Auxiliary Error Correction Method for High Precision IMU

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Abstract. According to the needs of error correction about the low precision IMU strapdown calculation parameters, error correction method is researched under the high precision IMU, an error correction scheme is proposed, established the state space equation and the corresponding measurement equation, The simulation experiment is carried out in the state of motion carrier, The results show that the method is convergent to parameters of the estimation results, the attitude error, the accelerometer constant zero bias and the gyro zero bias is estimated effectively, the estimation accuracy of the attitude error can be achieved within one arc min. The method has high autonomy and strong anti-interference ability for the error correction of the low precision IMU, which is can be used in the case of the weapon system with multi precision inertial group.

1. Introduction
In Strapdown Inertia calculation, due to the accumulation of errors in integral operation, the output accuracy of system parameters will decrease after a period of work, which requires error correction of the main parameters [1-3]. The general idea of parameter error correction of inertial system is to use external reference information to obtain observation data, estimate and correct parameter errors on the basis of establishing error state model. For example, position observation can be obtained by using satellite positioning auxiliary information, and attitude observation can be obtained by using satellite attitude determination auxiliary information. However, the satellite positioning process is affected by electromagnetic interference. The starlight orientation is influenced by weather conditions [4-6]. In order to further improve the autonomy of strapdown inertial solution error correction process, this paper studies a correction method of low precision inertial solution parameter error assisted by high precision inertial unit. This method can be applied to weapon systems with more than two sets of different precision inertial units, such as high precision inertial unit on missile launcher to modify low precision inertial unit system on missile, and multi-level inertial unit. Error correction before separation of bearing inertial navigation vehicle at all levels, etc.

2. Aided Error Correction Scheme of High Precision Inertia Unit
In order to correct the main parameter errors in attitude calculation of low-precision inertial units, a high-precision inertial unit can be selected as the external information source, and the results of comparison with the low-precision inertial unit can be used as the observation. The main error states are estimated and corrected. The implementation scheme is shown in Figure 1. This scheme is suitable
for the case of multiple inertial units on the carrier. It can be seen from the figure that the key of error correction is how to accurately estimate the errors of relevant parameters. On the basis of error estimation, the error correction of parameters is realized by subtracting the measured or calculated parameters from the corresponding error estimates.

Velocity information, angular velocity information, attitude information and the combination information of the above parameters can be selected in the selection of observation variables. Different observation variables will have different error estimation effects [7] [8]. While guaranteeing the real-time performance of the algorithm, more observation information can be selected to improve the accuracy of error estimation. According to the results of the research [9] [10], the best observation quantity can be selected. In this paper, the velocity and attitude information obtained by high precision inertial navigation system is used as the external reference information source to estimate the relative parameter errors of low precision attitude calculation.

3. Mathematical Model of Error Correction

Taking land-based equipment as the research background, the reference system is t (east-north-celestial coordinate system), ignoring the influence of altitude channel, and the carrier is located in the geographic latitude $\lambda$, longitude $\phi$, altitude $h$, the earth's long radius $R_e = 6378245m$, the earth's ellipticity $e = 1/298.3$, $R_p = R_e/(1-e \sin^2 \lambda)$ the principal curvature radius in the local unitary plane, and $R_s = R_e/(1+2e-3e\sin^2 \lambda)$ the principal curvature radius in the vertical plane with the unitary plane. The velocity error $\delta V = (\delta V_x, \delta V_y, \delta V_z)$, attitude error, accelerometer bias and gyro bias are selected as state variables, and the difference between velocity and attitude calculated by high precision inertial unit and low precision inertial unit is taken as observation variables. The state equation and measurement equation are established as follows.

