CONCEPT STUDY OF SPACECRAFT SYSTEM FOR END-OF-LIFE DEORBIT SERVICE USING LOW-THRUST FORMATION

Yoji Shirasawa(1), Yutaka Komatsu(1), Yuki Tsutsui(1), Haruhi Katsumata(1), Atsushi Wada(1), Takanori Iwata(1), Kazuma Adachi(2), and Tadanori Fukushima(2)

(1) Japan Aerospace Exploration Agency, 2-1-1 Sengen, Tsukuba, Ibaraki, Japan, Email: shirasawa.yoji@jaxa.jp
(2) SKY Perfect JSAT Corporation, 1-8-1 Akasaka, Minato-ku, Tokyo 107-0052, Japan

ABSTRACT

This paper discusses a system for an End-of-Life (EOL) deorbit service to deorbit a satellite no longer in operation. The system uses an ablation thrust force generated on the target satellite by a low-power laser beam. A service satellite keeps irradiating the laser on the target satellite, while maintaining formation flight with it by electric propulsion. To demonstrate the feasibility of the service, a concept study of the service spacecraft system was performed. Its summary here includes technical analyses of relative orbit determination and control, and relative attitude determination and pointing control. The paper also discusses the feasibility of a spacecraft system for deorbit servicing.

1 INTRODUCTION

The End-of-Life (EOL) deorbit service is an on-orbit service for changing the orbit of a satellite that has become non-operational. If a satellite is left on orbit after its operation is terminated or loses functionality, it becomes space debris giving a threat to spacecraft in operation. This can be prevented by developing a service to transfer such a satellite to a lower altitude where it will re-enter the atmosphere in a short time. This will reduce the creation of more space debris and maintain a sustainable space environment. Further, with an emergence of satellite mega-constellations, the need of the EOL deorbit service becomes greater than ever, and various organizations are studying or even undertaking such services.

The SKY Perfect JSAT Corporation has proposed an EOL deorbit service that uses a laser to change a target satellite's orbit [1][2]. In this service, an ablation thrust force is generated on a target satellite by a service satellite irradiating it with a low-power laser. This force moves the target satellite to a lower altitude. This method is safe because it does not require physical contact with the target satellite. It is also economical because it requires no additional propellant to apply the deorbiting force on the target satellite.

On the other hand, applying a thrust force with a laser beam to a non-cooperating target satellite requires the service satellite to have precise relative orbit control and pointing control. Also, the thrust force generated by the proposed laser ablation is very small, so it is necessary to continue laser irradiation for an extended time to transfer the target satellite to a lower altitude. There are many technical issues to deal with to achieve deorbiting, and the feasibility of the system must be demonstrated.

To demonstrate the feasibility of the EOL deorbit service system, the System Technology Unit (STU) of the Japan Aerospace Exploration Agency (JAXA) performed an early-phase mission study, in particular, a concept study of the proposed system to support the SKY Perfect JSAT Corporation. This paper presents the result of this study to demonstrate the feasibility of using the spacecraft system for the deorbit service. First, the paper gives the assumptions about mission requirements and preconditions. Next, an operational scenario is presented based on these assumptions, and the technical issues that the scenario faces are identified. To assess the scenario's feasibility, the technical studies' results are shown, including analyses of relative orbit determination and control, relative attitude determination, and pointing control.

2 PRECONDITIONS

2.1 Mission requirements

In this study, the general mission requirement is "lowering a 150 kg satellite from an altitude of 1200 km to 600 km." The target satellite is assumed to be part of a satellite constellation. At 600 km altitude, a 150 kg satellite with an area of 1.5 m^2 is expected to have a lifetime of 25 years or less, which meets the IADC Space Debris Mitigation Guideline. It is also assumed that the service satellite shall be developed based on a commercially available satellite bus to reduce mission costs.

2.2 Service satellite and target satellite

Next, the preconditions for target satellite and service satellite are given in Tabs. 1 and 2. The primary condition is that the service satellite uses low-thrust electric propulsion for efficient orbit transfer. Therefore, the service satellite needs its relative orbit control to be designed based on low-thrust orbit maneuvers. For laser systems, the focal range needed to cause ablation is also

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a significant constraint. Laser focal length is the distance from the service satellite to where laser ablation occurs, assumed to be between 100 m and 200 m. Ablation range is the range tolerance where laser ablation occurs.

