Simulation Research of BD Satellite Navigation Based on MATLAB

Tong Chang
Undergraduate building 22, University of Electronic Science and Technology of China, Chengdu, Sichuan, 611731, China
332081261@qq.com

Abstract. This paper studies and analyzes the key technical principles of BD2 satellite orbit modeling. It uses MATLAB to design and simulate the satellite constellation and satellite positioning principle of Beidou. It can display the selected orbit and the satellite used for user positioning. It realizes the simple carrier motion modeling and studies the ionospheric delay correction model. Through the testing and analysis of the key technologies of the system, it shows that the system can realize the navigation and positioning function, which has practical value.

1. Introduction
The Beidou satellite navigation system is a global satellite navigation system independently developed by China. The system is dedicated to providing users with high-quality global positioning and navigation. China's satellite navigation technology plays an important role in China's national defense and the national economy. The satellite navigation system can provide basic positioning and navigation for people. In addition, the construction of the Beidou satellite navigation system is important for security, for promoting economic and social development and for improving China's international status. With the independent Beidou satellite navigation system providing corresponding navigation, timing, force received and short message communication services, China will no longer be restricted by core technology.

In the first chapter, the research background of this paper, the research status of satellite navigation simulation technology and the main research contents of the thesis are briefly introduced. In the second chapter, we study and analyze the calculation model of the instantaneous position of the Beidou satellite and the key techniques such as the Chebyshev polynomial fitting algorithm [1-7]. And through the simulation of the observation point in the static state, the observation model’s [8] rationality is verified. In the third chapter, the accuracy of the simple carrier motion trajectory simulation principle [9] is verified by the user motion trajectory simulation and its corresponding position coordinate values. In the fourth chapter, the ionospheric Klobuchar delay correction model [10] is studied and simulated. At the end of this article, summarize this paper and analyze the inadequacies.

2. Beidou satellite orbit modeling
2.1. Orbital model analysis
When the satellite moves in space, besides being subjected to the gravitational field of the Earth, it is also subjected to the gravitational force of other celestial bodies. The orbit is very complicated and it is difficult to accurately represent it mathematically [1]. Among all the perturbative forces received by
satellite, the Earth's gravitational field has the greatest influence on the power of the gravitational field. Therefore, the orbital parameters of the simulated satellite mainly consider the Earth's gravitational field. According to Kepler’s law [2]: the orbit of satellite motion is an ellipse through the plane of the earth’s center, a focus of the ellipse coincides with the center of the earth. An ideal elliptical orbit can be represented by the following six orbital parameters (Kepler orbital parameters):

- Orbital semimajor axis: \(a\);
- Orbital eccentricity: \(e\); The size and shape of the elliptical orbit of the satellite motion is determined by the semi-major axis and the eccentricity of the orbit.
- Orbital inclination: \(i\); The dihedral angle of the orbit and the equatorial plane;
- RAAN: \(\Omega\); The angle between the vernal equinox and the ascending node on the equatorial plane of the Earth.
- Argument of perigee: \(\omega\); The Angle of the center of the earth between the perigee and the ascending node on the orbital plane. \(\Omega\), \(i\) and \(\omega\) determine the relative orientation and position of the satellite orbital plane and the Earth.
- True anomaly: \(f\); The angle between the satellite and the perigee on the orbital plane. The true anomaly determines the position of the satellite on the orbital plane.

### 2.2. Beidou satellite orbit simulation

The simulation of the satellite orbit is completed in the following five steps.

1) Establish a coordinate system

   Taking the earth as the origin, space right-handed rectangular coordinate system is established. The positive half-axis of the \(x\)-axis points to the east longitude direction, the positive half-axis of the \(y\)-axis points to the north pole, and the \(z\)-axis axis is determined by the \(x\)-axis and the \(y\)-axis direction.

2) Create an ellipse

   In the equatorial plane of the Earth, create an ellipse with a semi-major axis of \(a\) and an eccentricity of \(e\). Place the earth at a focus of the ellipse.

