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RESEARCH ON DYNAMIC MODEL OF SPACE
INTERCEPTOR IN END-GUIDANCE PHASE

by

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RESEARCH ON DYNAMIC MODEL OF SPACE INTERCEPTOR IN END-GUIDANCE PHASE

Zheng Huimin

ABSTRACT. This paper is intended to explain the realization of direct-collision kill mode by space interceptor against a target. In accordance with the requirements of the interceptor guidance and control system program, an orbital control engine is proposed to make a modulation over interceptor centroid movement in the end-guidance phase. Also, an attitude-control engine is used to make an attitude adjustment. Furthermore, a six-degree-of-freedom interceptor trajectory simulation model is constructed. The computational results of the model indicate that it is necessary and possible to achieve control over the interceptor by using attitude-control and orbital-control engines in the end-guidance phase.

Key Words: space interceptor, end guidance, engine control.

1. Introduction

The interceptor as a principal component of future anti-space-based weapon interception systems is used to cope with space-based weapon systems, including all kinds of space weapon platforms, relay satellites for communication and command, as well as valuable military-supplied satellites in different orbits. Technically speaking, an interceptor can strike the target through a direct and high-speed collision. To bring such a kill mode into effect, an accurate adjustment of interceptor

attitude is required in the end-guidance phase to continually lock on the target in the field of view. At the same time, adjustments of its centroid location are also needed so as to ensure an ultimate direct collision with the target. Centering on the foregoing problems, this paper discusses some relevant research and presents final conclusions.

2. Introduction to Interceptor Control System

The interceptor guidance and control system is integrated with an accelerometer complex, a guidance head, an in-missile computer, an attitude-control engine, and an orbital-control engine, as shown in Fig. 1.

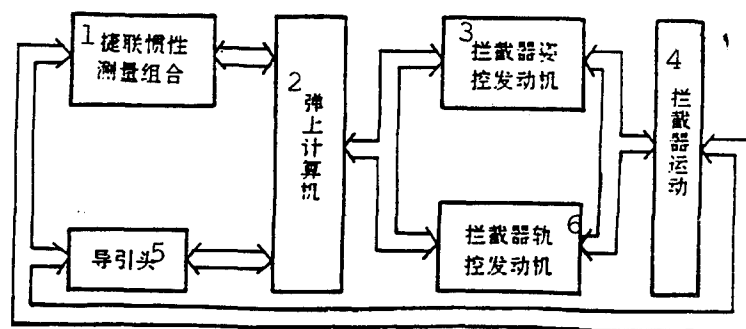


Fig. 1. Composition of control system
 KEY: 1 - accelerometer complex 2 - in-missile computer 3 - attitude-control engine of interceptor
 4 - interceptor movement 5 - guidance head 6 - orbital-control engine of interceptor

The operating process of the control system is as follows:

The initial guidance phase is defined as the period from interceptor launch to the jettisoning of the third-stage engine. The mid-guidance phase starts when the third-stage booster cuts off and the interceptor separates from the booster. The in-missile computer then, based on instantaneous interceptor kinetic parameters, predicts the target miss rate and computes the

attitude control space and working time of the orbital-control engine required to eliminate the target miss rate, and then takes control over attitude modulation as well as switching the orbital-control engine on and off in an effort to achieve the desired interceptor velocity increment and eliminate the target miss rate, and eventually to ensure a minimum target miss rate remaining to the end-guidance phase during the shift from mid-guidance to end-guidance phase.

When the interceptor reaches a distance of 300km from the target satellite and its fairings have been jettisoned, the interceptor attitude has to be adjusted, based on its location and on the extrapolated target location, to point the optical axis of the guidance head at the target. When the guidance head has locked on to the target, the interceptor attitude again has to be adjusted to make ready for the end-guidance phase.

During the end-guidance phase, the interceptor attitude-control system maintains a stable attitude, and the in-missile computer, based on the target view and other data provided by the guidance head, controls the switching on and off of the orbital-control engine in accordance with the chosen guidance law and directs the interceptor to proceed along a straight collision course with the target.

