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An Orbital Design Method for Satellite Formation Flying

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An orbital design method of the formation initialization based on the relative orbital element method is presented. It firstly constructed the relative motion equation of the satellite formation flying in terms of the leader and followers' orbital elements. Then the equation was simplified when the orbit eccentricity of the leader satellite was small. And according to the satellites' mission, a general design method for the relative trajectory was proposed. The advantage of this method is that one can get a very simple analytical formula of each follower satellite's orbital elements when the orbital eccentricity of the leader satellite is zero. The simulation results show that the method is effective.

Key Words: Satellite Formation Flying, Relative Orbital Element Method, Relative Motion Equation, Formation Initialization

1. Introduction

Recently, there are two development trends for satellite technology. One is the innovation and development of large satellite platform technology, which produces satellites of increasing mass and more complete function (Chae and Park, 2003). These more capable satellites are able to complete complex missions independently. The other trend is the miniaturization and even micromation of satellites. Mini-satellites and micro-satellites have been widely used in a lot of fields and their importance has already been admitted. Their advantages are that they can be grouped together and effectively cooperate with each other. For example, the Satellite Constellation (SC) and the Satellite Formation Flying (SFF) can complete difficult mission that only a large satellite could complete in the past but an improved reliability and lifetime (Kang et al., 2000).

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One of the research focuses of the SFF technology in recent years is relative motion dynamics and control. Generally, the technology involves several satellites who fly in particular formation and cooperate to perform a certain task. Since the satellites in formation distribute regularly in space, they have enormous aperture and measurement base-line, which enable three-dimensional observation of an object, or simultaneous observation of multiple points in space. So it has enormous practical value, including ground observation, three-dimensional imaging, precise orientation, atmosphere survey, astronomical observation and so on (Irvin, 2001).

There are three key techniques during the task cycle of SFF, i.e. formation initialization, station keeping and formation reconfiguration (Atkins and Pennecot, 2002). The formation initialization (Vadali and Vaddi, 2000) includes a design process of the formation orbit. Until now, there are two approaches to calculate the initial target formation orbits of SFF, namely C-W equation descriptive method and relative orbital element method. W.H. Clohessy and R.S. Wiltshire derived C-W equation when working on the spacecraft rendezvous problem in the 1960's. The C-W equation descriptive method can effectively solve the problem of short-term (within several circles) and short-distance SFF. There are many papers on SFF using C-W equation as dynamics model of relative motion. With the research more thoroughly, more and more scholars have realized that C-W equation is not suitable to describe long-term SFF because the error of the relative motion solution increases by time (Gim and Alfriend, 2001 ; Gao et al., 2003). Alfriend and his colleagues (Alfriend et al., 2000) obtained the first-ordered approximate relative motion equation in terms of the differences of leader and followers' corresponding orbital elements by linearizing the transfer matrix between the leader/follower satellites' orbital coordinate systems and the inertial coordinate systems. The J_2 effect on satellite formation was studied in their papers (Schaub and Alfriend, 1999). But the determination of the initial orbits of the member satellites had not been discussed from the viewpoint of design. Gao Yunfeng and his colleagues studied the mechanism of C-W equation not suitable for long-term SFF and pointed out that the non-periodic solution of the relative motion due to the linearization of C-W equation caused the unacceptable error of long-term SFF. In his paper, a new method named relative orbital element method (ROEM) is introduced to overcome the limitation of C-W equation. The ROEM derives the precise equation of the relative motion by formulating the transfer matrix between the leader/follower satellite's orbital coordinate systems and the inertial coordinate systems in terms of leader and followers' orbital elements. In practice of formation flying, the differences between the leader and the follower satellites' corresponding orbital element should be small enough to ensure that the leader and follower are close. As a result of $\Delta\Omega$ and Δi being small, the ROEM derives the simplified equation of the relative motion. On the basis of the ROEM, this paper presents an orbital design method that can provide a very simple analytical formula of each follower satellite's orbital elements when the orbital eccentricity of the leader satellite is zero. The result can not only be used for feasibility analysis of the formation flying, but also serve as the initial value of the nonlinear equation of the relative motion solving.