3) Set the orbital inclination

   According to the definition of the orbital inclination, the rotation matrix is obtained, and the ellipse is raised around the \(x\) coordinate axis by the angle \(i\).

\[
R_i = \begin{bmatrix}
1 & 0 & 0 \\
0 & \cos i & \sin i \\
0 & -\sin i & \cos i
\end{bmatrix}
\]  

(2.1)

4) Set the RAAN

   According to the definition of the RAAN, the rotation matrix is obtained

\[
R_\Omega = \begin{bmatrix}
\cos \Omega & 0 & -\sin \Omega \\
0 & 1 & 0 \\
\sin \Omega & 0 & \cos \Omega
\end{bmatrix}
\]  

(2.2)

5) Set the argument of perigee

   According to the definition of the argument of perigee, the rotation matrix is obtained. The earth is first moved to a focus of the ellipse, and the orbit is rotated by \(\omega\) around the axis that passes through the center of the earth and is perpendicular to the orbital plane.

\[
R_\omega = \begin{bmatrix}
\cos \omega + B_\omega^2(1 - \cos \omega) & B_\omega B_y(1 - \cos \omega) - B_z \sin \omega & B_x B_z(1 - \cos \omega) + B_y \sin \omega \\
B_x B_y(1 - \cos \omega) + B_z \sin \omega & \cos \omega + B_\omega^2(1 - \cos \omega) & B_y B_z(1 - \cos \omega) - B_x \sin \omega \\
B_x B_z(1 - \cos \omega) - B_y \sin \omega & B_y B_z(1 - \cos \omega) + B_x \sin \omega & \cos \omega + B_\omega^2(1 - \cos \omega)
\end{bmatrix}
\]  

(2.3)

\[
B = A R_i R_\Omega
\]

(2.4)

### 2.3. Beidou Ephemeris Parameter Simulation Model

In fact, in satellite navigation, the satellite orbit is calculated by the ephemeris parameters in the navigation message issued by the satellite. The Beidou satellite orbital model in this section is based on
16 parameters, including 6 Kepler orbital parameters, 6 tuning coefficients, 1 orbital inclination rate correction parameter, 1 ascending node right ascension rate correction parameter, and 1 average angular velocity correction parameter and 1 ephemeris data reference moment [3]. The representation of the 16 parameters and the specific meanings are shown in the table:

| Parameter       | Description                                                            |
|-----------------|------------------------------------------------------------------------|
| \( t_{oe} \)    | Ephemeris reference time                                               |
| \( \sqrt{a} \)  | The square root of the orbital semimajor axis                          |
| \( e \)         | Eccentricity                                                          |
| \( i \)         | Orbital inclination                                                   |
| \( \Delta n \)  | The difference between the average moving rate of the satellite and the calculated value |
| \( M_0 \)       | Mean anomaly of the reference time                                     |
| \( \Omega_0 \)  | RAAN of the reference time                                             |
| \( OMEGADOT \)  | Changing rate of RAAN                                                  |
| \( \omega \)    | Argument of perigee                                                   |
| \( IDOT \)      | Rate of change of orbital inclination                                  |
| \( C_{us} \)    | The amplitude of Latitude Angle sine harmonic correction term           |
| \( C_{uc} \)    | The amplitude of Latitude Angle cosine harmonic correction term         |
| \( C_{rc} \)    | The amplitude of radial cosine harmonic correction term                 |
| \( C_{is} \)    | The amplitude of orbital inclination angle sine harmonic correction term|
| \( C_{ic} \)    | The amplitude of orbital inclination angle cosine harmonic correction term|
| \( C_{rs} \)    | The amplitude of radial sine harmonic correction                        |

Among all the perturbations received by satellite, the Earth's gravitational field has the greatest influence on the perturbation, and it produces a perturbation acceleration of about \( 5 \times 10^{-5} \text{m/s}^2 \), which affects a satellite's periodic orbital arc of approximately 6 km [4]. Other perturbations have much less impact [5].

