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Estimation about Active Phase of Trajectory Based on the Satellite Passive Detection

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Abstract: To estimate a spacecraft trajectory, the observation satellite position should be calculated in advance. Through the simplified equation of motion based on the observation satellite for finishing the second order differential equation. By the MATLAB simulation interpolation operation with the existing list data, solution to satellite of three dimensional position in different moments. By the point wise intersection positioning method to estimating the trajectory of the spacecraft, the position and velocity of the spacecraft in a certain time will be obtained. And using simulated annealing for the mathematical model of a spacecraft trajectory and analyzing the rationality of the estimate. *Copyright © 2014 IFSA Publishing, S. L.*

Keywords: MATLAB interpolation, Point wise intersection positioning method, Trajectory reckon, Simulated annealing.

1. Introduction

Some countries will launch special purpose spacecrafts, such as ballistic missiles, reconnaissance satellites. Implementation of monitoring and quick response to the hostile spacecraft on the launch of other countries has important strategic significance for the maintenance of national security. Find emission and detecting its orbital parameters is to realize monitoring and respond to the first step, without which further judgment and reaction will be impossible. Being at the top of the earth, the satellite is an important platform of today's probe spacecraft launch and orbital parameters [1]. In recent study, there are lots of methods to estimate the spacecraft trajectory, such as statistical determination of orbits, base on the determination of federated filtering orbit, determination of orbits by USB and VLBI and so on [2, 3].

2. The Analysis of Satellite Trajectory and the Establishment of Coordinate

2.1. The Analysis of Spacecraft Trajectory

The spacecraft trajectory can generally be divided into three sections: the active phase, oblique segment and attack segment. Fig. 1 is the active phase diagram (not to scale). The ground launch point is located at point A, AB for vertical ascent stage, BC arcs for the program bend segment, CD arcs for gravity oblique stage, DE arcs for elliptical orbit [1].

2.2. Coordinate System

The first coordinate system is a translational basic coordinate system, which takes the center of the Earth O_c as the origin, takes axis z from the Earth rotation axis, takes axis x from O_c pointing to zero moment on 0 longitude line, and determine axis y by right-handed, it is the establishment of a rectangular coordinate system $O_c - X_cY_cZ_c$.

The second coordinate system is the observation coordinate $O_s - X_sY_sZ_s$ with the satellite motion, who's origin is taken from the satellite center O_s, the axis X_s is along the connection O_cO_s, leaving the Earth is the direction, the shaft Z_s and X_s vertical point north, the axis Y_s is determined by the right-handed [1]. The above-mentioned two coordinates see Fig. 2 below.

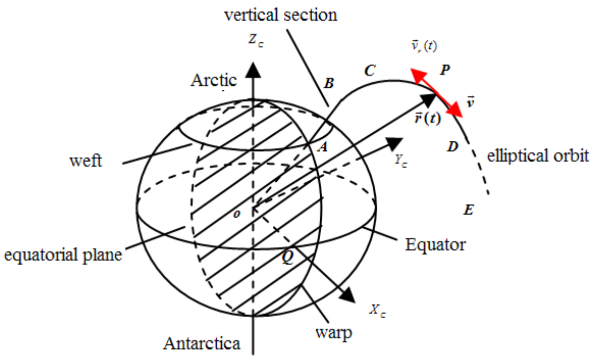

Fig. 1. Schematic diagram of the active phase of spacecraft trajectory.

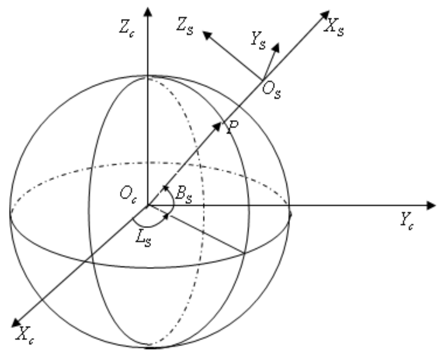

Fig. 2. Observation coordinate system diagram.