Table 1. Assumptions for target satellite

Item	Symbol	Assumption
Initial Orbit	-	1,200 km circular orbit
Total Mass	M_t	150 kg
Size	-	1 x 1 x 1 m with solar paddles
Inertia Moment	-	25 kg m^2 for each axis
Controllability	-	Non-cooperative

Table 2. Assumptions for service satellite

Item	Symbol	Assumption
Total Mass	M_s	150 kg
Propulsion Type	-	Electric propulsion
Maximum Delta-V	-	800 m/s
Thrust Force	F_{el}	10 mN
Minimum Impulse	-	30 s
Pointing Accuracy	-	Knowledge :0.07º (1σ) Control: 0.08º (1σ)
Lifetime	-	> 5 years
Laser Ablation Force*	F_{ab}	0.72 mN
Laser Focal Length**	-	100-200 m
Ablation range	-	10-20 m

*The force is generated in the opposite direction to the normal vector of the ablated surface

3 OPERATIONAL CONCEPT

3.1 Mission operation concept

Based on the mission requirements and preconditions, the mission operation was designed as shown in Fig. 1. The figure shows three major phases in mission operation: the approach phase, the detumbling phase, and the deorbit phase. The service satellite is launched after it is confirmed that the target satellite has stopped operating. Next, in the approach phase, the service satellite approaches the target satellite using electric propulsion. This phase has three sub-phases corresponding to the navigation methods shown later. In the detumbling phase, the service satellite directs a laser on the target satellite to stop its rotation by ablation torque. In the deorbit phase, the service satellite irradiates the laser on the target satellite to cause it to deorbit by ablation force. Electric propulsion also causes the service satellite to descend and maintain formation with the target satellite.

3.2 Technical issues to enable the scenario

To evaluate the feasibility of the operation concept, three mission-specific issues were identified for study:

1. Relative orbit maneuver during the approach phase: feasibility of a low-thrust maneuver for the service satellite to approach the target satellite.

2. Laser-pointing accuracy for detumbling ablation: possibility to stabilize the target satellite's rotation within a reasonable time with the given laser pointing accuracy.

3. Simultaneous deorbiting during the deorbit phase: feasibility of a low-thrust maneuver for the service satellite to follow the target satellite's de-orbit while maintaining a relative position that is sufficient for laser irradiation.

The results of the technical studies for these issues will be shown in the next chapter.

