of navigation system is non-linear, the high-order terms about error of the nonlinear equations can be ignored because the error is very small [6]. Thus the error equation of navigation system can be approximately described by one linear equation.

Integrated navigation system generally consists of 21 state variables: position errors $\delta\varphi, \delta\lambda, \delta h$, velocity errors $\delta V_e, \delta V_n, \delta V_u$, platform misalignment angles ϕ_e, ϕ_n, ϕ_u, constant drifts of gyro d_1, d_2, d_3, random drifts of gyro $\varepsilon_{b1}, \varepsilon_{b2}, \varepsilon_{b3}$, constant zero offsets of accelerometer b_1, b_2, b_3, and random drifts of accelerometer $\Delta_1, \Delta_2, \Delta_3$. Generally speaking, with laboratory conditions and field instrument testing, the constant drift of gyro and the constant zero offset of accelerometer can be obtained and the corresponding observables (angle velocity and acceleration) can be corrected [13]. Therefore, 15 states besides the constant drift of gyro and the constant zero offset of accelerometer can be used as variables in our study.

The state equation of integrated navigation system is as follows:

$$\dot{X}(t) = F(t)X(t) + G(t)w(t) \tag{17}$$

where $X(t) = [\delta\varphi, \delta\lambda, \delta h, \delta V_e, \delta V_n, \delta V_u, \phi_e, \phi_n, \phi_u, \varepsilon_{b1}, \varepsilon_{b2}, \varepsilon_{b3}, \Delta_1, \Delta_2, \Delta_3]^{\mathrm{T}}$.

The non-zero elements of $F(t)$ are $F(1,3) = -\frac{\dot{\varphi}}{\mathrm{R}+h}$, $F(1,5) = \frac{1}{\mathrm{R}+h}$, $F(2,1) = \dot{\lambda}\tan\varphi$, $F(2,3) = -\frac{\dot{\lambda}}{\mathrm{R}+h}$, $F(2,4) = \frac{1}{(\mathrm{R}+h)\cos\varphi}$, $F(3,6) = 1$, $F(4,1) = 2\omega_e(V_u\sin\varphi + V_n\cos\varphi) + \frac{V_n\dot{\lambda}}{\cos\varphi}$, $F(4,4) = -\frac{\dot{h}}{\mathrm{R}+h} + \dot{\varphi}\tan\varphi$, $F(4,5) = (2\omega_e + \dot{\lambda})\sin\varphi$, $F(4,6) = -(2\omega_e + \dot{\lambda})\cos\varphi$, $F(4,8) = f_u$, $F(4,9) = -f_n$, $F(4,13) = R_{11}$, $F(4,14) = R_{12}$, $F(4,15) = R_{13}$, $F(5,1) = -2\omega_e V_e\cos\varphi - \frac{V_e\dot{\lambda}}{\cos\varphi}$, $F(5,4) = -2(\omega_e + \dot{\lambda})\sin\varphi$, $F(5,5) = -\frac{\dot{h}}{\mathrm{R}+h}$, $F(5,6) = -\dot{\varphi}$, $F(5,7) = -f_u$, $F(5,9) = f_e$, $F(5,13) = R_{21}$, $F(5,14) = R_{22}$, $F(5,15) = R_{23}$, $F(6,1) = -2\omega_e V_e\sin\varphi$, $F(6,4) = 2(\omega_e + \dot{\lambda})\cos\varphi$, $F(6,5) = 2\dot{\varphi}$, $F(6,7) = f_n$, $F(6,8) = -f_e$, $F(6,13) = R_{31}$, $F(6,14) = R_{32}$, $F(6,15) = R_{33}$, $F(7,3) = -\frac{\dot{\varphi}}{\mathrm{R}+h}$, $F(7,5) = \frac{1}{\mathrm{R}+h}$, $F(7,8) = (\omega_e + \dot{\lambda})\sin\varphi$, $F(7,9) = -(\omega_e + \dot{\lambda})\cos\varphi$, $F(7,10) = R_{11}$, $F(7,11) = R_{12}$, $F(7,12) = R_{13}$, $F(8,1) = \omega_e\sin\varphi$, $F(8,3) = \frac{\dot{\lambda}\cos\varphi}{\mathrm{R}+h}$, $F(8,4) = -\frac{1}{\mathrm{R}+h}$, $F(8,7) = -(\omega_e + \dot{\lambda})\sin\varphi$, $F(8,9) = -\dot{\varphi}$, $F(8,10) = R_{21}$, $F(8,11) = R_{22}$, $F(8,12) = R_{23}$, $F(9,1) = -\omega_e\cos\varphi - \frac{\dot{\lambda}}{\cos\varphi}$, $F(9,3) = \frac{\dot{\lambda}\sin\varphi}{\mathrm{R}+h}$, $F(9,4) = \frac{-\tan\varphi}{\mathrm{R}+h}$, $F(9,7) = (\omega_e + \dot{\lambda})\cos\varphi$, $F(9,8) = \dot{\varphi}$, $F(9,10) = R_{31}$, $F(9,11) = R_{32}$, $F(9,12) = R_{33}$, where R represents the radius of earth, ω_e represents the rotational angular velocity of earth.

