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GNSS DISTRIBUTED AUTONOMOUS ORBIT DETERMINATION METHOD BASED ON INTER-SATELLITE TWO-WAY RANGING

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Abstract

In order to improve the ephemeris accuracy of GNSS satellite, this paper proposes a two-step Kalman method for autonomous orbit determination. Firstly, the paper gives the distributed autonomous orbit data process, and then builds the bidirectional pseudo-range measurement model for GNSS navigation constellation. As the direct Kalman filter algorithm is difficult to detect abnormal observation, the two-step Kalman filter is adopted. The first step is to obtain satellite position and covariance by single point positioning method, which is considered as the pseudo-range measurement in second step to calculate the orbital correction of satellite, and then update orbital and time parameters. According to the problem of rank deficiency of autonomous orbit determination problem, the article adopts the orbit orientation parameters to constraint the constellation rotation, and gives the best estimate orbit of the satellite. The simulation and analysis results illustrate the feasibility the proposed algorithm.

Introduction

As an important strategic space resource, Global Navigation Satellite Systems (GNSS) has been widely infiltrated into all areas of national defence and economic construction. It has been an important infrastructure to ensure national security and promote economic and social development.

Inter-Satellite Links (ISL) is one of key technologies in GNSS autonomous navigation, whose main purpose is to carry on high-precision ephemeris and clock measurement, thereby providing high-precision PNT services. ISL autonomous orbit determination refers to maintain autonomous ephemeris update through inter-satellite ranging and communication without the ground operations control systems. The current studies are generally three directions: GPS satellites autonomous orbit determination research, rank deficiency problem, mathematical simulation.

The concept of satellite autonomous orbit determination is first proposed by America. In 1984, Ananda proposed the basic framework of satellite autonomous orbit determination by GPS inter-satellite observation [1]. Subsequently, Codik discussed the feasibility of GPS autonomous navigation [2]. Next, IBM Company is commissioned by the US Air Force aerospace to study the autonomous orbit determination algorithm. Until June 1990, the autonomous orbit determination theory, design, and data simulation work is almost completed [3]. As USA modern weaponry is increasingly dependent on GPS, a series of progressive researches for GPS Block IIR [4,5], Block IIF [6,7], Block III [8,9] are carried out to guarantee precision of autonomous orbit determination of the GPS system.

Ananda proposed that autonomous orbit determination is rank deficient, that is, due to the nature of inter-satellite ranging is relative measure, the autonomous orbit determination exist constellation rotation problems [3]. Liu theoretically proved that the unobservability of constellation rotation lead to the lack of reference information, so that the coefficient matrix equations method is rank deficiency [10]. Until now, solutions of this problem is broadly divided into two types: one is constraining directed parameters to control orbit constellation rotation [11,12,13]; the other is to introduce the anchor station as space reference system to control the rotation of the whole constellation, while improving the autonomous positioning accuracy [14,15,16].

For mathematical model and simulation of autonomous orbit determination problem, mainly includes the new algorithm, evaluation indexes, simulation platform and so on. Song adopted simulation data to verify two-step Kalman filter algorithm and SRIF algorithm for distributed autonomous ephemeris update problems [17]. Eissfeller proved that satellite ranging can be significantly improved the accuracy of radial and normal direction. When the range accuracy is in the level of centimetre, satellite autonomous orbit determination accuracy is better than 0.1m [18]. Zeng established simulation platform of autonomous orbit determination for navigation satellite, and proposed the static successive adaptive filtering method that can overcome the model error and equation linearization error of autonomous orbit determination [19].

In summary, existing researches has made a lot of valuable achievements, and USA GPS autonomous orbit determination has been put into operation. However, studies mainly confine to the partial key technologies and solutions, but have not achieved a complete engineering solution idea. Therefore, this article study systematically the autonomous orbit determination problem based on inter-satellite links.