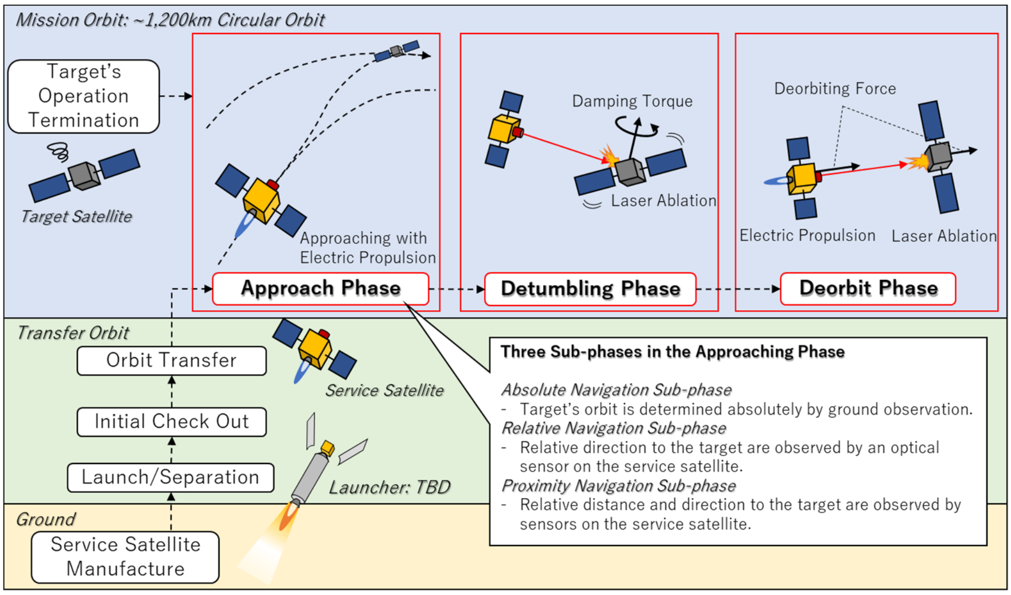

Figure 1. Operational concept

4 TECHNICAL ANALYSES

4.1 Relative orbit maneuver during the approach phase

In the approach phase, as shown in Tab. 4, three sub-phases are designed corresponding to navigation methods in reference to other rendezvous missions [3][4][5]. In the absolute navigation sub-phase, relative position is determined using ground-based observation, TLE, which is orbital element information of space objects published by NORAD [6]. In the relative navigation sub-phase, the target is captured in the service satellite's camera as a point, and direction to the target is obtained. In the proximity navigation sub-phase, direction and distance to the target are obtained.

To evaluate the relative orbit maneuver's feasibility using low-thrust electric propulsion during relative navigation and proximity navigation sub-phases, the accuracy of orbit determination when starting the relative navigation sub-phase is evaluated. Next, relative orbit determination accuracy is evaluated along a reference approach trajectory during the relative navigation phase. Then, relative orbit maneuvers using low-thrust propulsion are simulated.

To estimate the accuracy of orbit determination upon starting the relative navigation sub-phase, orbit determination accuracy using TLE during absolute navigation is evaluated. The accuracy of orbit determination by TLE depends on the shape, altitude of the target object, and time difference from epoch. The estimate is made using TLE information of ONEWEB-0012 from 2020/3/1 to 2020/3/31. This satellite has an altitude of 1170 km, an eccentricity of less than 0.003, and a mass of about 150 kg, close to those of our assumed target satellite. The value of TLE measured when t is propagated to $t+\Delta t$ using SGP4, and the difference to the other TLE measured at time $t+\Delta t$ is assumed as the orbit determination error. Comparisons are made for any combination of two TLEs during the above period then the RMS of their errors relative to the propagation time Δt is calculated. Tab. 3 shows the calculation result of the determination error. It was found that the along-track error was dominant increasing significantly with time from the epoch. Assuming that orbit propagation and position estimation of the target satellite will be performed within one or two days since the epoch, the relative navigation sub-phase will start with an initial relative orbit determination error of 20 km (1σ, along-track) and 1 km (1σ, radial).

Table 3. Estimated error of TLE orbit determination

Propagation date [day]	Determination error (RMS) [km]			
	Radial	Along-track	Cross-track	Position
1	0.27	8.32	0.04	8.33
2	0.66	48.43	0.09	48.43
3	1.88	130.55	0.18	130.57
4	5.06	236.12	0.16	236.17
5	11.66	381.98	0.14	382.16

Next, we estimate the accuracy of relative orbit determination during the relative navigation phase with angles only navigation by a simple calculation. In these calculations, relative orbital elements are estimated along a reference approach trajectory using the least-squares method based on angle information observed at 1Hz, which includes normally distributed errors of 0.071° (1σ). This mission assumes to use low-thrust propulsion; however, for simplicity, the reference trajectory design assumes impulsive delta-V as the thrust to increase the altitude for each orbit cycle.

These calculations are performed following steps:

1. Calculate initial relative orbital elements based on TLE information.
2. Estimate relative orbital elements by the least-squares method based on observed angle information includes normally distributed errors.
3. Calculate initial relative orbital elements of the next orbit considering the impulsive delta-V.
4. Repeat steps 2 and 3.

Table 4. Navigation method in three sub-phases

Sub-phase	(1) Absolute Navigation	(2) Relative Navigation	(3) Proximity Navigation
Relative range	~100 km	100 km-1 km	1 km-100 m
Navigation method	Absolute orbit determination	Angles only navigation	Model-matching navigation
Necessary info.	Orbital elements (TLE) of the target satellite and the service satellite	Relative direction from the service satellite to the target satellite	Relative direction and distance from the service satellite to the target satellite
Possible sensors	- Ground-based observation - GPS receiver (Service satellite)	- Narrow FOV optical sensor - IR camera (TBD) - GPS receiver (service satellite)	- Wide FOV optical sensor - Laser range finder (TBD) - GPS receiver (service satellite)
Constraints	Target is not visible.	Target is not always visible. (depends on the light source)	Target is always visible.