2. Relative Motion Equation of the Satellites

Assuming that there is no orbital perturbation, let us consider the relative motion of SFF based on relative orbital elements for the two-body problem. Take two satellites as example, one called "leader satellite" or simply "leader" (the subscript is denoted by "L"), the other called "follower satellite" or "follower" (the subscript is denoted by "F").

2.1 Definition of the coordinate system

1) Geocentric-equatorial inertial coordinate system $OXYZ$: The origin is at the earth's center, the fundamental plane is the equator and X-axis points in the vernal equinox direction ; Z-axis is perpendiculars to the fundamental plane and points in the direction of the North Pole. Y-axis completes the right-handed frame.

2) The Geocentric-orbital coordinate system $Oxyz$: The origin is at the earth's center. The fundamental plane is the orbit plane with the x-axis pointing in the direction of the satellite's center of mass. The z-axis is perpendiculars to the fundamental plane and points in the direction of the positive normal. y-axis completes the right-handed frame.

3) The satellite orbital coordinate system $Sxyz$: The origin is at the satellite's center of mass. The axial orientations are the same as the geocentric-orbital coordinate system $Oxyz$.

2.2 Description of the satellite motion

There are several orbital elements used to describe the satellite motion : right-ascension of the ascending node Ω, inclination i, argument of perigee ω, orbital semi-major axis a, orbital eccentricity e, mean anomaly M (or true anomaly f), and denoting $u=\omega+f$ the argument of latitude (Bate et al., 1971).

2.3 The relative motion equation

Assuming that position vector of the leader and

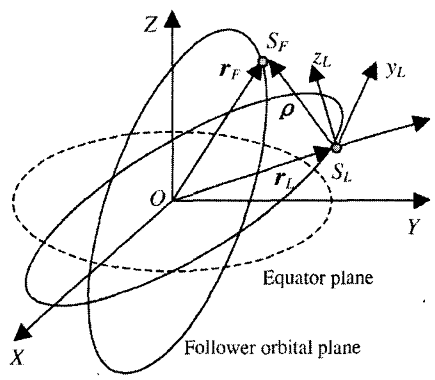

Fig. 1 SFF orbit

the follower are r_L and r_F respectively, as shown in Fig. 1, the relative motion of the follower to the leader is

$$\boldsymbol{\rho}=\boldsymbol{r}_F-\boldsymbol{r}_L \tag{1}$$

Eq. (1) described the vector form of the relative motion. To study the relative motion between the follower and the leader, one can project it into the leader's orbital coordinate system $S_Lx_Ly_Lz_L$, thus

$$\begin{Bmatrix} x \\ y \\ z \end{Bmatrix} = A_{12}\begin{Bmatrix} r_F \\ 0 \\ 0 \end{Bmatrix} - \begin{Bmatrix} r_L \\ 0 \\ 0 \end{Bmatrix}$$

where

$\{x\ y\ z\}^T$ is projection of the relative motion vector $\boldsymbol{\rho}$ into $S_Lx_Ly_Lz_L$,

$\{r_L\ 0\ 0\}^T$ is projection of the vector $\boldsymbol{r}_L$ into $S_Lx_L y_Lz_L$,

$\{r_F\ 0\ 0\}^T$ is projection of the vector $\boldsymbol{r}_F$ into $S_Fx_F y_Fz_F$,

A_{12} is transform matrix from $S_Fx_Fy_Fz_F$ to $S_Lx_Ly_L z_L$.