Data Procedure of Autonomous Orbit Determination

Autonomous orbit determination based on ISLs is composed of the data communication, data pre-processing, residual generation, Kalman filtering, constellation rotation correction and ephemeris fitting. Data procedure is shown in Fig. 1, and the function of each module as follows:

Data Communication is mainly to complete interaction information between satellite, including observation data, satellite ephemeris and covariance information, satellite orbit plane orientation parameters. This paper refers to the bidirectional inter-satellite pseudo-range.

After ranging, data pre-process is mainly to complete gross error elimination, satellite observation time imputed, ranging observations and measurement error correction hardware delay correction.

Process residual is mainly to complete correcting error caused by ionosphere, calculating theoretical value, decoupling of ranging data and time synchronization data between satellite and satellite. The quality of observation data can be seized by residual analysis. The inputs in residual generation processing are observation data after pre-treatment acquired by the inter-satellite communications, correction value generated by last orbit prediction and reference orbit uploaded by the ground control system. The output is the difference between the observation and the theoretical value.

The main advantages of the Kalman filter observations and its main measurement error to update.

Using constellation rotation correction amount of satellite orbit parameters to improve the orientation, which was calculated by the ascending node longitude and orbit inclination correction value of each satellite orbit.

Broadcast ephemeris and clock information used for the next observation epoch, which ultimately generated by the autonomous orbit determination process.

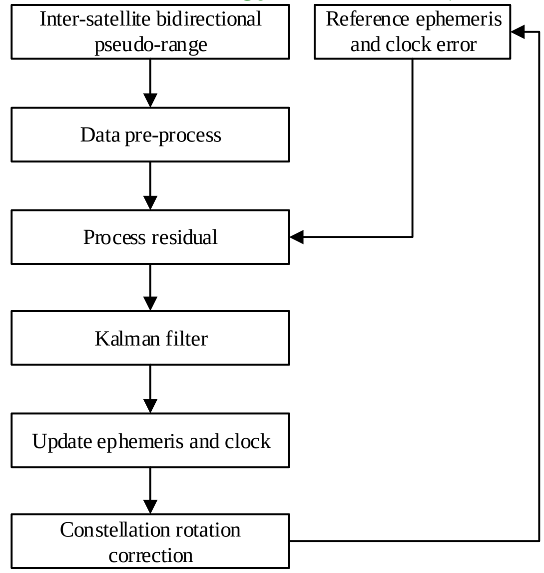

Figure 1. Data procedure of autonomous orbit determination

For navigation satellites, ephemeris prediction accuracy of the system is main technical indicator. Dynamic Orbit is a good choice for navigation satellite autonomous orbit determination to orbit forecasting easily. Autonomous Orbit Determination of ISL first need to build inter-satellite distance measurement equation, and then calculate satellite state transition matrix based on kinetic equations corresponding variational equations, and finally use Kalman filter to update parameters and the time. Considering the whole constellation rotation error correction generate the next observation epoch ephemeris and clock information. The estimated parameter can be satellite state or the orbital elements.

Algorithm Descriptions

A. Bidirectional pseudo-range measurement model of ISL

The basic observed value based on ISL of satellite autonomous operation is the measurements for satellite ranging. Satellite ranging observation has a variety of ways, such as inter-satellite laser ranging, inter-satellite pseudo-ranging, inter-satellite carrier phase measurements and inter-satellite Doppler measurements. For example, we have among the most commonly used satellite pseudo-range measurements to elaborate satellite ranging between observation equation and major errors involved.

Considering inter-satellite time synchronization error, the pseudo-range measurement equation is abbreviated as follows:

$$\rho_{ij} = R_{ij} + c \cdot \delta t_i - c \cdot \delta t_j + \left(d_{ri} + d_{clyi}\right) + \left(d_{tj} + d_{clyj}\right) + d_{ion} + d_{rel} + d_{mul} + \varepsilon \quad (1)$$

where ρ_{ij} is the pseudo-range value of satellite i received from satellite j at the observing time. R_{ij} is the theoretical distance between satellite i and j. δt_i and δt_j are clock error of satellite i and j respectively. d_{ri} is the receiving delay of satellite i. d_{tj} is the sending delay of satellite j. d_{clyi} and d_{clyj} are noise delay caused by temperature changes of receive and send terminal respectively. d_{ion} is the ionosphere delay. d_{rel} is the relativistic effect. d_{mul} is the multi-path effect. ε is the random measurement error.