As required by the proportional guidance law, the direction of view must be controlled and stabilized so as to keep the vision slewing rate within a desired range and to keep the guidance head instantaneously directed at the related parameters following the measurement view. The guidance head is physically connected to the interceptor, with its measurement axis matched with the interceptor longitudinal axis as to direction. The field of view of the guidance head is 1° . Thus, during the end-guidance phase, the first requirement is to keep the angle between the direction of view and the interceptor longitudinal

axis less than 0.5° , that is, $q < 0.5^\circ$ during the course of orbital control.

3. Ballistic Mathematical Model of Interceptor

1) Coordinate systems and their transformation

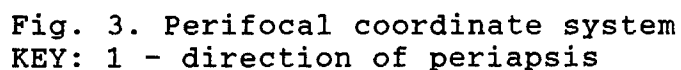
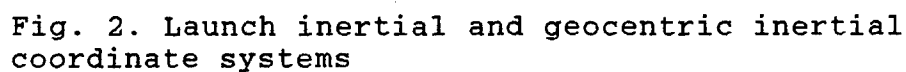
Several coordinate systems used during ballistic computations are expressed as follows:

(1) Launch inertial coordinate system $Ox_pY_pZ_p$. Its point of origin is set at the launch point. The Ox_p axis is located in the horizontal plane passing through the launch point and is directed in the launch direction. The OY_p axis is vertically upward, and the OZ_p axis together with Ox_p and OY_p constitutes a right-handed coordinate system.

(2) Geocentric inertial coordinate system $OeX_4Y_4Z_4$. Its point of origin is located at the center of the earth. OeX_4 lies within the equatorial plane pointing to the vernal equinox. OeZ_4 lies perpendicular to the equatorial plane pointing toward the North Pole, and the OeY_4 axis, together with the OeX_4 and the OeZ_4 axes, forms a right-handed coordinate system, as described in Fig. 2.

(3) Perifocal coordinate system $OeX_wY_wZ_w$. Its datum surface is the orbital plane of the satellite and its coordinate axes are X_w , Y_w , and Z_w , with X_w pointing to the periapsis. In the datum plane, when X_w is rotated by 90° along the direction of movement, it becomes Y_w . As for Z_w , it is defined in accordance with the right-handed method. This coordinate system is used to describe satellite motion, as Fig. 3 shows.

(4) Missile body coordinate system $O1X1Y1Z1$. Its point of origin $O1$ is located at the interceptor centroid, with its $O1X1$ moving forward along the missile longitudinal axis; the $O1Y1$ axis lies



in the interceptor plane of symmetry (positive upward), and the O1Z1 axis, along with the O1X1 and O1Y1 axes, constitutes a right-handed coordinate system.

(5) Relative Velocity Coordinate System $O_{X_v Y_v Z_v}$. Its point of origin is designed to be at the target center, with its X_v axis parallel to $\vec{V}_r$, the velocity vector of the missile-target

relative motion, with the $\vec{V_r}$ direction taken as positive; the Y_{vr} axis is selected in the vertical plane upward, as well as the Z_{vr} axis, along with the X_{vr} and Y_{vr} axes, constitutes a right-handed coordinate system. Moreover, in this coordinate system, the missile target miss rate $\dot{p}$, target miss azimuth θ , and the $Ox_{vr}Y_{vr}Z_{vr}$ plane is called the target miss plane.

These coordinate systems are mutually transformable, that is, the interchangeability between the body and the launch inertial coordinate systems can be achieved through the three Euler angles, including the pitch angle, the drift angle, and the roll angle. Likewise, a transformation between the geocentric inertial and the launch inertial coordinate systems can be accomplished through longitude λ , latitude B , and launch azimuth A , and a transformation between geocentric inertial and perifocal coordinate systems--also through the three Euler angles, i.e., the ascending node of the equatorial right ascension Ω , the argument of the perigee ω , and orbital inclination i . Additionally, transformation between launch inertia and relative velocity coordinate systems can be realized through the relative velocity inclination θ and the relative velocity edge angle ϕ_r .