Then the component form of the relative motion for SFF (Gao, et al., 2003) can be presented as

$$\begin{Bmatrix} x \\ y \\ z \end{Bmatrix} = \frac{a_F(1-e_F^2)}{1+e_F\cos f_F}\begin{Bmatrix} k_1 \\ k_2 \\ k_3 \end{Bmatrix} - \frac{a_L(1-e_L^2)}{1+e_L\cos f_L}\begin{Bmatrix} 1 \\ 0 \\ 0 \end{Bmatrix} \tag{2}$$

Where

$$\begin{aligned}k_1=&\cos\Delta\Omega\cos u_L\cos u_F-\sin\Delta\Omega\sin i_F\cos u_L\sin u_F\\&+\sin\Delta\Omega\cos i_L\sin u_L\cos u_F\\&+\cos\Delta\Omega\cos i_L\cos i_F\sin u_L\sin u_F\\&+\sin i_L\sin i_F\sin u_L\sin u_F\end{aligned}$$

$$\begin{aligned}k_2=&-\cos\Delta\Omega\sin u_L\cos u_F+\sin\Delta\Omega\cos i_F\sin u_L\sin u_F\\&+\sin\Delta\Omega\cos i_L\cos u_L\cos u_F\\&+\cos\Delta\Omega\cos i_L\cos i_F\cos u_L\sin u_F\\&+\sin i_L\sin i_F\cos u_L\sin u_F\end{aligned}$$

$$\begin{aligned}k_3=&-\sin\Delta\Omega\sin i_L\cos u_F-\cos\Delta\Omega\sin i_L\cos i_F\sin u_F\\&+\cos i_L\sin i_F\sin u_F\end{aligned}$$

$$\Delta\Omega=\Omega_F-\Omega_L$$

In Eq. (2), the orbital elements Ω, i, e and a are constants, and u and f are periodic functions of time. Being the synthesis of periodic movements, the relative motion should not diverge even if not periodic.

3. Simplification of the Relative Motion Equation

In SFF, the satellites should stay within the proximity of each other for the task cycle, so the orbital planes of the leader and follower satellites and their positions in the orbital planes i.e. $\Delta\Omega=\Omega_F-\Omega_L$, $\Delta i=i_F-i_L$, and $\Delta u=u_F-u_L$ are small. The precondition for prolonged formation flying is that the orbit periods of the leader and follower should be the same (Baoyin et al., 2002).

Generally, the distance variation between satellites and ground is small for ground reconnaissance satellite system. Assuming e_L and e_F are small, thus

$$f_L\approx M_L+2e_L\sin M_L,\ f_F\approx M_F+2e_F\sin M_F$$

$$\frac{a_L(1-e_L^2)}{1+e_L\cos f_L}=a_L(1-e_L\cos f_L)$$

$$\frac{a_F(1-e_F^2)}{1+e_F\cos f_F}=a_F(1-e_F\cos f_F)$$

$$\begin{Bmatrix} x \\ y \\ z \end{Bmatrix} = \begin{Bmatrix} e_L\cos M_L-e_F\cos M_F \\ \Delta\omega+\Delta M+\Delta\Omega\cos i_L+2e_F\sin M_F-2e_L\sin M_L \\ -\Delta\Omega\sin i_L\cos(\omega_F+M_F)+\Delta i\sin(\omega_F+M_F) \end{Bmatrix} a \tag{3}$$

Where $\Delta\omega=\omega_F-\omega_L$, $\Delta M=M_F-M_L$.

Solution to the relative motion problem of multiple follower satellites can be derived from Eq. (3). The equation of the k-th ($k=1, 2, 3, \cdots$) follower relative to the leader is

$$\begin{cases} x_k=-A_k\cos(nt+\alpha_k) \\ y_k=B_{k0}+2A_k\sin(nt+\alpha_k) \\ z_k=-C_k\cos(\omega_k+M_k+\beta_k) \end{cases} \tag{4}$$

with

$$A_k=a\sqrt{e_L^2+e_k^2-2a_Le_k\cos(\Delta M_k)} \quad (4a)$$

$$B_{k0}=a(\Delta\omega_k+\Delta M_k+\Delta\Omega_k\cos i_L) \quad (4b)$$

$$C_k=a\sqrt{(\Delta\Omega_k\sin i_L)^2+(\Delta i_k)^2} \quad (4c)$$

$$\tan\alpha_k=\frac{e_k\sin M_k(0)-e_L\sin M_L(0)}{e_k\cos M_k(0)-e_L\cos M_L(0)} \quad (4d)$$

where $M(0)$ is the mean anomaly when $t=0$

$$\beta_k=\frac{\Delta i_k}{\Delta\Omega_k\sin i_L} \quad (4e)$$

n is called the "mean motion of the leader orbit."