$$\dot{X} = FX + W$$
$$Z = HX + V$$

As $X = [\delta V_x, \delta V_y, \delta V_z, \phi, \phi, \phi, \rho, \rho, \rho, \epsilon, \epsilon, \epsilon]^T$, $W = [w_{\delta V_x}, w_{\delta V_y}, w_{\delta V_z}, w_{\phi}, w_{\phi}, w_{\phi}, w_{\rho}, w_{\rho}, w_{\rho}, w_{\epsilon}, w_{\epsilon}, w_{\epsilon}]^T$, $V = [V_x, V_y, V_z, V_x, V_y, V_z]^T$, $H = \begin{bmatrix} I_{2 \times 2} & 0_{2 \times 3} & 0_{2 \times 2} & 0_{2 \times 3} \\ 0_{2 \times 2} & I_{3 \times 3} & 0_{2 \times 2} & 0_{3 \times 3} \end{bmatrix}$.
In simulation, each element can be regarded as zero-mean white noise which is not correlated with each other.

4. Analysis of simulation experiment

The initial position of the carrier is longitude $116.3436^\circ$, latitude $39.9775^\circ$, constant zero offset $0.01^\circ/\text{h}$, random drift $0.005^\circ/\text{h}$, constant zero offset of accelerometer $10\mu g$ and random offset $5\mu g$ of high precision inertial gyroscope. Low precision inertial gyroscope constant zero offset $0.1^\circ/\text{h}$, random drift $0.05^\circ/\text{h}$, accelerometer constant offset $100\mu g$, random offset $50\mu g$. Under the initial condition, the carrier system coincides with the reference system, and the initial attitude error is $5^\circ$. The output data of two inertial units in a given maneuvering state are generated by the trajectory generator, and the error estimation process is simulated and analyzed by using the generated data. The specific process is as follows:

Trajectory setting: the first 20 seconds of the carrier moves in a straight line along the north with the initial velocity of 0, the acceleration of $0.5 \text{ m/s}^2$, and the attitude angle of the carrier changes according to the formula as follow. Starting from the 30th seconds, the carrier decelerates with $-0.1 \text{ m/s}^2$. After the 50th seconds, the carrier moves at a uniform speed, the 70s carrier accelerates with $0.1 \text{ m/s}^2$, the 90th carrier decelerates with $-0.1 \text{ m/s}^2$, and the carrier moves at a uniform speed until the 1st seconds. 20s is over.

The amplitude $A_\theta = 5^\circ$, $A_\phi = 5^\circ$, $A_\psi = 15^\circ$ of pitch angle $\theta$, roll angle $\phi$ and azimuth angle $\psi$ in the formula is $30^\circ$, $35^\circ$ and $45^\circ$.

In the maneuvering state of the above-mentioned carrier, the estimation results of the parameters related to the attitude calculation of the low-precision inertial navigation system are shown in Fig. 2 to 4, taking the high-precision inertial navigation system as the external reference information and the velocity and attitude information as the observation variables.
Fig. 2 Attitude error estimation results

Fig. 3 Error estimation results of accelerometer

Fig. 4 Error estimation results of gyro
By analyzing the simulation results, it can be seen that using high precision inertial units as external reference information can realize the error estimation of the attitude calculation parameters of low precision inertial units. Among them, the estimation accuracy of attitude error is relatively high, which can be within reach. The estimation process of accelerometer constant bias is more stable and the precision is higher, and gyro constant bias can basically be estimated. On the basis of error estimation, error correction is realized by subtracting the measured or calculated values of relevant parameters from the corresponding estimates.

5. Conclusion
According to the disadvantage of easy divergence of Strapdown calculation parameters of low-precision inertial units, a method of error correction assisted by high-precision inertial units is studied, a model of high-precision inertial units' auxiliary error correction is established, and simulation experiments are carried out. The results show that the error correction of Strapdown calculation parameters of low-precision inertial units can be realized under the given motion state, in which the attitude error and zero bias estimation of accelerometer constant value can be realized. The accuracy of the gyroscope is high, and the constant bias of the gyroscope can be estimated basically. The error correction method assisted by high precision inertial navigation system can provide a reference scheme for error correction of low precision inertial navigation system parameters. It is suitable for weapon system with multi-precision inertial navigation system and is less affected by electromagnetic interference and meteorological environment.

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