Format of G is taken as $G = [zeros(9,6); eye(6)]$.

3.4. Measurement Equation of INS/GPS Integrated Navigation System

The difference of corresponding navigation information (position and speed) between INS and GPS is considered as the observables of Kalman filter. In this difference, the real navigation parameters have been offset and the rest are the errors of the two navigation systems. Therefore, the measurement equation of navigation system is also linear equation. The measurement equation could be expressed as follows:

$$z = \begin{bmatrix} \varphi_I - \varphi_G \\ \lambda_I - \lambda_G \\ h_I - h_G \\ V_{Ie} - V_{Ge} \\ V_{In} - V_{Gn} \\ V_{Iu} - V_{Gu} \end{bmatrix} = H(t)X(t) + v(t) \tag{18}$$

where $H = [I_{6\times6} \quad 0_{6\times3} \quad 0_{6\times6}]$.

The subscripts I" and G" represent the states of INS and GPS navigation system respectively.

Thus, the INS mechanical arrangement is completed and the relationship between the platform misalignment angle and the attitude error angle has been achieved. Moreover, the state equation and measurement equation of integrated navigation system has been built.

4. Simulation of INS/GPS Integrated Navigation System

In order to verify the feasibility of the above scheme, the simulation platform of the INS/GPS integrated navigation system is established. The simulation of integrated navigation system of hypersonic UAV under sliding condition will be given in the later sections. The simulation conditions are as follows: Kamlan filter calculation cycle is 0.1 seconds, the simulation time is set to 600 seconds, random drift of gyro is $1^{\circ}/\mathrm{h}$ and random drift of accelerometer is $500\mu\mathrm{g}$. Under the above conditions, the simulation has been conducted for INS and INS/GPS integrated navigation system respectively. Figure 3 and Figure 4 show the position error of INS and INS/GPS integrated navigation system respectively; Figure 5 and Figure 6 show the velocity error of INS and INS/GPS integrated navigation system respectively.