3. Modeling and Solving for the Power Segment of Observation Satellites Trajectory to be Estimated

Position calculation of observation satellites is the premise to estimate the spacecraft trajectory.

3.1. Establishment and Solving of the Position Model of Observation Satellites at any Time

3.1.1. The Establishment of the Model

In a short period of time, it is assumed that the Rectangular coordinate system $O_c - X_cY_cZ_c$ does not rotate with the Earth as inertial coordinate system. Simplified equation of motion for observation satellites [1]:

$$\ddot{\vec{r}}(t) = \vec{F}_e = \frac{G_m}{|\vec{r}(t)|^3}\vec{r}(t), \quad (1)$$

$$\vec{r}(t) = [x(t) \quad y(t) \quad z(t)]^T, \quad (2)$$

$$\dot{\vec{r}}(t) = [\dot{x}(t) \quad \dot{y}(t) \quad \dot{z}(t)]^T, \quad (3)$$

Substituting Formula (2) and (3) in (1), we obtain the differential equation as the following identity holds:

$$d[\dot{x}(t) \quad \dot{y}(t) \quad \dot{z}(t)]^T = -\frac{G_m}{\|\vec{r}(t)\|^3}[x(t) \quad y(t) \quad z(t)]^T, \quad (4)$$

3.1.2. Solution of the Model

Extract references [1] satinfo.txt files, 00, 06,09 satellites pose parameters are as follows (Table 1):

Table 1. 00, 06, 09 satellites pose value.

Posture \ Number	00	06	09
x	0.000000	-1732113.220573	-6126905.483436
y	-6493774.465428	9092044.771852	3993014.549776
z	-6493774.465428	1732113.220573	6126905.483436
$\dot{x}$	6653.695256	-4453.807606	-1930.523921
$\dot{y}$	0.000000	-1566.513180	-5793.950363
$\dot{z}$	0.000000	4453.807606	1930.523921

Observation satellite data in Table 1 using MATLAB variable step Runge-Kutta method [4]. The three-dimensional position of the satellite at different times can be obtained by solving equation (4). Let us take 09 Satellite as an example: in 50.0 s, 100.0 s, 150.0 s, 200.0 s, 250.0 s five time three-dimensional position. See Table 2 and Fig. 3.

Table 2. Values of the five positions for satellite No. 09.

Time(s) \ Position (106 m)	X	Y	Z
50	1.773806	8.161384	4.516700
100	1.501626	8.126764	4.684680
150	1.227700	8.082699	4.847216
200	0.952349	8.029253	5.004127
250	0.675894	7.966502	5.155238