1) causing the rotation of the orbital plane in space [6]

Under the gravitational force of the Earth, the satellite's ascending node moves slowly along the equator, causing periodic change \( \dot{\Omega} \) in the RAAN \( \Omega \).

\[
k = \frac{n + J_2}{2} \left\{ \frac{R_e}{a(1-e^2)} \right\} \cdot -3 \ast k \cos(i)
\]

\[
\dot{\Omega} = -3 \ast k \cos(i)
\] (2.5)

\( n \): Average angular velocity of satellite motion;
\( J_2 \): The second-order harmonic coefficient which is \( 1.082645 \times 10^{-3} \);
\( R_e \): Earth's equatorial radius.

2) causing the rotation of the perigee in the orbital plane

In addition to the change in RAAN, it will also cause a slow change in the argument of perigee, and \( \dot{\omega} \) is used to indicate the rate of change of the argument of perigee.
\[
\dot{\omega} = 3 \cdot k(2 - \frac{5}{2} \sin^2(i))
\]  \hfill (2.6)

When orbital inclination \(i = 63^\circ 26\) or \(i = 116^\circ 34\), the value of \(\dot{\omega}\) is approximately 0.

3) causing changes in the mean anomaly \(M\) [7]

The satellite’s changing rate of orbital mean anomaly \(M\) under the influence of the second-order harmonic of the Earth's gravitational field can be approximated by the following equation.

\[
\dot{M} = 3 \cdot k \cdot \sqrt{1 - e^2} \cdot (1 - \frac{3}{2} \sin^2(i))
\]  \hfill (2.7)

When \(i = 54.73^\circ\), \(M\) is approximately 0.

4) calculation of basic orbital parameters [8] at the time of observation

Calculation about \(a(t), e(t), \Omega(t), \omega(t), M(t), i(t)\) at observation time \(t\):

\[
\begin{align*}
A(t) &= A(t_0) \\
e(t) &= e(t_0) \\
i(t) &= i(t_0) \\
\dot{\Omega}(t) &= \dot{\Omega}(t_0) + \dot{\Omega} \cdot (t - t_{oe}) \\
\dot{\omega}(t) &= \dot{\omega}(t_0) + \dot{\omega} \cdot (t - t_{oe}) \\
M(t) &= M(t_0) + M \cdot (t - t_{oe})
\end{align*}
\]  \hfill (2.8)

2.4. Beidou satellite instantaneous position calculation

The user receiver can calculate the position of the satellite through the ephemeris parameters [9]. The instantaneous position calculation process of the Beidou satellite is as follows.

1) Calculate the average angular velocity of the satellite \(n\)

\[
\begin{align*}
n_0 &= \sqrt{GM/a^3} \\
n &= n_0 + \Delta n
\end{align*}
\]  \hfill (2.9)

\(G\) is the Earth’s gravity constant of the Earth’s CGCS2000 coordinates;
- long semi-axle;
- The average angular velocity of the ephemeris reference moment.

2) Calculate the normalization time \(t_k\)

\[
t_k = t - t_{oe}
\]  \hfill (2.10)

\(t\) —— the observation time input by the user. Since the BD time system is based on the number of seconds in the week:

\[
\begin{align*}
t_k &= t_k - 604800, \quad t_k > 302400 \\
t_k &= t_k + 604800, \quad t_k < -302400
\end{align*}
\]  \hfill (2.11)

3) Calculate the mean anomaly \(M\)

\[
M = M_0 + n \cdot t_k
\]  \hfill (2.12)

4) Calculate the eccentric anomaly \(E\)

\[
E = M + e \cdot \sin(E)
\]  \hfill (2.13)

Set the initial value \(E_0\) of \(E\) to \(M\), and perform iterations several times to calculate \(E\).