4. Method of the Relative Trajectory Design

Designing relative trajectory (RT) is to confirm orbital elements of each member satellite based on the geometric pattern of the formation. It is only related to the initial state of RT (Gao and Li, 2003). It can be found from Eq. (4) that the projection of the relative motion onto the $x_L y_L$ plane is elliptical. The ratio of the semi-major axis to semi-minor axis is 2 : 1. The angle between the orbital plane formed by the follower satellites and the horizontal plane is 26.565°.

The geometric relation in the y_Lz_L plane (horizontal plane) is the key for ground reconnaissance satellite system. When $t=0$, if $\alpha_k=\omega_k+M_k+\beta_k$, there is

$$\left(\frac{y_k-B_{k0}}{2A_k}\right)^2+\left(\frac{z_k}{-C_k}\right)^2=1 \quad (5)$$

Where B_{k0} is the distance between the center of the RT and the leader's projection onto the y_Lz_L plane. Parameter A_k and C_k can be determined based on RT.

According to the characteristic of relative motion trajectory on the y_L direction and z_L direction, there are

$$y_{\max}=B_{k0}+2A_k =B_{k0}+2a\sqrt{e_L^2+e_k^2-2e_Le_k\cos(\Delta M_k)} \quad (6)$$

$$z_{\max}=C_k=a\sqrt{(\Delta\Omega_k\sin i_L)^2+(\Delta i_k)^2} \quad (7)$$

And each follower satellites positions on the y_L direction and z_L direction at the initial moment are

$$y_k(0)=\beta_{k0}+2A_k\sin\alpha_k \quad (8)$$

$$z_k(0)=-C_k\cos(\omega_k+M_k(0)+\beta_k) \quad (9)$$

The orbital elements of the follower satellites can be obtained from Eq. (4b) and Eqs. (6) ~ (9). But the solution to the non-linear equation group is very sensitive to the choice of the iterative initial value. Thus the method of RT design can be further simplified based on certain conditions.

If $e_L=0$ for the leader, the simple analytic solutions can be obtained from Eq. (4b) and Eqs. (6) ~ (9). The process is as follows

From Eq. (6), there is

$$e_k\frac{y_{\max}-B_{k0}}{2a} \quad (10)$$

From Eq. (4d), there is

$$M_k(0)=\alpha_k \quad (11)$$

where α_k can be obtained from Eq. (8) as

$$\alpha_k=m\pi+(-1)^m\,\mathrm{arsin}\frac{y_k(0)-B_{k0}}{2A_k}$$

Because m is arbitrary integer, so $M_k(0)$ has multiple groups of numerical solutions.

From Eqs. (4b), (7), (9), assuming $B_{k0}/a+\omega_L+M_L(0)=X$ and considering Δi_k and $\Delta\Omega_k$ to be small, thus

$$\sec^2 X\Delta i_k^2-\frac{2z_k(0)}{a\cos X}\tan X\,\Delta i_k+\left(\frac{z_k(0)}{a\cos X}\right)^2-\left(\frac{z_{\max}}{a}\right)^2=0$$

The above equation is a quadratic equation with respect to X, and because

$$\left(-\frac{2z_k(0)}{a\cos X}\tan X\right)^2-4\sec^2 X\left(\left(\frac{z_k(0)}{a\cos X}\right)^2-\left(\frac{z_{\max}}{a}\right)^2\right)=\frac{4}{a^2\cos^2 X}\left(z_{\max}^2-z_k(0)^2\right)\geq 0$$

so this equation has two unequal real roots

$$\Delta i_k=\frac{z_k(0)}{a}\sin X\pm\frac{1}{a|\cos X|\sec^2 X}\sqrt{z_{\max}^2-z_k(0)^2} \quad (12)$$

where $\pm$ means that the relative trajectory is symmetrical.