After the transceiver delay correction, dual-frequency ionosphere correction, relativistic corrections, and multi-path correction of inter-satellite pseudo-range measurements, the bidirectional pseudo-range measurement equation of satellite i and j can be abbreviated as follows:

$$\rho_{ij} = R_{ij} + c \cdot \delta t_i - c \cdot \delta t_j + \varepsilon_{ij} \tag{2}$$

$$\rho_{ji} = R_{ji} + c \cdot \delta t_j - c \cdot \delta t_i + \varepsilon_{ji} \tag{3}$$

where R_{ij}、 R_{ji} are the theoretical distances between satellite i and j, which expressed as

$$R_{ij} = R_{ji} = \sqrt{(x_i - x_j)^2 + (y_i - y_j)^2 + (z_i - z_j)^2} \tag{4}$$

In order to eliminate the effect of orbit determination by the satellite clock error, the bidirectional pseudo-range measurement method is adopted. Add Eqs. (2) and (3), and linearize the reference orbit, the linear observation equation without clock error can be obtained as

$$\rho = \frac{\rho_{ij}+\rho_{ji}}{2} = \bar{R}_{ij} + \sum_{m=1}^{3} L_m (dr_i)_m - \sum_{m=1}^{3} L_m (dr_j)_m + \varepsilon' \tag{5}$$

where $\bar{R}_{ij}$ is the approximate distance which is calculated by the reference orbit. dr_i and dr_j are the correction of satellite position relative to reference orbit. L_m is the unit vector of inter-satellite distance, whose form can be written as $\left[\frac{x_i - x_j}{\bar{R}_{ij}} \quad \frac{y_i - y_j}{\bar{R}_{ij}} \quad \frac{z_i - z_j}{\bar{R}_{ij}}\right]$.

If the orbital elements are chosen as the estimated parameters, Eq. (5) can be written as:

$$\rho = \frac{\rho_{ij} + \rho_{ji}}{2} = \bar{R}_{ij} + \sum_{m=1}^{6} (H_i)_m (de_i)_m - \sum_{m=1}^{6} (H_j)_m (de_j)_m + \varepsilon' \tag{6}$$

Where $(de_i)_m$ and $(de_j)_m$ are the correction of orbital elements relative to reference orbit. $(H_i)_m$ and $(H_j)_m$ are the partial derivative of observed value relative to the orbital elements, the relationship between H and L_m are

$$(H_i)_m = L_m \frac{\partial (r_i)_m}{\partial (e_i)_m^T} \tag{7}$$

Where $\frac{\partial (r_i)_m}{\partial (e_i^T)_m}$ is the partial derivative of position r_i relative to the orbital elements e_i, expressed as

$$\frac{\partial r}{\partial e^T} = \begin{pmatrix} \frac{\partial r}{\partial a} & \frac{\partial r}{\partial i} & \frac{\partial r}{\partial \Omega} & \frac{\partial r}{\partial \alpha} & \frac{\partial r}{\partial \beta} & \frac{\partial r}{\partial \gamma} \end{pmatrix} \tag{8}$$

The expressions of six elements[20] in Eq. (8) can be written as

$$\begin{cases} \frac{\partial r}{\partial a} = \frac{r}{a}, & \frac{\partial r}{\partial \alpha} = Ar + B\dot{r} \\ \frac{\partial r}{\partial i} = G, & \frac{\partial r}{\partial \beta} = Cr + D\dot{r} \\ \frac{\partial r}{\partial \Omega} = Q, & \frac{\partial r}{\partial \gamma} = \sqrt{a^3}\dot{r} \end{cases} \tag{9}$$

Where A, B, C, D, G, Q are the function of the orbital elements α, β, Ω and u, r, which detailed in literature [20].