As the reference trajectory, the service satellite starts 100 km behind and 3 km lower than the target satellite. Observable time in which the target satellite is visible from the service satellite is assumed to be more than 30% of the orbit cycle by considering 1200 km altitude circular orbit.

Fig. 2 shows the result of calculating the determination error variance by relative range. The determination error in the along-track direction decreases linearly with the distance to the target, while the nadir direction error does not decrease significantly. It seems that relative distance information needs to be obtained to improve accuracy from about 10 km away from the target; however, it is known that this error can be improved by devising a coordinate system to estimate. The estimation in this study was performed in Clohessy-Wiltshire equations, which is linearized around the target position, and this results in low observability of the relative position. Mapping the estimation method into the non-linear coordinate increase the observability of the relative position and the relative orbit determination accuracy will be improved [7]. To evaluate the required performance of sensors for transitioning to the proximity phase, it is important to analyse the accuracy of orbit determination in detail based on this method in the next step of the study.

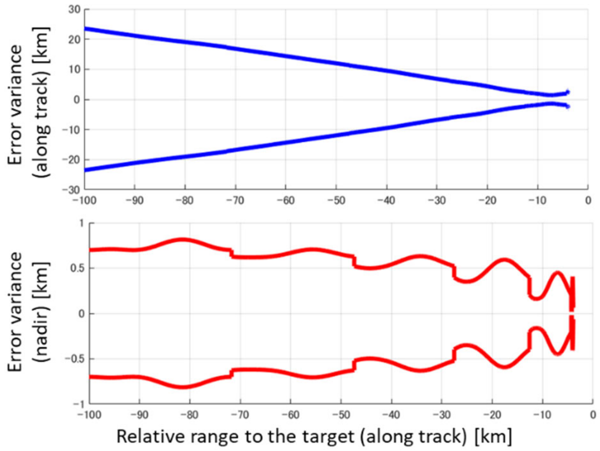

Figure 2. Variance in relative position determination error in the along-track direction (top) and the nadir direction (bottom)

Then, relative orbit maneuvers using low-thrust propulsion are simulated. We chose an approach maneuver method corresponding to navigation method. For relative navigation, the dual co-elliptic orbit is applied (Fig. 3), and for proximity navigation, the V-bar hopping approach is used (Fig. 4). The trajectories are described in the Local Vertical Local Horizontal frame (LVLH). In both cases, the trajectory is designed to avoid a possible collision caused by an emergency stop of the system. An important constraint is that low-thrust propulsion is used, so the service satellite's ascent speed is limited to about 1 km per orbit. By taking into account this constraint, the feasibility of the relative orbit maneuver is evaluated.

Figs. 5 and 6 shows the result of the simulation. Relative navigation of the service satellite starts from 100 km behind and 3 km below the target. In the dual co-elliptic orbit approach, it is found that the service satellite can approach behind the target satellite without overtaking it, even under the constraints of low-thrust electric propulsion. In the V-bar hopping approach, it can be seen that the distance to the target satellite can be reduced with an accuracy of at least about 9.8 meters due to the minimum impulse constraint of electric propulsion.

Note that a combination of relative orbit estimation and maneuver has not been simulated. A safe maneuver method based on quantitative orbit estimation errors should be demonstrated as a next step study.

