Integrated Guidance, Navigation and Control in Flying-by Small Bodies using Interceptor
Osamu Mori (ISAS/JAXA) and Kohta Tarao (Toyota Motor Corporation)

Abstract
Institute of Space and Astronautical Science (ISAS) of Japan Aerospace Exploration Agency (JAXA) is currently planning the missions that the small probe ‘interceptor’ flybys near-Earth objects. Interceptor is very small probe. An interceptor observes spectrum, takes close images, and determines mass of a NEO (near earth object) during a flyby. The weight of interceptor is less than 10 kg. This paper shows three types of missions. In general, it is impossible to determine the relative orbit during flyby only with optical information. Thus the optical navigation needs to be combined with the radio navigation that should provide the relative velocity vector information. In this paper, the integrated guidance and navigation strategy of interceptor is proposed.

1. Introduction
More than 300 thousands asteroids and comets are in the solar system. They are conceived to have preserved primordial state well. The origin and developing process of the solar system can be known by investigating the typical one of them intensively. Institute of Space and Astronautical Science (ISAS) of Japan Aerospace Exploration Agency (JAXA) was launched the spacecraft ‘HAYABUSA’. It arrived at asteroid ‘ITOKAWA’ in September, 2005, and it will collect samples and return them to earth. On the other hand, only taking pictures of many small bodies are also very significant. We are investigating flyby missions which aim at NEOs using very small probes ‘interceptor’[1].

An interceptor observes spectrum, takes close images, and determines mass of a NEO during a flyby. These measurements make clear the geographical and geological features of a target. The weight of interceptor is less than 10 kg. Three types of missions are proposed[2]. First type is one or two interceptors are launched as piggyback mission. The second type is several interceptors are launched and each one aims at each target. The third type is a few interceptors are mounted to a spacecraft for submissions. In any type of mission, the structure of each interceptor is the same. There are a lot of chances of carrying out missions. A list of the target candidates is shown.

In order to accomplish the high speed flyby, an active orbit control guiding a spacecraft passing through the specified point in the B-plane around an object is required[3]. At the same time, it requests an autonomous navigation capability using optical cameras. In this paper, an integrated guidance and navigation strategy during high speed flyby is described.

2. Mission and System of Interceptor
The observation of interceptor consists of three steps as shown in Fig. 1.
1) A few hours before the flyby
The interceptor takes images of the asteroid by the narrow view multi-band camera. These images give the information of the composition of the asteroid or the motion of the asteroid, such as the rotation axis or the rotation period. On the way of its approach to the asteroid, the interceptor is guided autonomously using its onboard optical navigation system to flyby through the asteroid with small distance.
2) Just before the flyby
The interceptor takes images by the wide view camera. These images give the terrestrial information of the asteroid surface: size of the crater, thickness of the regolith, distribution of the boulder and so on.
3) After the flyby

The deviation of the trajectory of the interceptor gives the information of the mass of the asteroid. Considering the volume and true density, the bulk density and the porosity can be calculated.

Only one flyby of one interceptor can perform many observations. It can be an interesting mission if more than one interceptor fly by the same asteroid. By adjusting the distance, direction and time of each flyby, the whole body of the target asteroid can be observed. On the other hand, by crashing one interceptor into the asteroid, the internal as well as the surface of the target can be measured. This can be applied to the space guard mission, in which the orbital element of an asteroid is changed in order to avoid the collection.