2) Interceptor kinetic differential equation

The equation set forth here is used to describe interceptor motion after the third-stage booster is jettisoned at a height of approximately 60km. To simplify this model, the aerodynamic effects can be ignored, and also the coriolis effect. Therefore, the kinetic equation of the interceptor in the launch inertial coordinate system is as follows (earth is considered as a circular body with radius R_e):

$$(1) \dot{t} = 1$$

$$(2) \dot{X}_d = V_{xd}$$

$$(3) \dot{Y}_d = V_{yd}$$

$$(4) \dot{Z}_d = V_{zd}$$

$$(5) \dot{V}_{xd} = P_{xd}/m - \mu X_d/R_d^3$$

$$(6) \dot{V}_{yd} = P_{yd}/m - \mu(Y_d + R_e)/R_d^3$$

$$(7) \dot{V}_{zd} = P_{zd}/m - \mu Z_d/R_d^3$$

$$(8) \dot{m} = -P_0/I_s$$

$$(9) \dot{\omega}_{x1} = M_{x1}/J_{x1}$$

$$(10) \dot{\omega}_{y1} = (M_{y1} + (J_{z1} - J_{x1})\omega_{x1}\omega_{z1})/J_{y1}$$

$$(11) \dot{\omega}_{z1} = (M_{z1} + (J_{y1} - J_{x1})\omega_{x1}\omega_{y1})/J_{z1}$$

$$(12) \dot{\psi} = \omega_{y1}\sin\gamma + \omega_{z1}\cos\gamma$$

$$(13) \dot{\phi} = (\omega_{y1}\cos\gamma - \omega_{z1}\sin\gamma)/\cos\psi$$

$$(14) \dot{\gamma} = \omega_{x1} - \tan\psi(\omega_{y1}\cos\gamma - \omega_{z1}\sin\gamma)$$

where the principal symbols are as follows:

- t - interceptor flight time
- X_d, Y_d, Z_d - interceptor centroid coordinates under the centroid inertial coordinate system
- V_{xd}, V_{yd}, V_{zd} - three velocity components of the interceptor under the inertial coordinate system
- P_{xd}, P_{yd}, P_{zd} - three components of interceptor engine thrust under the inertial coordinate system
- μ - gravitational constant
- m - interceptor mass
- P_0 - engine specific impulse
- $\omega_{x1}, \omega_{y1}, \omega_{z1}$ - angular velocity of interceptor rotating about three axes of the missile body coordinate system
- M_{x1}, M_{y1}, M_{z1} - control movement of interceptor about three axes of the missile body coordinate system
- J_{x1}, J_{y1}, J_{z1} - rotational momenta of three interceptor axes
- ψ, ϕ, γ - pitch angle, drift angle, and roll angle of interceptor.

3) Mathematical model of target

If the satellite is the main target of the interceptor, then the six main orbital elements are required to determine the

satellite location along its orbit at a particular instant. These are elliptic long-axis a , eccentricity ratio e , orbital inclination i , longitude of ascending node Ω , perifocal angular distance ω , and over-perifocal time T .

The Kepler equation is: $M(t) = n(t - \tau)$

where M is a rectilinear anomaly, n is the average angular orbital velocity, $n = \sqrt{\mu/a^3}$, t is time; if the rectilinear anomaly is M_0 at point 0, then $M(t) = M_0 + n(t - t_0)$, then the target kinetic parameters under the perifocal coordinate system are as follows:

$$\begin{aligned} p &= a(1 - e^2) & r_p &= r \cos f \\ E &= M + e \sin E & r_a &= r \sin f \\ \operatorname{tg} f / 2 &= \sqrt{(1 + e)/(1 - e)} \operatorname{tg}(E/2) & V_p &= -\sqrt{\mu/p} \sin f \\ r &= p/(1 + e \cos f) & V_a &= \sqrt{\mu/p} (\cos f + e) \end{aligned}$$