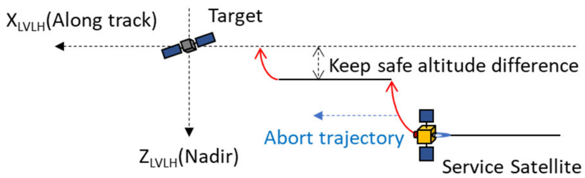

Figure 3. Dual co-elliptic orbit approach

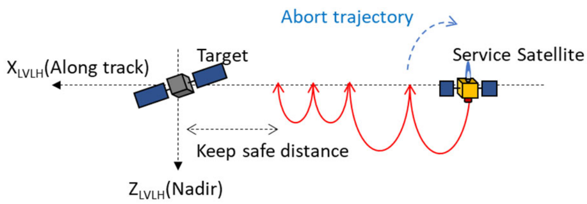

Figure 4. V-bar hopping approach

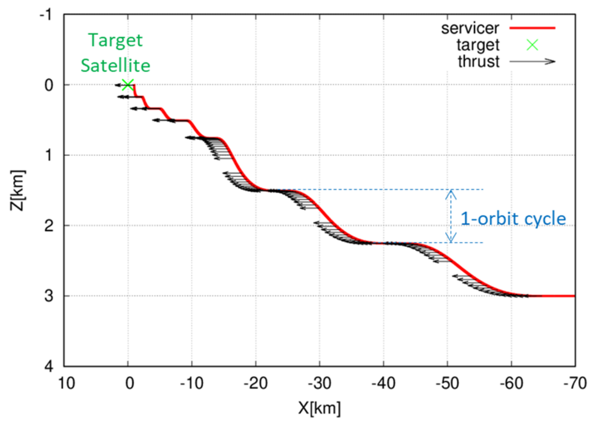

Figure 5. Simulation result of dual co-elliptic approach.

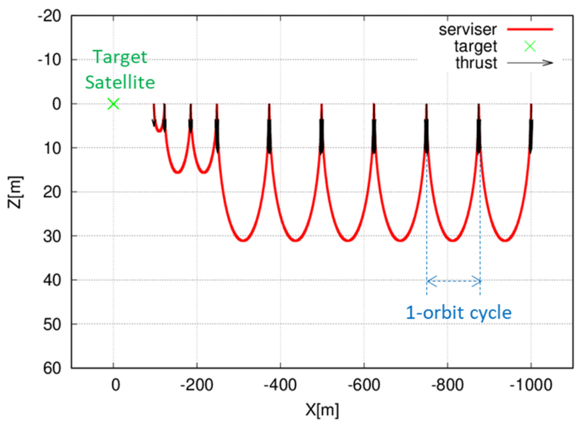

Figure 6. Simulation result of V-bar hopping approach.

4.2 Laser-pointing accuracy for detumbling ablation

In this section, the requirement for laser pointing accuracy is evaluated for stabilizing the target's rotation in the detumbling phase. If the target is rotating, the effective deorbit thrust force by laser ablation will be reduced, so the service satellite needs to stop the target's rotation by laser ablation.

As shown in Fig. 7, to generate detumbling torque on the target satellite, the service satellite keeps its relative position in front of the target satellite along its orbiting direction. From this position, the service satellite can reduce the target satellite's angular velocity around the Y-axis and the Z-axis in LVLH by irradiating a part of the target offset from the centre of mass. The target's angular velocity around the X-axis in LVLH is translated into one around the Z-axis due to the orbital motion. Therefore, the limited controllability at one instance is alleviated over time, and a three-axis angular velocity can be detumbled using orbital motion.

As shown in Fig. 8, the propulsive force is generated perpendicular to the ablation surface, and the irradiation point will change with the rotation of the target. Hence, the ablation force and moment arm of torque on the target are not constant. Numerical simulations are performed to study this issue to estimate the time required to stabilize the target's rotation by calculating the ablation torque as it changes with relative attitude.

The simulation cases set by the target's tumbling pattern are shown in Fig. 9. In cases (a)-1, (a)-2, and (a)-3, four surfaces are parallel to the rotation axis. In cases (b)-1 and (b-2), two surfaces are parallel to the rotation axis. In cases (c)-1 and (c)-2, no surfaces are parallel to the rotation axis. In case (a)-1, (b)-1, and (c)-1, the rotation axis is in the orbital plane. In cases (a)-2, (b)-2, and (c)-2, the rotation axis is orthogonal to the orbital plane. Case (a)-3 is simulated to confirm that rotation around X-axis and Z-axis have equivalent effects due to the orbital motion.