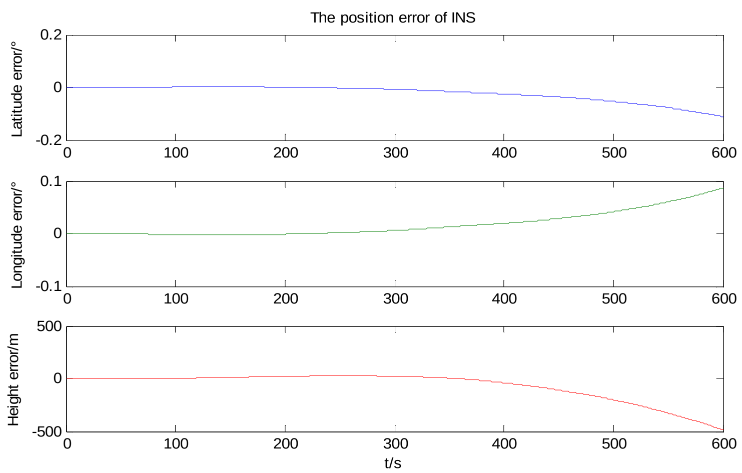

Figure 3. Position errors of INS navigation system

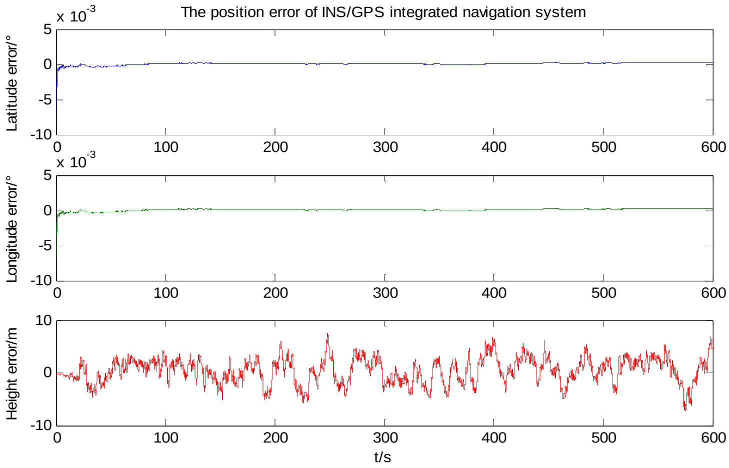

Figure 4. Position errors of INS/GPS integrated navigation system

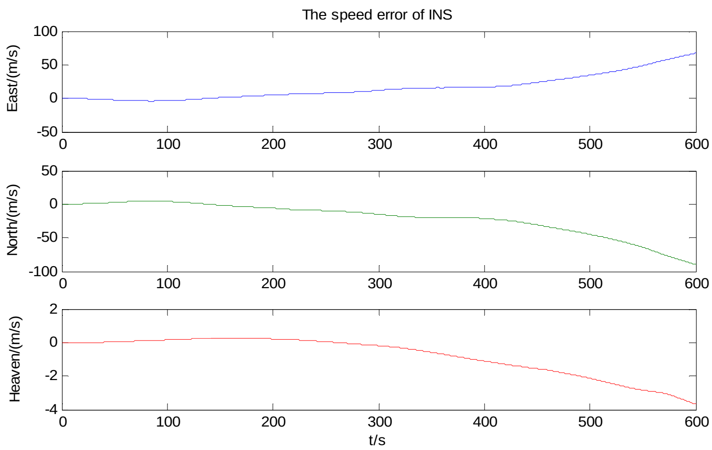

Figure 5. Speed errors of INS navigation system

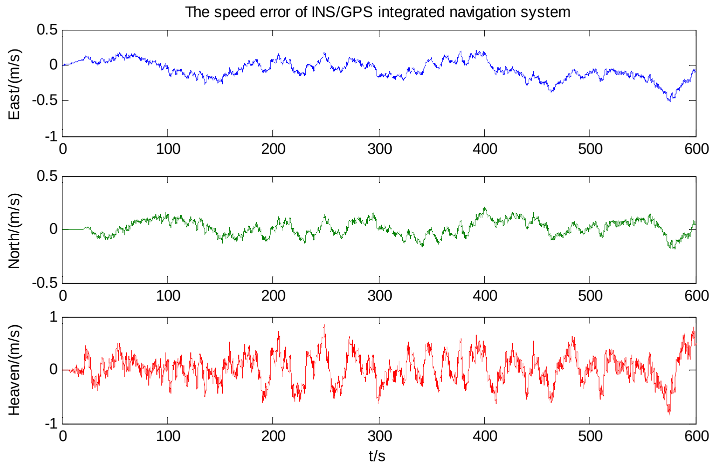

Figure 6. Speed errors of INS/GPS integrated navigation system

These figures show that the accumulated error of INS will cause the obvious increasing of position error and velocity error with the increase of time, so separate application can not satisfy the navigation requirements. INS/GPS integrated navigation system significantly improved the navigation accuracy since it is able to output much more accurate information of position and velocity. Besides, the navigation errors of INS/GPS integrated navigation system will not be accumulated over time. Therefore, INS/GPS integrated navigation system is a relatively ideal integrated navigation system which can satisfy the navigation requirements of hypersonic UAV.

5. Conclusion

INS/GPS integrated navigation system is designed for hypersonic UAV. The integrated navigation accuracy is further improved through introducing the conversion relationship between misalignment angle and attitude angle error. Simulation results show that the INS/GPS integrated navigation system is able to achieve higher level of navigation accuracy than INS and meanwhile the reliability of system can be improved. The design requirements of navigation system for hypersonic UAV can be satisfied.

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