Interceptor is a very small probe whose weight is only 10 kg. Then the various flyby missions using interceptor can be proposed according to the carrier.
1) Type 1: (Solid rocket motor + interceptor) × 1 or 2

They are launched as piggyback payloads and separated on GTO (geostationary transfer orbit). The mass of a solid rocket motor is less than 15 kg. Then total mass is 25 or 50 kg.
Event sequence: launch -> spin up and solar cell deployment -> transfer the orbit from GTO by solid rocket motor -> radio navigation -> optical navigation -> flyby

2) Type 2: Solid rocket motor + interceptor carrier + interceptor × 4

This type is suitable for a small mission payload in ISAS. Four interceptors are connected to the interceptor carrier whose mass is assumed to be 40 kg. The carrier is separated on LEO (low earth orbit) or GTO, and the total mass is 400 or 200 kg, respectively. Each interceptor is separated just before earth swing-by and aims at each target. If four interceptors are independent except for their launch. Multiple interceptors must be operated independently even at the critical phase. The simultaneous critical operations can be avoided by the earth swing-by using interceptor carrier.
Event sequence: launch -> spin up of interceptor carrier -> transfer the orbit from GTO or LEO to earth synchronous orbit by solid rocket motor -> (maneuver and separation of interceptor) × 4 -> spin up and solar cell deployment of each interceptor -> earth swing-by

-> radio navigation -> optical navigation -> flyby

3) Type 3: Mother spacecraft and interceptor × n

It is a submission payload. The several interceptors are carried by the mother spacecraft which has a main mission. The flyby of NEO on the way to the target for next sample return mission is investigated. And flybys of a few Trojan asteroids using interceptors are proposed in solar sail mission. The interceptor is separated from mother spacecraft one day before flyby. The mother spacecraft receive the data from the interceptor. If the mother spacecraft itself flies by the asteroid, it needs much fuel and large thrust. Therefore the flyby using the interceptor has advantage to decrease the total mass and risk. In this type, the interceptors can communicate with a mother spacecraft (not with the earth) and the total electric power is small compared with the other types. Thus the solar cells are not required.
Event sequence: transfer the orbit of mother spacecraft -> radio navigation -> separation of interceptor -> spin up of each interceptor -> optical navigation -> flyby -> transfer the orbit of mother spacecraft for next flyby or other mission

(a) Earth swing-by
(b) Asteroid flyby

Fig. 2 Mission image (type 2)

Fig. 3 shows the instruments of interceptor and interceptor carrier.
a. Narrow view multi-band camera -> taking image and orbit determination
b. Wide view camera -> orbit determination
c. Sun sensors -> attitude determination
d. RF sensors -> attitude determination
e. Coherent transponder -> orbit determination
f. Non-directional antenna -> communication
g. RCS -> attitude and orbit control
h. DHU/DR -> data control
i. Solar cells and battery -> power supply

The size of each interceptor is \(40\text{cm} \times 40\text{cm}\).

Interceptor and interceptor carrier spin around the vertical axes with the rate of 20 rpm.

3. Guidance and Navigation

In this chapter, an integrated guidance and navigation strategy during high speed flyby is proposed based on the filter devised in NOZOMI mission[^4][^5].

The narrow view camera is used for navigation of the interceptor. Fig. 4 shows a simplified two-body encounter model.

\[
\tan \phi_1 = A(t_1 - T), \quad \tan \phi_2 = A(t_2 - T), \quad A = v / B
\]

In this simple situation, \(\alpha\) and \(T\) are obtained by two distinct measurements.

\[
A = \frac{\tan \phi_1 - \tan \phi_2}{t_1 - t_2}, \quad T = \frac{t_2 \tan \phi_1 - t_1 \tan \phi_2}{\tan \phi_1 - \tan \phi_2}
\]

In general, it is impossible to determine the relative orbit during flyby only with optical information due to the so called scaling effect in ballistic flight. The nearest distance \(B\) and relative velocity \(v\) can not be obtained only with optical navigation. Therefore the optical navigation needs to be combined with the radio navigation that should provide the relative velocity vector information. In the high speed flyby, camera elevation angle \(\phi\) varies rapidly. Fig. 5 shows its history at the encounter. \(\phi\) is nearly constant at the point far from the asteroid. So the information of camera shortly before the encounter is very important for this mission.

The coordinate system is illustrated in Fig. 6. \(z\) is parallel to \(v\). The relative velocity and position between the interceptor and the target are expressed as

\[
v = v_0 + \sum \Delta v
\]

\[
r = B(\cos \phi_n x + \sin \phi_n y) + v_0 z
\]

\(v_0\) is the velocity just before the optical navigation, and it is obtained by two-way Doppler. \(\Delta v\) can be calculated by the thrust history.