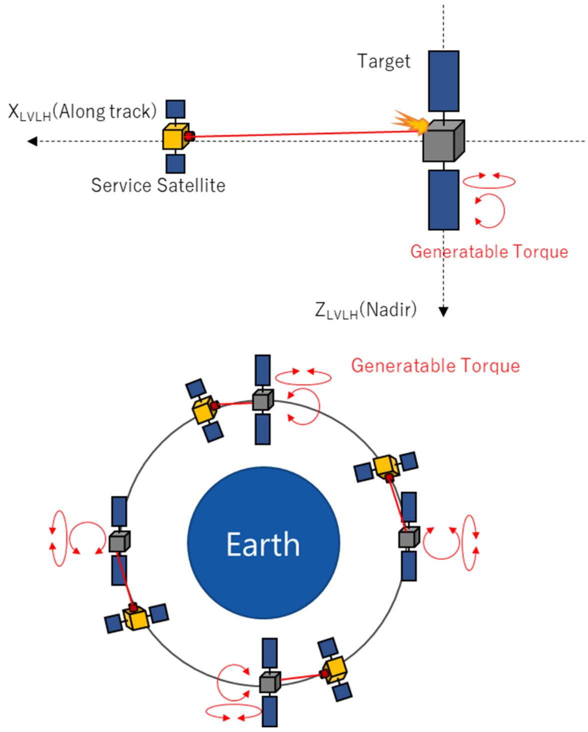

Figure 7. Relative position for detumbling phase

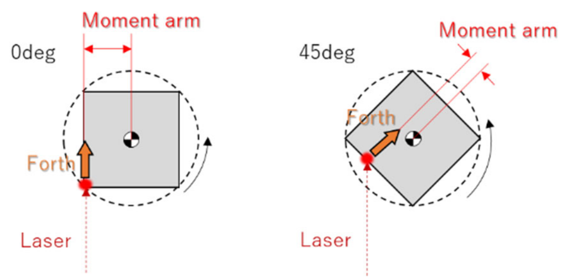

Figure 8. Detumbling torque model by laser ablation

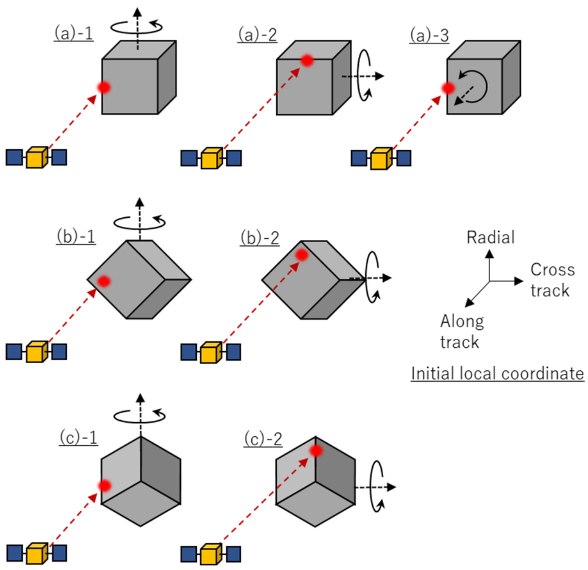

Figure 9. Simulation cases (tumbling pattern)

The irradiation target is fixed at 0.5 m away from the centre of mass of the target satellite. If the angle between the axis of rotation and the direction vector from the service satellite to the target satellite is small, the effective torque for detumbling is low. Hence the irradiation is turned off while the angle is within 0±30º or 180±30º; it is on otherwise. The simulation includes bias and random laser pointing error, as shown in Fig. 10.

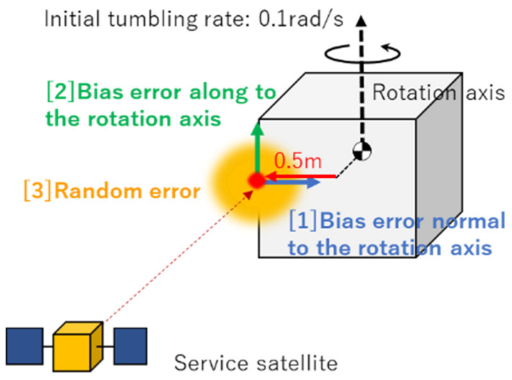

Figure 10. Laser irradiation error model

Figs.11, 12 and 13 show the results of simulations of each type of error. Left-side figures show cases that the rotation axis is in orbital plane, and right-side figures show cases that the rotation axis is orthogonal to orbital plane. It was expected that the latter to be able to stabilize quickly because there is no turn off time; however, the results is not so simple, and there is no clear trend. The result of (a)-3 is almost same as the one of (a)-1, and it confirms that the rotations around the X-axis and the Z-axis have equivalent effects due to the orbital motion.