5) Calculate the true anomaly \(f\)

According to the "two-body problem" formula:

\[
\begin{align*}
\cos(f) &= (\cos(E) - e)/(1 - e \cdot \cos(E)) \\
\sin(f) &= (\sqrt{1 - e^2} \cdot \sin(E))/(1 - e \cdot \cos(E)) \\
f &= \arctan((\sqrt{1 - e^2} \cdot \sin(E))/(\cos(E) - e))
\end{align*}
\]  \hfill (2.14)
\[ f = \arctan \left( \frac{\sin(f)}{\cos(f)} \right), \cos(f) > 0 \text{ and } \sin(f) > 0 \]

\[ f = 2 \pi - \arccos(\cos(f)), \cos(f) > 0 \text{ and } \sin(f) < 0 \]  

\[ f = \arccos(\cos(f)), \cos(f) < 0 \text{ and } \sin(f) > 0 \]  

\[ f = \pi - \arcsin(\sin(f)), \cos(f) < 0 \text{ and } \sin(f) < 0 \]  

(2.16)

6) Calculate RAAN \( \Omega \)

\[ \Omega = \Omega_0 + (\text{OMEGA}\text{DOT} - \omega_e) * t_k - \omega_e * t_{oe} \]  

(2.17)

\[ \omega_e : \text{ the angular velocity of the earth.} \]

7) Calculate the argument of latitude \( \Phi \) of MEO/EGSO

\[ \Phi = f + \omega \]  

(2.18)

8) Calculate the orbital perturbation error

\[ \Delta u = C_{us} * \sin(2 * \Phi) + C_{uc} * \cos(2 * \Phi) \]

\[ \Delta r = C_{rc} * \sin(2 * \Phi) + C_{rc} * \cos(2 * \Phi) \]

\[ \Delta i = C_{ic} * \sin(2 * \Phi) + C_{ic} * \cos(2 * \Phi) \]  

(2.19)

\[ \Delta u: \text{ correction term of argument of latitude;} \]

\[ \Delta r: \text{ Radial correction term} \]

\[ \Delta i: \text{ Orbital inclination correction term} \]

9) Calculate the corrected argument of latitude \( u \)

\[ u = \Phi + \Delta u \]

\[ r = a^2 * (1 - e * \cos(E)) + \Delta r \]

\[ i = i_0 + IDOT * t_k + \Delta i \]  

(2.20)

10) Calculate the coordinates of the satellite in a two-dimensional plane

\[ x = r * \cos(u) \]

\[ y = r * \sin(u) \]

\[ z = 0 \]  

(2.21)

11) Calculate the three-dimensional position of the MEO/IGSO satellite in the CGCS2000 coordinate system

\[ X = x * \cos(\Omega) - y * \cos(i) * \sin(\Omega) \]

\[ Y = x * \sin(\Omega) - y * \cos(i) * \cos(\Omega) \]

\[ Z = y * \sin(i) \]  

(2.22)

2.5. Chebyshev algorithm fitting satellite orbit

Since the satellite ephemeris update period of Beidou is one hour, there will be a certain error in the ephemeris parameters, and thus the calculated instantaneous position and instantaneous velocity of the
Beidou satellite will also have a certain error. Therefore, the simulation of satellite orbits needs to be approximated by a certain algorithm. This paper uses the Chebyshev fitting algorithm [10].

It is assumed that within the time interval \( [t_0, t_0 + \Delta t] \), the \( n \)-order Chebyshev polynomial is used to fit the approximation, where \( t_0 \) is the start time and \( \Delta t \) is the interval time. First, convert the time interval:

\[
\tau = \frac{2}{\Delta t} (t - t_0) - 1, \quad t \in [t_0, t_0 + \Delta t]
\]  

(2.26)

The instantaneous position of the satellite and the instantaneous velocity along the x-axis can be expressed as:

\[
x(t) = \sum_{i=0}^{n} C_{x} T_i(\tau)
\]

\[
\dot{x}(t) = \sum_{i=1}^{n} C_{x} T'_i(\tau)
\]