Furthermore

$$\Delta\Omega_k = \frac{\Delta i_k \sin X - \frac{z_k(0)}{a}}{\cos X \sin i_L} \tag{13}$$

$$\Delta(\omega_k + M_k(0)) = \frac{B_{k0}}{a} - \Delta\Omega_k \cos i_L \tag{14}$$

Because $M_k(0)$ has multiple groups of numerical solutions, accordingly ω_k has multiple solutions. But because the value of $\Delta(\omega_k + M_k)$ is always small and unique, we usually choose $\omega_k + M_k$ as the orbital element for small eccentricity orbit.

If $u_L=0$, $B_{k0}=0$, and $\Delta(\omega_k+M_k)$ is small, Eqs. (12) ~ (14) can be simplified as

$$\Delta i_k = \pm\sqrt{\left(\frac{z_{\max}}{a}\right)^2 - \left(\frac{z_k(0)}{a}\right)^2} \tag{15}$$

$$\Delta\Omega_k = \frac{z_k(0)}{-a \sin i_L} \tag{16}$$

$$\omega_k = \frac{z_k(0)}{a} \cot i_L - M_k(0) \tag{17}$$

5. Example Analysis

Consider a formation satellite system made up of four mini-satellites distributed as a diamond in space. The satellite system flies in low earth orbit at an altitude of 680 kilometers in dense formation (whose interval is about 10 kilometers). It is used to monitor the radio and radar activity for the whole earth surface. Now one assumes that each satellite should accept the management of the ground station, which is located at latitude 43°36′ north and longitude 1°26′ east. We suppose that the system will be launched in October 2004.

5.1 Determination for the leader orbital elements of satellite system

When designing the formation orbit based on the relative motion equation, it is necessary to first determine the orbital elements of the leader. Since the two diagonals of a diamond are perpendicular and bisecting, we can choose the intersection as the leader satellite. Four follower satellites remain in a diamond form and surround the leader. Then the orbital elements can be determined according to the satellite system mission requirement.

Semi-major axis: the semi-major axis $a=R+h$, where R is the earth average radius 6371.1 km, h is the orbital altitude 680 km, we can get orbital semi-major axis to be 7051.1 km.

Orbital eccentricity: Observing the earth surface and monitoring the ground signal by satellite, it is hoped that satellite altitude from the ground does not change much. Therefore we choose a circular orbit, i.e. the orbital eccentricity of the leader satellite is 0.

Orbital inclination: Because the satellite system should observe the whole earth surface, we consider the satellites system as running on a sun-synchronous orbit. According to relative equation of the sun-synchronous orbit

$$-9.97\left(\frac{R_e}{a}\right)^{3.5} \cos i = 0.9856 \tag{18}$$

where R_e is the earth equatorial radius 6378.14 km, thus the orbital inclination can be obtained as 98.07°.

Right-ascension of the ascending node: We select the northern hemisphere as the key area of monitoring, and the satellite descending through the equator in the morning. We consider a ground station located at latitude 43°36′ north and longitude 1°26′ east, and a satellite descending through the equator at 10 a.m. Therefore, if satellite chooses repeating orbit, every time the satellite flies over the ground station at 9:48 a.m. From October 1st to the October 31st 2004, at latitude 75° north, the solar vertical angle monotonically decreases from 10.08° to −0.90°, as shown in Fig. 2. At latitude 75° south, the solar vertical angle monotonically increases from 12.34° to 23.02°, as shown in Fig. 3. So we choose to launch on October 1st, 2004, so that the satellite system can reconnoiter by visible light between latitude 75° north and south.