It found that a bias error normal to the rotation axis is most critical, and if it is within a range of -0.25 m to +0.45 m, the target satellite with an initial rate of 0.1 rad/s will be stabilized within about ten days. These results indicate that the irradiation target should be fixed at 0.35 m away from the centre of mass to maximize the bias error tolerance.

If the random error is less than 0.84 m (1σ), it can be stabilized within about ten days.

Errors of ±0.35 m and 0.84 m (1σ) at a distance of 200 m from the service satellite are equivalent to pointing errors of ±0.1º and 0.24º (1σ), respectively. The assumed pointing accuracy of the service satellite in Tab. 2 satisfies these pointing requirements. However, the accuracies of relative attitude estimation and laser pointing control also should be considered for these requirements. These will depend on an image processing and laser subsystem's design, and these will be studied in the next step.

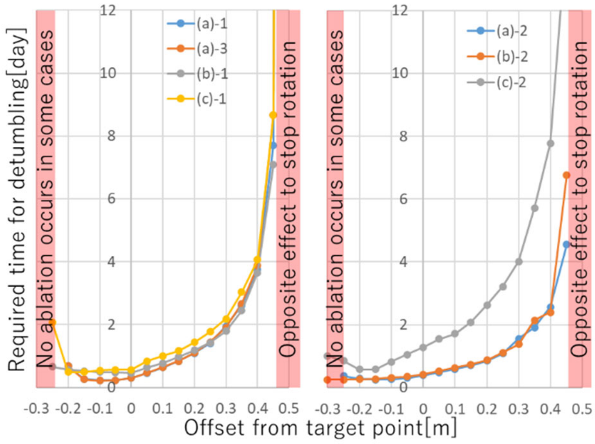

Figure 11. Estimation of required time for detumbling with bias error normal to the rotation axis

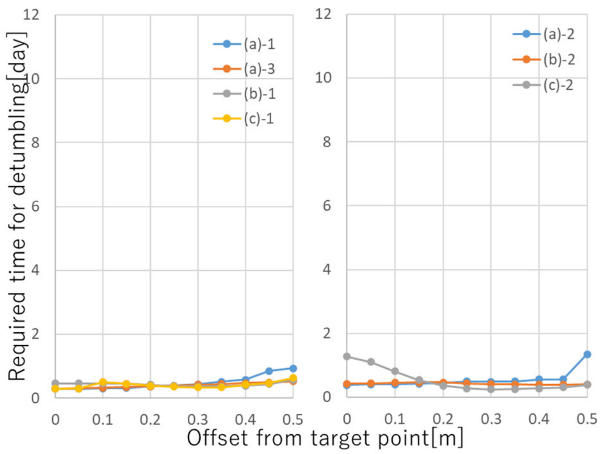

Figure 12. Estimation of required time for detumbling with bias error along to the rotation axis

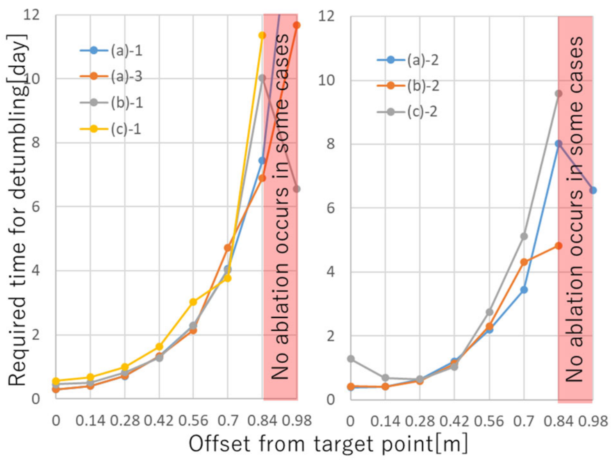

Figure 13. Estimation of required time for detumbling with random error

4.3 Simultaneous deorbiting maneuver during deorbit phase

For deorbiting maneuver design, some constraints are considered. The first is to maintain the relative positions. To generate deorbiting force by laser ablation, the service satellite is required to maintain its relative position, ahead of the target satellite, within a range of distance variation of about ±20 m. The second is the available delta-V for the relative orbit control. To make the spacecraft descend from 1200 km to 600 km, more than 305 m/s of effective delta-V needed to be allocated to the service satellite's deorbit, which is more than 38% of the total impulse (800 m/s). The third is the time available for the deorbit control. Generating 305 m/s of delta-V requires more than 736 days of propulsion by laser ablation, which is more than 40% of the lifetime of five years.