(2.27)

\( n \): the order of the Chebyshev polynomial

\( C_{x} \): Chebyshev polynomial coefficient

The calculation method of the components along the y-axis direction and the z-axis direction is similar to the x-axis direction. The expressions of \( T_i(\tau) \) and \( T'_i(\tau) \) in equation (2.27) are as follows:

\[
T_0(\tau) = 1
\]

\[
T_1(\tau) = \tau
\]

\[
T_n(\tau) = 2T_{n-1}(\tau) - T_{n-2}(\tau) \quad |\tau| \leq 1, \quad n \geq 2
\]

\[
T'_1(\tau) = \tau
\]

\[
T'_2(\tau) = 4\tau
\]

\[
T'_n(\tau) = \frac{2n}{n-1} \tau T'_{n-1}(\tau) - \frac{n}{n-2} T_{n-2}(\tau) \quad n \geq 3
\]

(2.28)

(2.29)

The coefficient expression of the Chebyshev polynomial can be obtained by the least square method fitting.

\[
C_{x} = (B^T B)^{-1} B^T X
\]

(2.30)

In equation (2.30):

\[
B = \begin{bmatrix}
T_0(\tau_1) & T_1(\tau_1) & T_2(\tau_1) & \cdots & T_n(\tau_1) \\
T_0(\tau_2) & T_1(\tau_2) & T_2(\tau_2) & \cdots & T_n(\tau_2) \\
\vdots & \vdots & \vdots & \ddots & \vdots \\
T_0(\tau_m) & T_1(\tau_m) & T_2(\tau_m) & \cdots & T_n(\tau_m)
\end{bmatrix}
\]

(2.31)

\[
X = [x_1 \quad x_2 \quad x_3 \quad \cdots \quad x_n]^T
\]

(2.32)

After obtaining the coefficient, the position and velocity of the satellite at any time in the corresponding time interval can be calculated using equation (2.27). The 13th-order Chebyshev polynomial is used for the simulation in this paper.

Using MATLAB to simulate the Earth, satellite orbits, I get a GIF image of the Earth spinning and the satellites moving around the Earth over time.
Figure 1. Spinning earth.

Figure 2. The earth rotates and the satellite moves around the earth in orbit.

Figure 3. Show selected satellite orbit.
2.6. User-visible satellite judgment principle

For the visualization of signal coverage, the coverage of a satellite signal must first be determined by calculation. It can be divided into two cases [11]: coverage of the signal to the ground user; coverage of the signal to the space user.

2.6.1 Judgment about ground users’ visibility to the satellite. Land users' visibility to the satellites began to rise off the surface of the Earth, and their visible capabilities gradually expanded. Considering the heat loss of the ground and the effects of atmospheric propagation caused by the different elevation angles of the receiver and the effects of ground obstructions, it is usually necessary to determine an effective range of observations, that is, to determine the minimum occlusion angle that the user receiver can provide [12], assumed to be $\vartheta$. Let user U’s position at time $t$ be $(X_u, Y_u, Z_u)$ and satellite S at position $(X_s, Y_s, Z_s)$. Then in the Earth Cartesian coordinate system: the user's position vector $\vec{P}_u$, the satellite's position vector $\vec{P}_s$, $\theta$ is the user's elevation angle relative to the satellite.

$$\vec{P}_u = X_u \hat{i} + Y_u \hat{j} + Z_u \hat{k}$$

$$\vec{P}_s = X_s \hat{i} + Y_s \hat{j} + Z_s \hat{k}$$

$$\vec{P}_{us} = \vec{P}_s - \vec{P}_u$$

$$\arccos \Theta = \frac{\vec{P}_u \cdot \vec{P}_{us}}{|\vec{P}_u||\vec{P}_{us}|}$$

When $\theta > (90^\circ + \vartheta)$, the satellite is observable; otherwise, the satellite is not within the user's view.