A nonlinear observer used in NOZOMI is similar to Kalman filter. The state update and propagation schemes are expressed by

\[
\hat{\mathbf{r}}_{k+1} = \hat{\mathbf{r}}_k + v_k \Delta t + K_k \left( \frac{\mathbf{r}_{k+1}}{\mathbf{r}_{k+1}} - \hat{\mathbf{r}}_k + v_k \Delta t \right)
\]

where

\[
K_k \equiv \left| \mathbf{r}_k + v_k \Delta t \right|
\]

The error covariance matrix \(P\) propagates with a measurement noise property characterized by \(R\) governed by

\[
P_{k+1} = \frac{\hat{\mathbf{r}}^T_{k+1}}{\mathbf{r}_{k+1}^2} P_k \left( \frac{\hat{\mathbf{r}}^T_{k+1}}{\mathbf{r}_{k+1}^2} \right) + \hat{\mathbf{r}}^2_{k+1} R_{k+1}
\]

When the interceptor is far from the target, the errors of \(x, y\)-axes are equal to zero by the first term of Eq. (7). This caused by the fact that the direction \(z\)-axis is nearly equal to sight direction. Thus the position in \(xy\)-plane can be estimated and the distance out of the plane cannot be estimated. On the other hand, when the interceptor is near to the target, the distance out of the
plane can be estimated and the position error in the plane is increased around the target. The position in the plane needs to be estimated by the other method around the target.

The camera elevation angle is expressed as

$$\phi = \frac{\pi}{2} - \cos^{-1} \left( \frac{r \cdot v}{r \cdot v} \right)$$

(8)

The relationship between \(A\) and \(\phi\) is

$$A = d \left( \tan \phi \right) / dt$$

(9)

It can be estimated by the following low-pass filter:

$$A_{k+1} = \alpha A_k + (1 - \alpha) q_k$$

(10)

$$q_k = \left( 3 \tan \phi_k - 4 \tan \phi_{k-1} + \tan \phi_{k-2} \right) / (2T_f)$$

(11)

$$\alpha = \exp \left( -T_f / T_A \right)$$

(12)

\(T_f\) is the imaging period and \(T_A\) is time constant.

The nearest point can be derived as follows.

Step 1: \(\phi\) is acquired by the image and Eq. (8).

Step 2: \(A\) is calculated by the low-pass filter.

Step 3: \(B\) is obtained by Eq. (1).

The differentiation of the Eq. (9) is

$$\delta A = \frac{d}{dt} \left( \frac{1}{\cos \phi} \delta \phi \right) = \frac{2v \tan \phi}{B} \delta \phi$$

(13)

If \(\phi\) is nearly equal to 90 deg, the error of \(A\) due to the error of \(\phi\) is increased. Thus the filter cannot be used when the interceptor is far from the target. If \(\phi\) is nearly equal to 0 deg, the error becomes zero. The filter is useful around the target.

The position of the interceptor is estimated as follows.

<Far Point>
position in \(xy\)-plane: Nonlinear observer

distance out of \(xy\)-plane: None

<Near Point>
position in \(xy\)-plane: low-pass filter
distance out of \(xy\)-plane: Nonlinear observer

As often as imaging, the interceptor calculates the magnitude and the direction of maneuver. Interceptor can provide only one pulse thrust for one spin. If required magnitude is bigger than the half of one pulse thrust, interceptor starts the orbit change. The direction of this maneuver has to be in the plane perpendicular to the spin axis because all thrusters are located to the normal direction of the spin axis and the calculated orbit has to pass through as near as possible with minimum fuel consumption for the gravity measurement. In calculating the timing of injecting thrust, nutation motion is neglected because expected nutation angle is small and this motion is very complicated and so it’s difficult to calculate exactly. During the maneuver, imaging is not performed. Fig. 7 shows guidance and navigation algorithm.