B. Orbit dynamics model

Considering the non-spherical gravitational perturbations of the Earth, then the orbit dynamics model [20] can be expressed as follows:

$$\begin{cases} \frac{dx_i}{dt} = v_{x_i} \\ \frac{dy_i}{dt} = v_{y_i} \\ \frac{dz_i}{dt} = v_{z_i} \\ \frac{d^2x_i}{dt^2} = -\frac{\mu x_i}{{r_i}^2}\left[1 + \frac{3}{2}J_2\left(\frac{R_e}{r_i}\right)^2\left(1 - 5\left(\frac{z_i}{r_i}\right)^2\right)\right] + \Delta F_{x_i} \\ \frac{d^2y_i}{dt^2} = -\frac{\mu y_i}{{r_i}^2}\left[1 + \frac{3}{2}J_2\left(\frac{R_e}{r_i}\right)^2\left(1 - 5\left(\frac{z_i}{r_i}\right)^2\right)\right] + \Delta F_{y_i} \\ \frac{d^2z_i}{dt^2} = -\frac{\mu z_i}{{r_i}^2}\left[1 + \frac{3}{2}J_2\left(\frac{R_e}{r_i}\right)^2\left(1 - 5\left(\frac{z_i}{r_i}\right)^2\right)\right] + \Delta F_{z_i} \end{cases} \tag{10}$$

wherer $r_i = \sqrt{{x_i}^2 + {y_i}^2 + {z_i}^2}$is the satellites position vector length, $x_i, y_i, z_i, v_{x_i}, v_{y_i}, v_{z_i}$represent the position and velocity components of satellite in inertial, μ is Earth gravitational constant, J_2 is second-order harmonic coefficient, R_e is Earth equatorial radius, $\Delta F_{x_i}, \Delta F_{y_i}, \Delta F_{z_i}$ indicates satellite J2 items outside perturbation acceleration, mainly noise representation model.

C. Two-step Kalman filter

The nature of the distributed orbit determination is the process of orbit determination using inter-satellite pseudo-range, where the prediction orbit by satellite as a reference point. Thus, the distributed inter-satellite pseudo-range measurement equation by modified Eq. (6) can be written as

$$\Delta\rho = \frac{\rho^{ij} + \rho^{ji}}{2} + \sum_{m=1}^{6} (H_j)_m (de_j)_m - \bar{R}_{ij}$$
$$= \sum_{m=1}^{6} (H_i)_m (de_i)_m + \varepsilon' \quad (11)$$

In the autonomous orbit determination problem, if gross errors in observed value or transferring orbit leading to exception, the direct Kalman filter algorithm is difficult to detect the abnormal observation. Therefore, the two-step approach is used in this paper. The first step is using single point positioning method to calculate the satellite position and covariance. The second step is the satellite position and covariance information as the pseudo-range measurement to carry on the Kalman filter algorithm processing. Eq. (11) can be written as:

$$\Delta\bar{\rho} = H\Delta X + \bar{\varepsilon}' \quad (12)$$

Where $\Delta\bar{\rho}$ is the observation, ΔX is the corrections of satellite position. considering the observations covariance, the satellite position corrections and covariance can be calculated by the weighted least square method as follows:

$$\Delta X = (H^T P H)^{-1} H^T P \Delta\bar{\rho} \quad (13)$$
$$E(\Delta X \cdot \Delta X^T) = \sigma^2 (H^T P H)^{-1} \quad (14)$$

Where $P = \left(E(\bar{\varepsilon}' \cdot \bar{\varepsilon}'^T)\right)^{-1}$ is the weight matrix of observation, σ is the units weight error.