4) Relative motion of interceptor against target

(1) Relative distance, velocity, and field of view rotation rate. The interceptor coordinates and velocity under the launch inertial coordinate system are: X, Y, Z, V_x, V_y, V_z , while the target satellite coordinates and velocity are: $X_m, Y_m, Z_m, V_{xm}, V_{ym}, V_{zm}$. In addition, the relative distance R and relative velocity V_r of the interceptor against the target are:

$$\begin{aligned} X_r &= X_m - X & Y_r &= Y_m - Y & Z_r &= Z_m - Z \\ R &= \sqrt{X_r^2 + Y_r^2 + Z_r^2} \\ V_{xr} &= V_{xm} - V_x & V_{yr} &= V_{ym} - V_y & V_{zr} &= V_{zm} - V_z \\ V_r &= \sqrt{V_{xr}^2 + V_{yr}^2 + V_{zr}^2} & \dot{R} &= (X_r V_{xr} + Y_r V_{yr} + Z_r V_{zr}) / R \end{aligned}$$

The relative velocity inclination θ_r and the edge angle ϕ_r are:

$$\theta_r = \arcsin(V_y/V_r) \quad \phi_r = \arctg(-V_zr/V_xr)$$

The angle of view and the angular velocity of the interceptor-to-target connecting line in the pitch plane and drift plane are as follows, respectively:

$$\begin{aligned} q_p &= \arcsin(Y_r/R) & \dot{q}_p &= (RV_{yr} - Y_r\dot{R})/R/(\sqrt{X_r^2 + Z_r^2}) \\ q_y &= \arctg(-Z_r/X_r) & \dot{q}_y &= (Z_rV_{xr} - X_rV_{zr})/(X_r^2 + Z_r^2) \\ q &= \sqrt{\dot{q}_p^2 + \dot{q}_y^2} \end{aligned}$$

(2) Target miss rate calculations:

Let the three components X_r , Y_r , Z_r of the interceptor-target relative distance under the launch inertial coordinate system be transformed under the relative velocity coordinate system, that is,

$$(X_{vr}, Y_{vr}, Z_{vr})^T = C(X_r, Y_r, Z_r)^T.$$

then the target miss rate and the target miss azimuth are:

$$\rho = \sqrt{Y_{vr}^2 + Z_{vr}^2} \quad \phi_{vr} = \arctg(Z_{vr}/Y_{vr})$$

5) Several major problems

The switching-on and switching-off control rules of the orbital-control engine and the attitude-control engine are a major problem in computing the interceptor-control trajectory, for which an analysis is given as follows:

Switching-on and off conditions of orbital-control engine and proportional guidance law. There are four orbital-control engines, mounted at equal intervals on the interceptor perimeter passing through the interceptor centroid. The jet pipe axial line crosses the interceptor centroid in order to accomplish control over the interceptor centroid movement. Installation of the orbital-control engines can be done in two ways, as described in Fig. 4a and 4b. Technically, the orbital-control engine has a small pulse feature:

During the interception of targets, the orbital-control engine starts to operate when the target and the interceptor exhibit a relative rotation rate of the field of view $\dot{q}$, that is, the condition of starting up the orbital-control engine is that the relative field of view rotation rate $\dot{q}$ not equal to 0. Following the operation of the orbital-control engine, $\dot{q}$ approaches 0 to eliminate the field of view rotation rate. When $\dot{q}$ approaches 0, the orbital-control engine is required to perform with the vernier ring to maintain $\dot{q}=0$. Still, in the initial stage of the end-guidance phase, the orbital-control engine operates with a normal and stable thrust due to the relatively high initial $\dot{q}_0$. After $\dot{q}=0$, the orbital-control engine is required to operate with small pulses and make an adjustment over the orbit in order that $\dot{q}$ is maintained until there is direct collision with the target.

The proportional guidance law is used to control the interceptor centroid movement in three-dimensional space, with an acceleration commanded as follows:

$$\vec{a}_r = \lambda |\dot{V}r| (\dot{q}_p \cdot \vec{i}_p + \dot{q}_y \cdot \vec{i}_y)$$

that is, the commanded acceleration rate all upward and the drift plane are as follows, respectively:

$$a_p = \lambda |\dot{V}r| \dot{q}_p \quad a_y = \lambda |\dot{V}r| \dot{q}_y$$

When the attitude-control engine has accomplished its attitude adjustment, the four orbital-control engines can perform switch control and large- and small-pulse control in accordance with the acceleration command.