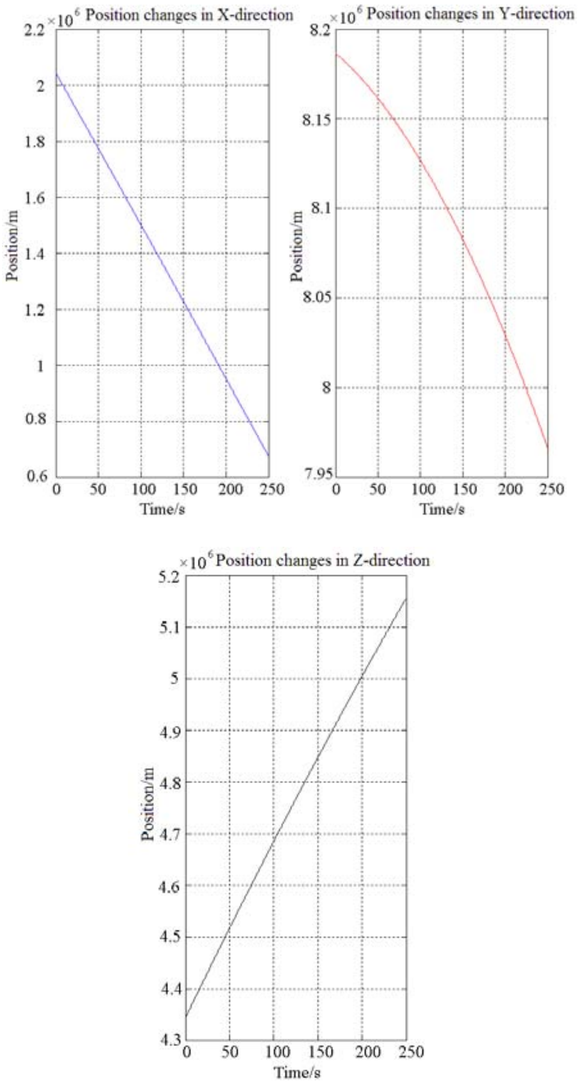

Fig. 3. Five position curves for satellite No. 09.

4. Spacecraft Trajectory Estimation Modeling and Solving

4.1. Calculate the Actual Position of the Spacecraft Relative to the Cartesian Coordinate System $O_c - X_cY_cZ_c$

4.1.1. Determination of the Direction Vector $\vec{\mathbf{k}}$ of the Aircraft in the Coordinate System of Observation Satellite.

Observation data of spacecraft observations to Observation Satellites can be determined through simplification two dimensionless ratios in observation coordinates:

$$\alpha = \frac{y_s}{x_s}; \quad \beta = \frac{z_s}{x_s}, \tag{5}$$

where x_s, y_s, z_s are coordinates of the spacecraft in the observation coordinate system. Combination of simulation data files meadata_i_j.txt [1] shows that the direction vector $\vec{\mathbf{k}}=(1,\alpha,\beta)$ of the aircraft in the coordinate system of the satellite.

4.1.2. Find the Rotation Matrix by Observation Coordinate System $O_s - X_sY_sZ_s$ Relative to the Cartesian Coordinate System $O_c - X_cY_cZ_c$ [5]

The angle between the projection of vector $\overrightarrow{\mathbf{o_c o_s}}$ within plane $x_c o_c y_c$ and the shaft x_c and shaft x_s are L_s and B_s. Through analysis, the shaft x_s, the shaft z_c, the shaft z_s are coplanar. The posture of the observation coordinate system $O_s - X_sY_sZ_s$ may be obtained by rotating a Cartesian coordinate system $O_c - X_cY_cZ_c$, the rotation matrix is:

$$ {}^{\mathbf{o_c}}_{\mathbf{o_s}}R = R(z_{\mathbf{c}}, L_{\mathrm{s}})R(\mathrm{y}^1_{\mathbf{s}}, B_{\mathrm{s}}), \tag{6}$$

$$ {}^{o_c}_{o_s}R=\begin{bmatrix} \frac{x_c}{\sqrt{x_c^2+y_c^2+z_c^2}} & \frac{-y_c}{\sqrt{x_c^2+y_c^2}} & \frac{-x_cz_c}{\sqrt{(x_c^2+y_c^2)(x_c^2+y_c^2+z_c^2)}} \\ \frac{y_c}{\sqrt{x_c^2+y_c^2+z_c^2}} & \frac{x_c}{\sqrt{x_c^2+y_c^2}} & \frac{-y_cz_c}{\sqrt{(x_c^2+y_c^2)(x_c^2+y_c^2+z_c^2)}} \\ \frac{z_c}{\sqrt{x_c^2+y_c^2+z_c^2}} & 0 & \frac{\sqrt{x_c^2+y_c^2}}{\sqrt{x_c^2+y_c^2+z_c^2}} \end{bmatrix}$$