2.6.2 Judgment about space users’ visibility to the satellites. In the Earth Centered Earth Fixed system, set the coordinate of the satellite S $(X_s, Y_s, Z_s)$, the orbital height tangent to the radiation range of the satellite signal is $h$, and $\gamma$ is the half-radiation angle of the satellite signal, then:

$$h = (R + H_s) \sin \gamma - R$$

It is calculated that for medium Earth-orbiting satellites, the range of $h$ is about 500--5000 km. For geostationary orbit satellites and inclined geosynchronous orbit satellites, the range of $h$ is about 5000--10000 km. When the user’s track height is less than $h$, the coverage performance can be studied without the beam angle limit. When the user's orbital height is greater than $h$, both the beam angle and the occlusion angle must be considered.

![Figure 4](image-url) Determine the user's current location (6400,0,0) and show the satellites that are used.
3. Carrier motion modeling

3.1. Overview of carrier motion modeling
Carrier motion information simulation is an important part of the satellite navigation mathematics simulation system. I completed the simple motion modeling by mathematical model calculation [13], which is divided into spatial linear motion and uniform circular motion in the same plane. After the simple motion modeling is completed, it only needs to be combined in the time dimension to realize the design of the simple motion carrier [14].

3.2. Simple carrier motion modeling
1) Spatial linear motion
The reference time is the Beidou time system. The initial position is represented using the Earth's geodetic coordinate system, and the earth's Cartesian coordinate system is used for calculation. The mathematical principle is to decompose the initial position, initial velocity and acceleration into three directions of X, Y and Z according to the direction of motion, and perform uniform accelerated rectilinear motion in three directions. Suppose the coordinate at time t is \((x, y, z)\), the acceleration is \(a\), the initial velocity is \(v_0\), the elevation angle is \(\theta_1\), and the azimuth angle is \(\theta_2\). The specific calculation of the coordinates at time t is as follows:
2) Uniform circular motion in the same plane

The initial position is represented using the Earth’s geodetic coordinate system. The calculation uses the Earth’s Cartesian coordinate system. If the coordinate at time $t$ is $(x_0, y_0, z_0)$ and the velocity is $v$, the specific calculation of the coordinates at time $t$ is as follows:

\[
\begin{align*}
    x &= x_0 + v_0 \sin \theta_1 + \frac{1}{2} a \sin \theta_1 t^2 \\
    y &= y_0 + v_0 \cos \theta_1 \cos \theta_2 + \frac{1}{2} a \cos \theta_1 \cos \theta_2 t^2 \\
    z &= z_0 + v_0 \cos \theta_1 \sin \theta_2 + \frac{1}{2} a \cos \theta_1 \sin \theta_2 t^2
\end{align*}
\] (3.1)

Linear motion and uniform circular motion of the carrier is obtained by MATLAB.
4. The ionospheric delay correction model

The ionosphere of the peripheral atmosphere contains a large number of free electrons and positive ions, and such an ionized atmosphere refracts electromagnetic waves. For single-frequency receivers, corrections are typically made through the ionospheric model. This paper uses the Klobuchar model in the simulation process.

The ionospheric vertical delay correction of the B1I signal is $I_z(t)$ in seconds, as follows:

$$I_z(t) = \begin{cases} 
5 \times 10^{-9} + A_2 \cos[\frac{2\pi(t-50400)}{A_4}], & |t - 50400| < \frac{A_4}{4} \\
5 \times 10^{-9}, & |t - 50400| \geq \frac{A_4}{4}
\end{cases}$$

(4.1)

$t$: The intersection (puncture point M) of the ionosphere and the line which connects the receiver and the satellite, the unit is second.