So the right-ascension of the ascending node Ω is

$$\Omega = \alpha_{\otimes} + \gamma = 2\pi t/365 + t_0 \times \pi/12 \qquad t \in [0, 365],\ t_0 \in [0, 24] \tag{19}$$

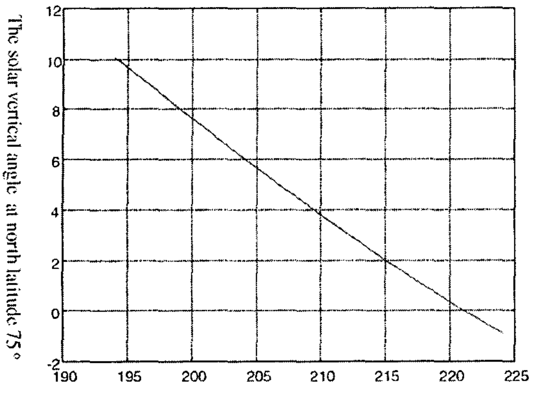

Fig. 2 The change of the solar vertical angle during October in 2004 at north latitude 75°

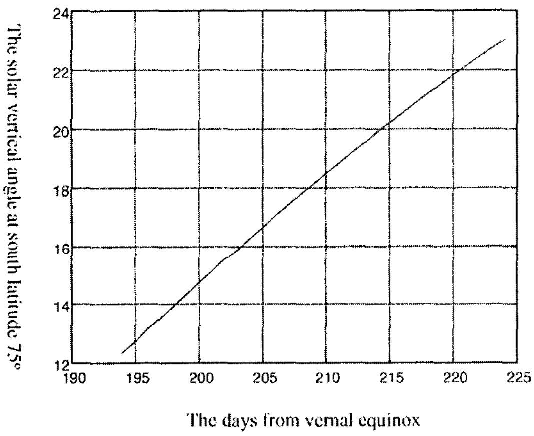

Fig. 3 The change of the solar vertical angle during October in 2004 at south latitude 75°

where $\alpha_{\otimes}$ is the mean solar right-ascension. γ is the difference of the right-ascension between the ascending node and the mean solar. t is the days calculated from vernal equinox, so that October 1^{st} is the 194-th day. t_0 is the local time of descending node. Then we can obtain Ω as 206.34°.

Argument of perigee and mean anomaly: To a circular orbit, the true anomaly and the mean anomaly are equal. Both the argument of perigee and mean anomaly are nonsensical. For the convenience of calculation, choose $\omega_L=0$ and $M_L(0)$, so that the orbital elements of the leader are: (see Table 1)

Table 1 The orbital elements of the leader

a_L(km)	e_L	Ω_L	i_L	ω_L	$M_L(0)$
7051.1	0	206.34°	98.0730°	0	0

5.2 Design for the relative trajectory of the satellite system

The relative trajectory of the satellite formation must be subjected to orbital dynamics. There are two impossible formations: two satellites "fly" side by side forever or a satellite "flies" above or below another with the same speed (Yeh and Sparks, 2000). The satellite system requires the four satellites to keep the diamond formation flying forever, so that it can observe and monitor an identical signal from different points. It works as an enormous antenna in the space and improves the accuracy of getting information. For this kind of ground reconnaissance satellite system, the projection in the $y_L z_L$ plane (horizontal plane) should be a diamond. Therefore, the only possible trajectory is the four followers circulating around the leader, i.e. $B_{k0}=0$. The leader is imaginary, and S_1 and S_3, S_2 and S_4 occupy different circular orbits that share the same center, as shown in Fig. 4.