4.1.3. Direction Vector $\vec{k}$ of the Satellite in the Coordinate System of Observation Satellites

Direction vector $\vec{k}$ of the satellite in the coordinate system of observation satellites can be showed in the Cartesian coordinate system $O_c - X_cY_cZ_c$, said:

$$\vec{\mathbf{l}} = (m, n, p) = {}^{o_c}_{o_s}R\vec{k}^T \tag{7}$$

4.1.4. To give the Equations of the Straight Lines Connect to Aircraft

Let *L1* and *L2* be the two straight lines which connect to aircraft respectively from the 6th and the 9th satellite. In the Cartesian coordinate system $O_c - X_cY_cZ_c$, the equations of *L1* and *L2* can be expressed as follows:

$$\begin{cases} l_1 : \dfrac{x-x_{c1}}{m_1} = \dfrac{y-y_{c1}}{n_1} = \dfrac{z\text{-}z_{c1}}{p_1} \\ l_2 : \dfrac{x-x_{c2}}{m_2} = \dfrac{y-y_{c1}}{n_2} = \dfrac{z\text{-}z_{c1}}{p_2} \end{cases}, \quad (8)$$

where (i=1,2) are the position coordinates of the two satellites, and (i=1,2) are the direction vectors of *L1* and *L2*.

4.1.5. To Solve the Midpoint Coordinate of Common Perpendicular Segment

Let *L3* be common perpendicular segment of *L1* and *L2*. Then a direction vector of *L3* can be expressed by

$$W = (m_1, n_1, p_1) \times (m_2, n_2, p_2) = \left(\begin{vmatrix} n_1 & p_1 \\ n_2 & p_2 \end{vmatrix}, \begin{vmatrix} p_1 & m_1 \\ p_2 & m_2 \end{vmatrix}, \begin{vmatrix} m_1 & n_1 \\ m_2 & n_2 \end{vmatrix} \right) \quad (9)$$

Then the equation of the plane π_1 determined by L1 and L3 can be given as follow

$$N_1(x - x_{c1}) + R_1(y - y_{c1}) + Q_1(z - z_{c1}) = 0, \quad (10)$$

where (N_1, R_1, Q_1) is vector product of the direction vectors of *L1* and *L2*.

Let C be the point of intersection of the straight line *L2* and the plane π_1, then C is an endpoint of common perpendicular segment, and its coordinate can be given by

$$(m_2 t + x_{c2}, n_2 t + y_{c2}, p_2 t + z_{c2}), \quad (11)$$

where

$$t = \frac{N_1(x_{c1} - x_{c2}) + R_1(y_{c1} - y_{c2}) + Q_1(z_{c1} - z_{c2})}{m_2 N_1 + n_2 R_1 + p_2 Q_1}$$

Similarly, another coordinate of endpoint of common perpendicular segment can be expressed by

$$(m_1 s + x_{c1}, n_1 s + y_{c1}, p_1 s + z_{c1}), \quad (12)$$

where

$$s = \frac{N_2(x_{c2} - x_{c1}) + R_2(y_{c2} - y_{c1}) + Q_2(z_{c2} - z_{c1})}{m_1 N_2 + n_1 R_2 + p_1 Q_2},$$

(N_2, R_2, Q_2) is vector product of the direction vectors of *L2* and *L3*.

Using (11), (12) and formula of midpoint coordinates, we can get midpoint coordinate of common perpendicular segment CD easily. The coordinates of the $\overrightarrow{CD}$ center:

$$\begin{pmatrix} x & y & z \end{pmatrix} = \begin{pmatrix} \dfrac{x_1+x_2}{2} & \dfrac{y_1+y_2}{2} & \dfrac{z_1+z_2}{2} \end{pmatrix} \quad (13)$$

4.2. Solution of the Speed and Position of the Sample Data

The spacecraft tracking measurement data sequence is $\{y(t_i), t_i = t_0 + ih, i = 1, 2, 3, \cdots\}$ assumed where h is the sampling interval. For measurement data sequence $\{y_i, i = 1, 2, 3, \cdots\}$, the true value of the measured object is $\left\{\hat{y}_i, i = 1, 2, 3, \cdots\right\}$, measurement data and true value subject to the following model:

$$y_i = \hat{y}_i + \varepsilon_i \quad (i = 1, 2, 3, \cdots, n).$$

P-order differential measurement to data sequence

$$\Delta^p y_i = x_{i+p} - C_p^1 x_{i+p-1} + C_p^2 x_{i+p-2} + \cdots + (-1)^v C_p^v y_{i+p-v} + \cdots + (-1)^p x_i = \sum_{j=0}^{p} (-1)^j C_p^j y_{i-jl+pl}$$