Linearized the orbit dynamics model (10) at the reference orbit, and represented the higher order error by a random noise, Eq.(15) can be obtained as

$$\Delta\bar{q}_i = \bar{\Phi}_{i+1,i} \Delta q_i + \Gamma\omega \quad (15)$$

Where $\bar{\Phi}_{i+1,i}$ is the amplified state transition matrix, which is defined as:

$$\bar{\Phi}_{i+1,i} = \begin{bmatrix} \Phi_{i+1,i} & H_{i+1,i} \\ 0 & I \end{bmatrix} \quad (16)$$

Where Φ is the state transition matrix, and its initial value is the unit matrix; H is the partial derivative of satellite state parameters relative to the kinetic parameters, and its initial value is zero; ω is the kinetics noise vector, where is content to $E(\omega) = 0$, $\bar{Q} = E(\omega\omega^T)$; Γ is the noise coefficient matrix; I is the identity matrix.

Taking satellite position corrections and covariance as pseudo-measurement, the Kalman filter equation according to the minimum variance principle can be obtained as

$$\Delta\bar{q}_k = \bar{\Phi}_{k+1,k} \Delta q_k \quad (17)$$
$$\bar{P}_{k+1} = \bar{\Phi}_{k+1,k} P_k \bar{\Phi}_{k+1,k}^T + \Gamma\bar{Q}_k\Gamma^T \quad (18)$$
$$\Delta q_{k+1} = \Delta\bar{q}_k + K_k(\bar{y}_k - H_k\Delta\bar{q}_k) \quad (19)$$
$$P_{k+1} = (I - K_k H_k)\bar{P}_{k+1} \quad (20)$$
$$K_k = \bar{P}_{k+1} H_k^T \left(H_k \bar{P}_{k+1} H_k^T + R_k\right)^{-1} \quad (21)$$

The satellite orbit parameters corrections can be calculated by Kalman filter algorithm, simultaneously, the time can be update.

D. Constellation rotation correction

Due to the inter-satellite distance is a relative measure in essential, autonomous orbit determination by inter-satellite range measurement is rank deficiency. Therefore, if the overall navigation systems which maintained by inter-satellite pseudo-range rotate at any angle, the navigation system performance does not change. Thus the autonomous orbit determination needs constraint constellation rotation. In this paper, we adopt orbit orientation parameters (forecast orbit ascending node longitude and orbit inclination) to constraint the constellation rotation, and give the best estimate orbit of the satellite. Specific methods are: six orbital parameters of each satellite is estimated, but constraint or correct orbit ascending node longitude and orbit inclination corrections of the entire constellation.

Suppose orbital orientation parameters i, Ω, ω corrections V of n satellites is determined by inter-satellite ranging data in certain epoch, and V is a $3 \times n$ vector. Since the satellite constellation rotation with inter-satellite ranging is unconstrained, the constellation around the three axes rotates $\theta_x, \theta_y, \theta_z$. According to the differential relationship between overall coordinate frame rotation and the orbital plane orientation parameters [17], the new orbit orientation corrections will be $(V - \bar{M}\vec{\theta})$, wherein $\bar{M}$ is n matrix composed of 3x3, three components of $\vec{\theta}$ is are $(\theta_x \quad \theta_y \quad \theta_z)$. Theoretically, V and $(V - \bar{M}\vec{\theta})$ is not better or worse, because the overall rotation is unpredictable. In order to ensure close to the true value V, select a reasonable $\vec{\theta}$ [17]. The error of i, Ω and the true value should be in a controllable range, $\vec{\theta}$ content as follows:

$$V - \bar{M}\vec{\theta} = 0 \quad (22)$$

Adopting the least squares principle, we have

$$\vec{\theta} = (\bar{M}^T W \bar{M})^{-1} \bar{M}^T W V \qquad (23)$$

where W is a priori weight matrix. Considering control error propagation of reference orbit, Menn [11] recommend W is chosen by a reference value. Thus the new satellite orbit orientation corrections can be written as $(V - \bar{M}\vec{\theta})$. The corresponding constraint equation can be obtained as

$$\bar{M}^T W V = 0 \qquad (24)$$

When the orbit orientation parameters requires only constraint i, Ω, correspond component value of W can be set to 1, and the rest set to zero.