4. Numerical Simulation

In this section, the simulations of guidance and navigation are shown. We assume:

- gravity of the asteroid is neglected.
- misalignments of thruster and principal axis of inertia are neglected.
- Attitude error is defined as 1 deg (spin axis) and 0.03 deg (the other axes), and 1 pixel error is included in the image information.
- interceptor can perform only one orbit change for one spin.
- we use one pair thruster for orbit control and two thrusters for attitude control. Thrust of one thruster is 0.406 N. 2 % error of thrust is considered.

Each parameter is defined as follows.

Initial position: \(r_0 = (-6, -6, 50000)\) [km]

(14)

Initial velocity: \(v_0 = (0, 0, -5)\) [km/s]

(15)

Target position: \(r^d = (-6, -6, 0)\) [km]

(16)

Initial position error: \(\delta r_0 = (50, 50, 200)\) [km]

(17)

In the case that the distance between the interceptor and the target is larger than 1000 km, the position of interceptor is estimated by the nonlinear filter. And in the other case, the position in plane is calculated by the low-pass filter and the distance out of plane is obtained by the nonlinear filter. The simulation results are shown in Fig. 8.

<Guidance Validation>

The interceptor is guided to the target position as shown in (a). The interceptor passes the nearest point at 10000 s. The error average of nearest position is \((-0.0325, 0.0372)\) [km]. The one sigma of guidance error is \((0.7727, 0.7434)\) [km]. If the initial position is changed from 50000 km to 35000 km, the error average and one sigma are \((-0.0827, 1.6539)\) [km].
[km] and \((11.464, 12.252)\) [km], respectively. These values are increased because the distance is too small to guide to the target position. Thrust of interceptor is very small and timing of orbit change is only one time for one spin. Thus it takes 50000 km to change orbit to pass the destination point.

<Navigation Validation>

The position error in plane is converged in zero around 9000 s. And it is constant after 9800 s, because the position is acquired by the low-pass filter. If it is calculated by the nonlinear observer, it is increased rapidly.

The distance out of plane is almost constant before 9950 s, because the camera elevation angle is constant at the point far from the target. It changes rapidly after the time, and it becomes zero at 10000 s.

![Graph of Navigation Error](image)

Fig. 8 Simulation results

5. Ground Experiment

We verify the guidance and navigation algorithm using Robotics Simulator (RS) with 9 degree of freedom in ISAS. This simulator has the chaser and the target. The chaser can realize in-plane motion (2 degree of freedom) and rotational motion (3 degree of freedom). The target can realize out-of-plane motion (1 degree of freedom) and rotational motion (3 degree of freedom). In the experiment, we use the chaser as the asteroid and the target as the interceptor. The target asteroid is irradiated. The angle is 45 deg. The nearest passing point of the chaser is estimated by the derived strategy, which quickly moves the chaser to the aimed point automatically via real-time image processing.

The experiment results are shown in Fig. 10. The scale is transformed from [mm] to [km].

![Experiment System](image)

Fig. 9 Experiment system
The interceptor is guided to the desired position as shown in (a). The distance out of plane is almost constant before 630 s, because the camera elevation angle is constant at the far point. It changes rapidly after the time, and it becomes zero finally. (a) and (b) are nearly equal to the simulation results. However, if the irradiating angle is changed to 0 deg. The guidance and navigation are failed because the position of the asteroid is not calculated precisely as shown in Fig. 11. Therefore the irradiating angle is considered in the process of image processing.

6. Conclusion
This paper showed the mission and systems of interceptor. An integrated guidance and navigation strategy was proposed.
1) Mission and System: The observation of interceptor consists of three steps. The three types of the flyby using interceptor were proposed according to the carrier. The instruments of the interceptor are shown.
2) Guidance and navigation: Combining optical navigation with the radio navigation, the relative position between interceptor and asteroid was estimated. The strategy of attitude and orbit control was proposed. It was verified by numerical simulations.

References