Take the mathematical expectation to $C_p^j = \dfrac{p!}{v!(p-v)!}$ we can get: $E[\sum_{i=1}^{N-p} (\Delta^p y_i)^2] = (N-p) C_{2p}^p \sigma^2$. Known statistic is $S_p^2 = \dfrac{\sum_{i=0}^{N-p} (\Delta^p y_i)^2}{(N-p) C_{2p}^p}$ which is the unbiased estimator for σ^2, and note $\hat{\sigma}_p^2 = S_p^2$.

Smooth assumption of random error, the establishment of the estimates of the variance of the random error roots:

$$S_{Lp} = \left[\frac{(p!)^2}{(N-pl)(2p)!} \sum_{i=(k-1)N+1}^{kN-pl} (\Delta^p y_i)^2 \right]^{1/2},$$

where N is the number of data, P is the order of the polynomial, l for the step length [6].

35 gradient calculated the aircraft observations trajectory data based on a point-by-point intersection positioning method [7], using MATLAB programming observational data column is divided into a linear combination of zero mean random error, random error root square statistic: SLP = 1.588471750317061 × 1030.

Found that random errors a great impact so their measurement results should be polynomial fitting in the 0 number spacecraft position before seeking to eliminate the impact of random error model [8]. Fitting the data see reference [1] the meadata_i_j.txt files as follows:

T	No. 06 satellite	
	α	β
50.1754448	0.07497794549	0.06482026689
51.1754448	0.07585474572	0.06558806708
52.1754448	0.07672282878	0.06635250780
53.1754448	0.07758017828	0.06711446064
54.1754448	0.07844558434	0.06787123160
55.1754448	0.07930404513	0.06861333503
56.1754448	0.08015617465	0.06936130117
57.1754448	0.08100664145	0.07010785332
58.1754448	0.08185815426	0.07083236257
59.1754448	0.08270038898	0.07157563304
60.1754448	0.08354715060	0.07229466494
61.1754448	0.08438509898	0.07301968020
62.1754448	0.08521193562	0.07373495355
63.1754448	0.08604820283	0.07444731581
64.1754448	0.08687804762	0.07515805930
65.1754448	0.08769966121	0.07585776605
66.1754448	0.08851820402	0.07655323902
67.1754448	0.08933341865	0.07724900956
68.1754448	0.09014441362	0.07793941078

T	No. 09 satellite	
	α	β
50.1812189	-0.62660920295	0.462858412
51.1812189	-0.62614155788	0.463655906
52.1812189	-0.62568239172	0.464464432
53.1812189	-0.62522653205	0.465269558
54.1812189	-0.62477153493	0.466065624
55.1812189	-0.62433376202	0.466864309
56.1812189	-0.62388123607	0.467652105
57.1812189	-0.62344228853	0.468445212
58.1812189	-0.62301013023	0.469238632
59.1812189	-0.62258081598	0.470025125
60.1812189	-0.62216080791	0.470814603
61.1812189	-0.62173950285	0.471592301
62.1812189	-0.62132782588	0.472374964
63.1812189	-0.62091718246	0.473160570
64.1812189	-0.62050858984	0.473931949
65.1812189	-0.62011462832	0.474709560
66.1812189	-0.61972647038	0.475481201
67.1812189	-0.61932991804	0.476253386
68.1812189	-0.61894508795	0.477019812

Note: Due to the limited length of the thesis, we only list part of data. More data can be seen in the references [1] metadata_i_j.txt.

Location coordinates using MATLAB to the meeting point of the point-by-point and said in Fig. 4 - Fig. 9 and Table 3.