Choose the radii of the projection circles in $y_L z_L$ plane as five and ten kilometers respectively. From Eqs. (10), (11) and (15) ~ (17), the orbital elements of the four followers at the initial moment can be obtained. (See Table 2)

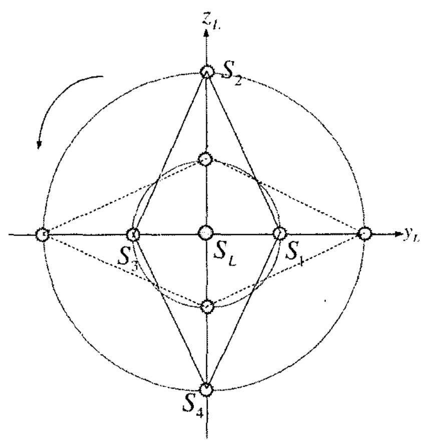

Fig. 4 The relative trajectory projected onto the horizontal plane

Table 2 The orbital elements of the four followers at initial moment

	a_k (km)	e_k	Ω_k	i_k	ω_k	M_k
k=1	7051.1	0.000354554	3.601312379	1.712405866	−1.570796327	1.570796327
k=2	7051.1	0.0007091092	3.599879965	1.711696757	−3.141793814	3.141592654
k=3	7051.1	0.000354554	3.601312379	1.710987648	1.570796327	−1.570796327
k=4	7051.1	0.0007091092	3.602744792	1.711696757	0.000201161	0

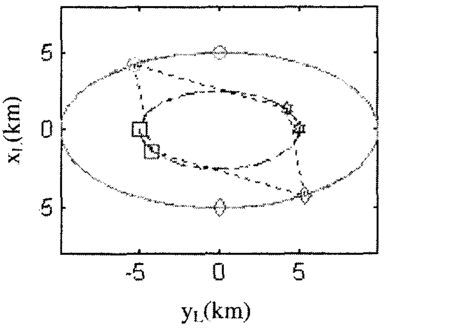

Fig. 5 The relative trajectory projected onto the y_Lx_L plane

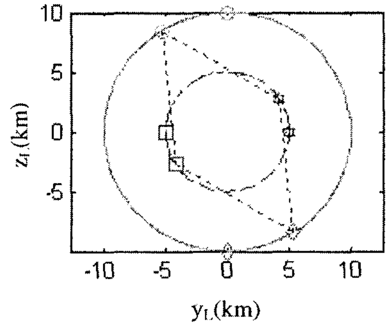

Fig. 6 The relative trajectory projected onto the y_Lz_L plane

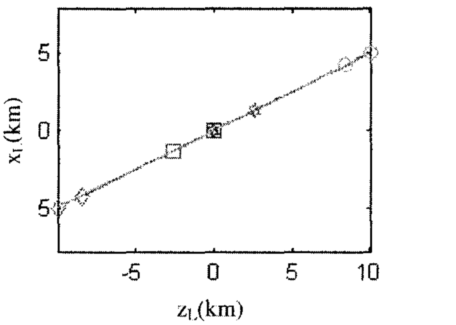

Fig. 7 The relative trajectory projected onto the z_Lx_L plane

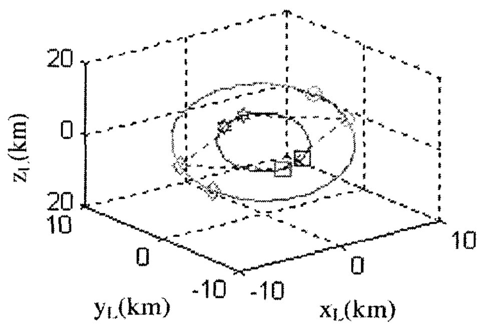

Fig. 8 The relative trajectory in space

5.3 Simulations

Substituting the follower's parameter into the relative motion Eq. (2), one can get the relative trajectory of the satellite system in the leader's orbital coordinate system $Sxyz$, (as shown in Figs. 5~8). The figures show that the four satellites keep forever the diamond formation in the y_Lz_L plane.

6. Conclusions

This paper has constructed the relative motion equation for the SFF based on relative orbital element method. Combining with practice, the model has been simplified. An orbit has been designed by the mission requirements of the satellite system. Numerical simulation results have shown that the method is effective.

This paper has used the general form to construct the relative motion model. It is feasible to design the relative trajectory when the orbital eccentricity of the leader satellite is zero, e.g. the ground reconnaissance satellite system.

Acknowledgments

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