(2) Phase plane control law of attitude-control engine. There are four attitude-control engines. Fig. 5 shows a schematic diagram of their structures. The jet (1+2) or (3+4) is needed in the control of all-upward movement, while jet (1+4) or (2+3) is needed to control the drift movement, and jet (1+3) or (2+4) is favored in roll control. Since these three channels are

interrelated, the attitude-control engine can perform a time-sharing control over the three channels by using logic judgment and logic control combined with all-upward and drift as priorities. The attitude-control engine equally exhibits a small-pulse capability.

When the attitude-control engine is operating, the interceptor rotates about its centroid. The differential equation of motion is as follows: $d\dot{E}/dt = M/I = a$

where E is the error angle; M is the control moment; $\dot{E}$ is the angular-velocity deviation; and I is the angular momentum.

When the action movement is positive, the differential equation is:

$$d\dot{E}/dt = a, \quad dE/dt \cdot d\dot{E}/dE = a, \quad \dot{E} d\dot{E}/ds = a$$

By integrating both sides, we get:

$$\int_{\dot{E}_0}^{\dot{E}} \dot{E} d\dot{E} = a \int_{E_0}^E dE$$

and the result is $E = E_0 + (\dot{E}^2 - \dot{E}_0^2) / 2 / a$

When the action movement is negative,

$$E = E_0 - (\dot{E}^2 - \dot{E}_0^2) / 2 / a$$

The phase switch curve can be used to control the switching on and off of the engine and to select a corresponding working state allowing the interceptor to move within the limiting ring. The horizontal axis input volume of the phase plane is the attitude singular deviation E -dot, while the vertical axis input volume is the attitude angular velocity E . The phase plane switch curve is shown in Fig. 6.

The curve is made up of four curves:

ray AC: $\dot{E} = DDB$

ray CB: $\dot{E} = -DDB$

ray AD: $E = DB = \dot{E}^2 | 2 | a$

ray CD: $E = -DB + \dot{E}^2 | 2 | a$.

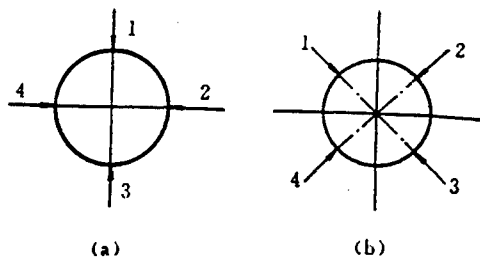


Fig. 4. Installation of orbital-control engines

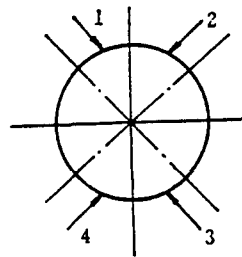


Fig. 5. Attitude Control engine arrangement

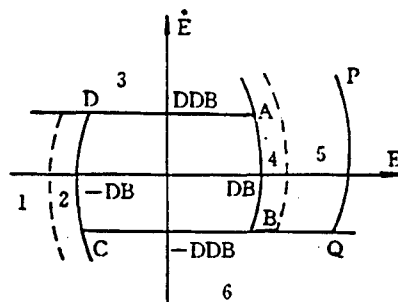


Fig. 6. Limiting ring and phase plane

The control moments in regions 1 and 2 of the plane are negative, positive in regions 4 and 5, the engine does not operate in regions 2 and 6, and there is small-pulse moment in region 2 and 4.