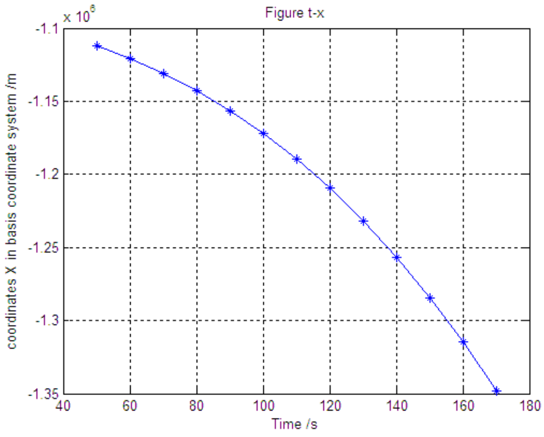

Fig. 4. t-x fitting curve.

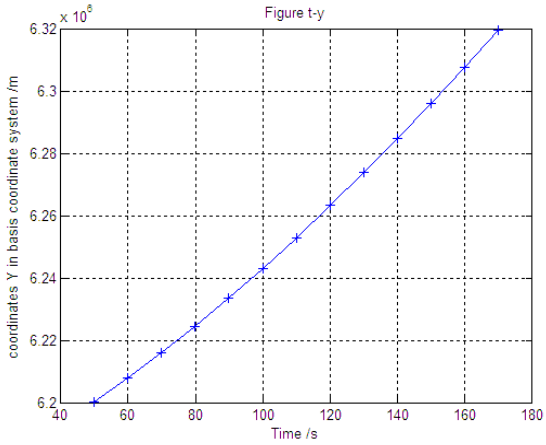

Fig. 5. t-y fitting graph.

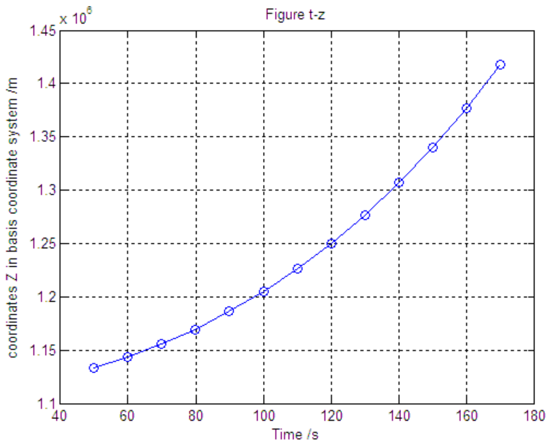

Fig. 6. t-z fitting graph speed fitting.

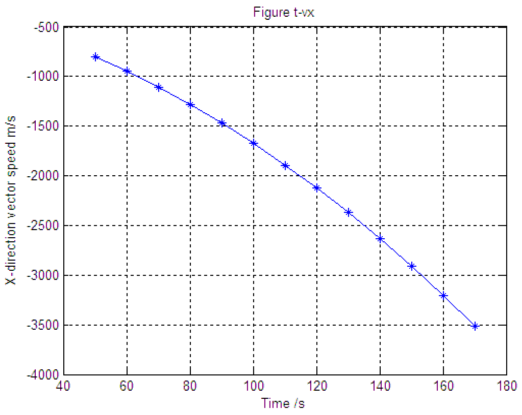

Fig. 7. t-vx fitting graph.

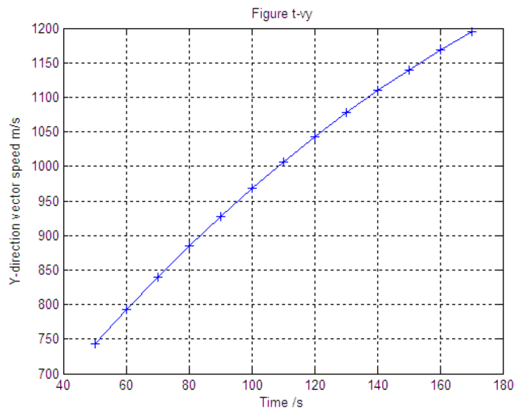

Fig. 8. t-vy fitting curve.