Simulation and Analysis

According to the GPS ephemeris by international GNSS service (IGS), the autonomous orbit determination method of this paper is validated. Analyse the accuracy of autonomous orbit determination at 180 days by the user range error (URE) [1], which is approximated as:

$$URE = \sqrt{R^2 + \frac{1}{49}(T^2 + N^2)} \qquad (25)$$

where R represents a radial error, T indicates the tangential error, N is the normal error.

Simulations are carried out in two situations: one is all orbital elements of satellites involved in filter, and the simulation results are shown in Figs. 2-5, the other is the orientation parameters i, Ω constraint constellation rotation, and the simulation results are shown in Figs. 6-9.

A. case 1: all orbital elements involved in filtering

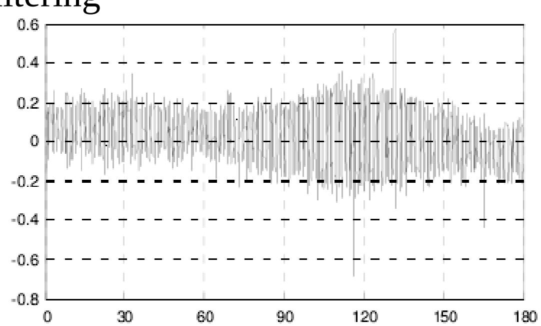

Figure 2. Radial error (meter/day)

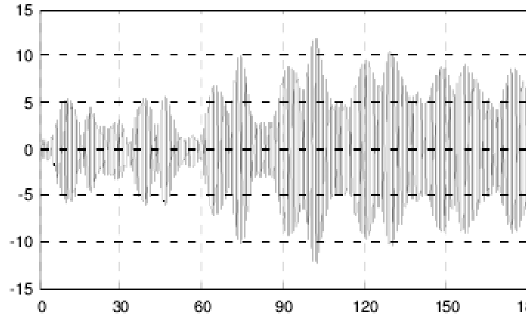

Figure 3. Normal error (meter/day)

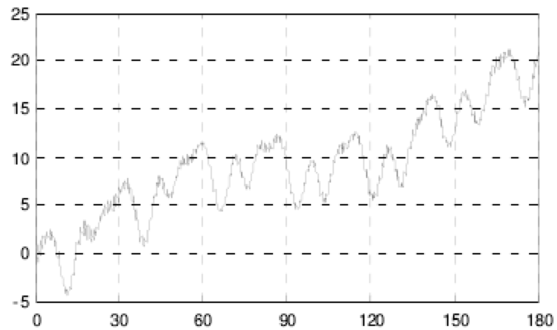

Figure 4. Tangential error (meter/day)

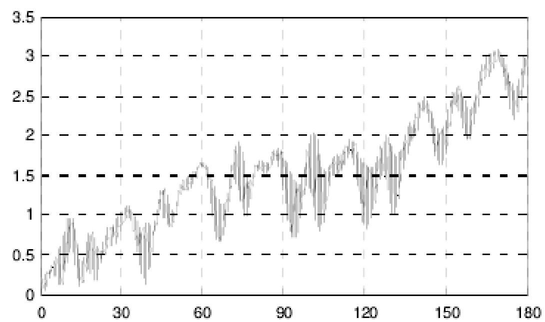

Figure 5. User ranging error (meter/day)

As can be seen from Figs. 2-5, in the given 180-day orbit arcs, the radial error always maintain high accuracy within 0.4m, while the normal error and the tangential error are grow rapidly over time, but the corresponding UREs are less than 3.5m.

In order to reflect the autonomous orbit determination precision more clearly, the URE of satellites PRN30 was statistic, its mean square error is 0.7028m, and the maximum error is 3.0852m, the average error is 1.4752m.

B. Case 2: constraint the orientation parameters i, Ω

Constraining orientation parameters i, Ω of PRN30 to simulate, that is, the orientation parameters i, Ω are not involved in filtering, whose estimate value is replaced by the priori value of orbit forecast. The other simulation conditions is same with case 1.