For example, at point P in the phase plane, the process of entering the limiting ring is as follows:

- PQ - positive control moment acting on the interceptor
- CDABC - limited circulation
- QBC - angular-velocity drift rotation at point Q

The phase plane error angle and the angular velocity are selected as follows:

pitch channel: $E = \psi - q_p$ $\dot{E} = \dot{\psi} - \dot{q}_p$
 drift channel: $E = \psi - q_p$ $\dot{E} = \dot{\psi}$
 roll channel: $E = \gamma$ $\dot{E} = \dot{\gamma}$

4. Computational Results and Analysis

A six-degree-of-freedom simulation model of an interceptor has been established as outlined above. The following calculations are based on data derived from a foreign interceptor:

1) initial data

(1) Overall interceptor parameters

Quality, distance between centroid and peak point, angular momentum about three axes

(2) Engine parameters of interceptor

Thrust range, pulsing time, specific impulse, fuel weight, etc., of orbital-control engine and attitude-control engine.

(3) Launch vehicle and interceptor separation instant parameters

Interceptor kinetic parameters at shutoff instant of launch vehicle third-stage engine (that is, the separation instant of the launch vehicle and interceptor) are initial binding parameters during simulation:

$X_0 = 22160\text{m}$	$Y_0 = 71810\text{m}$	$Z_0 = -11480\text{m}$
$V_{x0} = 1860\text{m/s}$	$V_{y0} = 4620\text{m/s}$	$V_{z0} = -70\text{m/s}$
$\omega_{x0} = 0$	$\omega_{y0} = 0$	$\omega_{z0} = 0$
$v_0 = 67.681^\circ$	$\psi_0 = -10.482^\circ$	$\gamma_0 = 0.376^\circ$

(4) Target initial parameters

$a = 7871139\text{m}$ $e = 0$ $i = 74^\circ$ $\Omega = 97.281^\circ$ $w = 0$

(5) Selection of launch site parameters

$A = 103^\circ$ $B = 186^\circ$ $\lambda = 25.5^\circ$

(6) Engine shutoff point parameters

$$\begin{array}{lll} \Delta X_0 = 4478\text{m} & \Delta Y_0 = 2000\text{m} & \Delta Z_0 = -4000\text{m} \\ \Delta \omega_{x0} = 0 & \Delta \omega_{y0} = 0.5^\circ/\text{s} & \Delta \omega_{z0} = 0.5^\circ/\text{s} \end{array}$$

2) Computational results and conclusions

Based on the initial parametric calculations, the parameters of the intersection point between interceptor and target are as obtained as follows:

$$\begin{array}{lll} x = 1077678\text{m} & y = 1425854\text{m} & z = -48917\text{m} \\ R = 1500132\text{m} & V = 1606\text{m/s} & V_m = 7116\text{m/s} \end{array}$$

The contact angle between interceptor and target is: $\psi_m = 24.6^\circ$

The target miss rate is:

$$\rho = 0.46\text{m}$$

After computation and analysis, several conclusions can be presented as follows:

(1) During computations, when the interceptor is 400km from the target, the attitude-control engine switches on to control the interceptor-attitude angle. The result shows that at 250km from the target, the angle between guidance head axial line and target field of view is 0.5° , which satisfies the requirement. If the orbital-control engine switches on at that instant, it can make an accurate adjustment over the centroid because the interceptor axial line (also the guidance head axial line) and target are basically on the same straight line at that instant. The final target miss rate is derived at 0.46m.

(2) Single-precision data variables were adopted in this computation. It is held that higher precision might be attained if double-precision data variables had been used.

(3) By using the time-sharing three-channel control, three channels can be controlled in consistency, but this results in frequently shifting control among individual channels, greatly

increasing the time period of jet pipe opening and therefore, reducing efficiency. It is then believed, after preliminary analysis, that it will be better to adopt an interrogatory time-sharing control and make a proper arrangement for three channels, respectively. However, if the pitch channel does not require control, then the drift channel and the roll channel can be interrogated and control handed over to either of them upon request. If both these two channels also do not need control, then the interceptor is not to be controlled at that time.

(5) In our computations, the maximum thrust of the interceptor orbital-control engine used was 50kg. But in the process, a more powerful engine was tried, such as a 200kg engine, which proved to be able to reduce the control time and showed some other advantages. Nevertheless, this requires further verification.

(6) Selection of the phase plane control rule curve directly affects the computational accuracy.

(7) This procedure is universal in character and therefore is above to handle various errors.

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