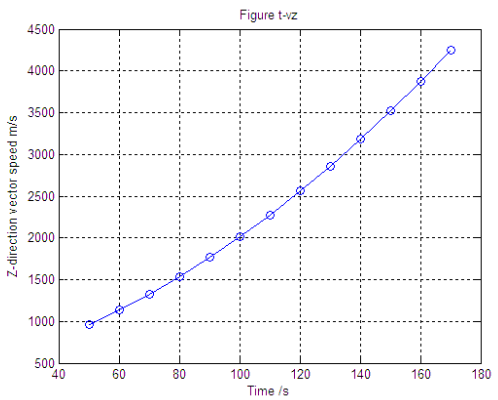

Fig. 9. t-vz fitting.

Table 3. Space vehicle 0 in the position of the respective sampling points.

T	*X*	*y*	*z*
50	-1112040.17722374	6200221.89641194	1133051.55573184
60	-1120549.71280121	6207842.02827011	1143248.21658795
70	-1130710.30165180	6215944.66818326	1155380.91428514
80	-1142596.50042729	6224556.74368745	1169582.47689512
90	-1156310.89206745	6233606.28657414	1186013.75719826
100	-1171937.12382432	6243096.50890875	1204841.67923599
110	-1189679.74891736	6252954.03101137	1226134.86733462
120	-1209631.28576888	6263187.06364858	1250139.53096298

Table 4. No. 0 spacecraft in the speed of the respective sampling points.

T	*Vx*	*vy*	*vz*
50	-806.433795199264	743.849441840313	960.258836861234
60	-950.287843786879	792.789936975576	1134.01532309060
70	-1108.97735900292	839.657492000610	1326.02463475964
80	-1282.50234084739	884.452106914483	1536.28677186859
90	-1470.86278932053	927.173781715334	1764.80173441721
100	-1674.05870442232	967.822516405955	2011.56952240574
110	-1892.09008615231	1006.39831098448	2276.59013583418
120	-2124.95693451120	1042.90116545092	2559.86357470229

4.3. To Establish the Accuracy of the Orbit Mathematical Model is Validated by Simulated Annealing Solution [9]

According to variable mass particle dynamics, a simplified equation of motion of the spacecraft in the active section of the base coordinate system [1] as follows:

$$\ddot{\vec{r}}(t)=\vec{F}_e+\vec{F}_T=-\frac{G_m}{|\vec{r}(t)|^3}\vec{r}(t)+\vec{v}_r(t)\frac{\dot{m}(t)}{m(t)}, \quad (14)$$

where the vector $\vec{F}_e$ representation of the external force acceleration and $\vec{F}_T$ representation of the thrust acceleration by rocket, $m(t)$ is the instantaneous quality, $\dot{m}(t)$ is the quality rate of change, $\vec{r}(t)$ is the spacecraft position vector of the base coordinate system, $\ddot{\vec{r}}(t)$ is the second derivative to $\vec{r}(t)$ of the time t, call acceleration. G_m is the earth's gravitational constant (the gravitational constant is $G_m = 3.986005 * 10^{14} m^3/s^2$ in this problem). In order to get direction of the thrust acceleration clearly. $\vec{v}_r(t)$ is fuel jet velocity relative to the rocket tail vents general direction and its direction is close to the reverse of the line to the aircraft speed direction, and its size is generally more stable [3].

Type sizes and typefaces: follow the type sizes specified in Table 1. As an aid in gauging type size, 1 point is about 0.35 mm.