$A_2$: Daytime ionospheric delay cosine curve amplitude $\alpha_n$, obtained by coefficient:

$$A_2 = \begin{cases} 
\sum_{n=0}^{3} \alpha_n |\phi_M|^n, & A_2 \geq 0 \\
0, & A_2 < 0
\end{cases}$$

(4.2)

$A_4$: The period of the cosine curve, in seconds, and is obtained by the coefficient $\beta_n$:

$$A_4 = \begin{cases} 
172800, & A_4 \geq 172800 \\
\sum_{n=0}^{3} \beta_n |\phi_M|^n, & 172800 > A_4 \geq 72000 \\
72000, & A_4 < 72000
\end{cases}$$

(4.3)

$\phi_M$ in the above two formulas is the geographical latitude of the ionospheric puncture point, the unit is half cycle ($\pi$), the geographic latitude $\phi_M$ of the puncture point M, and the geographical longitude $\lambda_M$. The calculation formula is:

$$\phi_M = \arcsin\left(\sin\phi_u \cdot \cos \varphi + \cos\phi_u \cdot \sin \varphi \cdot \cos A\right)$$

$$\lambda_M = \lambda_u + \arcsin\left(\frac{\sin \varphi \cdot \sin A}{\cos \phi_M}\right)$$

(4.4)

$\phi_u$: User geographic latitude

$\lambda_u$: User geographic longitude, the unit is radians;

$A$: Satellite azimuth in radians;

$\varphi$: The angle of the centroid of the user and the puncture point, in radians:

$$\varphi = \frac{\pi}{2} - E - \arcsin\left(\frac{R}{R+h} \cdot \cos E\right)$$

(4.5)

$R$ is the radius of the Earth, with a value of 6378 km; $E$ is the satellite elevation angle in radians; $h$ is the height of the ionosphere, which is 375 km.

By formula (4.6):

$$I_{B1I}(t) = \frac{1}{\sqrt{1-(\frac{R}{R+h} \cdot \cos E)^2}} \cdot I_z(t)$$

(4.6)

Convert $I_z(t)$ to the ionospheric delay $I_{B1I}(t)$ on the B1I signal propagation path in seconds.

For B2I signals, the ionospheric delay $I_{B2I}(t)$ on the propagation path is multiplied by $I_{B1I}(t)$ by a frequency-dependent factor $k(f)$, which is:

$$k(f) = \frac{f_1^2}{f_2^2} = \left(\frac{1561.098}{1207.140}\right)^2$$

(4.7)

$f_1$: The nominal carrier frequency of the B1I signal;

$f_2$: The nominal carrier frequency of the B2I signal in MHz.

Thus, the distance corrected is:

$$\Delta_{ion} = C \times I_{B1I}(t)$$

(4.8)

It can be obtained that the distance correction of $f_1$ is about 2 meters and the distance correction of
$f_2$ is about 8 meters.

Figure 9. Distance correction $\Delta \rho_{1on}$ is about 2 m and $\Delta \rho_{2on}$ is about 8 m

However, if the user's position is out of range, the distance correction will be wrong.

5. Conclusion

This paper analyzes the coordinate reference system used and the conversion between different reference systems, the calculation model of the Beidou satellite instantaneous position and the Chebyshev polynomial fitting algorithm. The satellite constellation simulation and visible satellite constellation simulation of Beidou were designed and completed by Matlab. The accuracy of the Beidou satellite orbit modeling is proved by combining Beidou satellite position calculation and constellation simulation. The simulation of the observation station in the static state further illustrates the rationality of the observation error model. After that, this paper discusses the simulation of spatial linear motion trajectory and uniform circular motion trajectory simulation. Finally, the correctness of the Klobuchar ionospheric delay correction model is verified by simulation.

The Beidou satellite constellation and its positioning error simulation prove that the system can meet the established target and run stably. However, for the Beidou satellite navigation integrated simulation test platform, there are still many places in the system that need further improvement. For the Beidou satellite navigation system, the Beidou navigation message is also an extremely important attribute. Later work can carry out an in-depth study on the generation principle of Beidou navigation message, and realize the simulation of the Beidou navigation message.

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