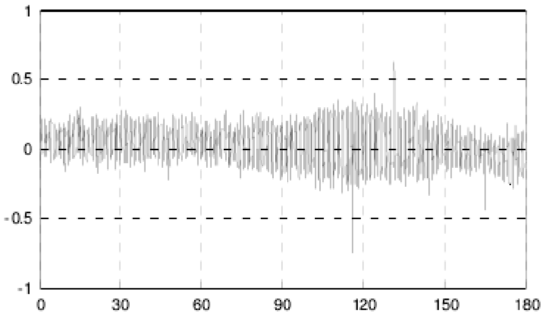

Figure 6. Radial error (meter/day)

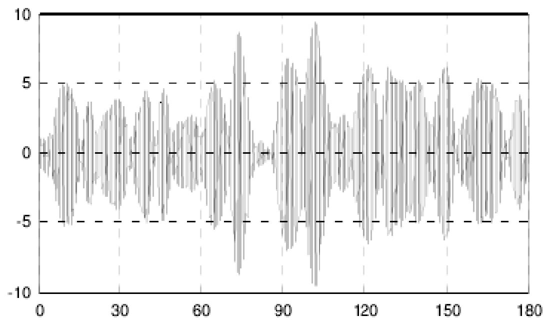

Figure 7. Normal error (meter/day)

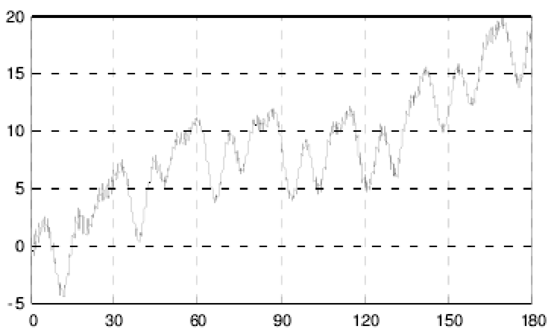

Figure 8. Tangential error (meter/day)

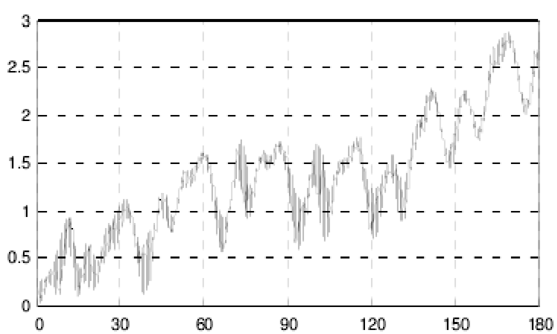

Figure 9. User ranging error (meter/day)

Compared with Figs. 2-5, it is clearly seen from Figs. 6-9 that the radial error remained changeless, the upper and lower limit of the normal error dropped from 15m 10m, and the upper limit of the tangential error fall from 25m to 20m, and the upper limit of the user range error dropped from 3.5m to 3m.

In order to reflect the orbit determination precision more clearly, compare the URE of two cases listed in Tab.1as follows.

Table 1. URE of all orbital elements in filtering compared with URE of constraint orientation parameters i、Ω

Simulation cases	URE（m）		
	mean square error	maximum error	average error
all orbital elements involved in filtering	0.7028	3.0852	1.4752
constraint the orientation parameters i, Ω	0.6346	2.8757	1.3194
The difference of two cases	0.0682	0.2095	0.1558

As can be seen from Tab. 1, the orbit determination accuracy is improved of case 2 compared with case 1. The URE mean square error improves 0.0682m, the maximum error improves 0.2095 m, and the average error improves 0.1558m. This further shows the effectiveness of the proposed algorithm.

Conclusion

This paper studies GNSS autonomous orbit determination problem based on ISL systematically. The data procedure is given in autonomous orbit determination. According to the inter-satellite pseudo-range measurement model, a two-step Kalman filter is designed and the constellation rotation constraint is studied. The simulation results verify the feasibility of the proposed scheme.

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Biography

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