Assumptions: 1. Air resistance of the aircraft is minimal in gravity Oblique stage and negligible atmospheric drag, small perturbations suffered while ignoring its effects. 2. Objects fall off the aircraft in this phase of flight in the instantaneous mass monotonically decreasing non- negative function, represented by the mathematical model:

$$m(t) = M - \dot{m}(t)t, \quad (15)$$

where M represents the initial mass of the aircraft.3. The size of the fuel injection is stable and constant relation equation, represented by the mathematical model:

$$\vec{v}_r(t) = -v_r \frac{\vec{v}_r(t)}{\left|\vec{v}_r(t)\right|}, \quad (16)$$

where v_r is the ejection velocity of the fuel relative to the rocket tail orifice; $\frac{\vec{v}_r(t)}{\left|\vec{v}_r(t)\right|}$ is the aircraft speed direction.

The coefficients are constants in equation (15) and (16), and substituting into Equation (14) can be drawn in the mathematical equation of the spacecraft, and the spacecraft's flight path can be drawn by the solution of ordinary differential equations. Thereby by known space coordinate of the aircraft, we can get inverse solution of the coefficients of the equation (15) and (16) to simulation.

Of the spacecraft mathematical equations to fit the optimization objective observations and fitting the residual sum of squares mean optimal use of simulated annealing, in order to determine a good mathematical expression of $m(t), \vec{v}_r(t)$.

Simulated annealing algorithm [10] is described as following:

1) Solution space.

S describes as all the cycle collection is $\{50, 60, \cdots, 120\}$, include all starting point and ending point. According to the datum, usually the quality of the aircraft can be defer as 4500 kg around, assume that loss quality value of setting jet flame is 0.7 kg/s and the speed of this flame is 1500 m/s [11].

2) Objective function.

Objective function is square mean of the residual between observed value and fitted value. We use mathematical equation fitting for trajectory.

3) New solution.

Assume the previous solutions are m_i, dm_i, Vr_i, solve the change value by choosing a random direction.

4) Value range.

$$\begin{cases} m0 > 1200 \\ 5000 > Vr > 1500 \\ 1 > dm > 0.01 \end{cases}$$

5) Acceptance criterion.

If $\Delta f < 0$ and new solution meet objective acquire to accept new path, otherwise accept new path using probability of $\exp(-\Delta f / T)$.

6) Drop temperature.

Decreasing temperature by selecting drop temperature coefficient $\alpha = 0.8$, replace by new temperature.

7) Termination condition.

Select termination temperature $e = 10^{-5}$ to determine whether the annealing process is ended. If T<e, the end of the algorithm, the output of the current data.

It shows that

m_0 =1.203868423329133×103,
d_m=0.689934373898759,
V_r =2.789987571940146×103,

where m_0 is the initial mass of the spacecraft, d_m is the injection amount of the jet flame, V_r is the flame jet velocity.

The difference between the observed and predicted values (fitted values) can be obtained, that is, the actual observed values and regression estimates squares mean difference: SSE=2.260605653126576 × 104; compared with the impact of the variance of the random error, we can know the actual situation of this kind of model composite and calculation result has more accuracy.

5. Conclusion

Thesis based on analysis of trajectory and the establishment of an appropriate coordinate system, which cite a large number of existing information and data from satellite observations to estimate the satellite orbit. Estimating the active segment of trajectory, we have improved the accuracy of the point wise intersection positioning method of double satellite observation than single-satellite observations. Because of some deviation due to the bifacial space intersected, taken two straight, the midpoint of the common perpendicular to further improve the accuracy of the track as the location of the spacecraft. To get trajectory mathematical model with simulated annealing model and verify that a reasonable estimate.

Upon this study, it can be expanded to multi-satellite observations of the same aircraft to estimate its orbit, thereby further consideration how satellites observing multiple spacecrafts at the same time, and observation error estimates in different circumstances, which will bring about practical application produce an important significance.

The method and model used in the paper are applied to scientific research, industrial and agricultural production, military reconnaissance, railway line selection, coastal and ocean mapping, mapping, target positioning, but also can be used for space target collision warning and circumvent maneuver, of great significance to safeguard national security and promote the growth of economy.

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