For these constraints, a deorbit trajectory is designed to descend the service satellite almost simultaneously in loose formation with the target satellite, as shown in Fig. 14. It is also designed in one orbit as one unit so that the service satellite returns to the original relative position and velocity after orbit to reduce additional orbit control.

In the trajectory design is also needed to avoid collisions caused by an emergency stop of electric propulsion or laser ablation. For passive abort safety, the service satellite's orbit is slightly lower than the target.

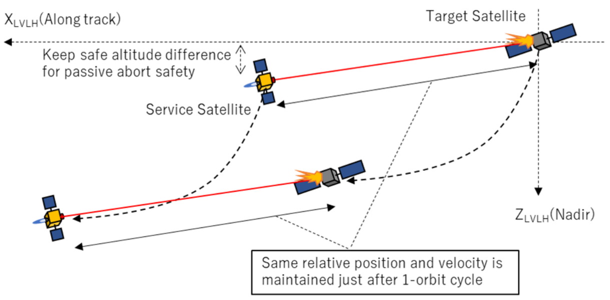

Figure 14. Relative deorbiting maneuver concept

For relative maneuver design, it is assumed that the service satellite's deorbit by electric propulsion and the deorbit of the service satellite by laser ablation are operated exclusively due to an electric power constraint. That is, if the orbital period is T and the operating time of electric propulsion per orbit is T_{el}, the rest of the time T-T_{el} is the operating time for ablation.

The initial position of the service satellite is set to be 100 m ahead of the target satellite, and the altitude is offset downward by ΔZ. The value of ΔZ is designed for passive abort safety and is obtained by iteration. This difference in altitude between the target satellite and the service satellite causes an along-track drift motion. An impulse per orbital period T required to cancel the drift is calculated as $12\pi^2 M_s \Delta Z/T$ by Clohessy-Wiltshire equations.

This impulse is generated by tilting the thrust vector for the deorbit to reduce the thrust vector control maneuver. Assuming that the tilt angle is θ and the electric propulsion force is F_{el}, the deorbit thrust component, and drift cancellation thrust components can be expressed as $F_{el}\cos\theta$ and $F_{el}\sin\theta$, respectively. To cancel the drift motion, $F_{el}\sin\theta$ is designed to satisfy Eq. 1.

$$\frac{F_{el}\sin\theta\, T_{el}}{M_s} = \frac{12\pi^2 \Delta Z}{T} \tag{1}$$

The service satellite's delta-V generated by the deorbit thrust component per orbit should coincide with one of the target satellites generated by ablation thrust force per orbit. To coincide these delta-Vs, $F_{el}\cos\theta$ and T_{el} should satisfy Eq. 2.

By solving simultaneous equations of Eqs. 1 and 2, tilt angle θ and the electric propulsion operating time T_{el} is derived as follows.

$$\frac{F_{el}\cos\theta\, T_{el}}{M_s} = \frac{F_{ab}(T - T_{el})}{M_t} \tag{2}$$

From the assumptions of Tabs. 1 and 2, the initial altitude offset is calculated as ΔZ=1.98 m by iteration. Then, by solving Eqs. 1 and 2, θ and T_{el} are calculated to be about 51.7° and 683 s, respectively. The electric propulsion and laser ablation's thrust pattern are designed symmetrically during an orbit, as shown in Fig. 15. The service satellite returns to the original relative position and velocity after one orbit.

Fig. 16 shows the result of the simultaneous deorbiting maneuver based on the designed thrust pattern calculated by numerical simulation. The simulation shows the following results:

- The relative distance variation is within the range of -2.2 m to 7.9 m, which is in the range where laser ablation is always possible.
- The ratio of effective ΔV for the service satellite's deorbit to the total generated ΔV is about 44%, which is higher than 38% in the condition.
- The ratio of the effective time of propulsion by laser ablation to the orbital period T is about 86%, which is higher than 40% in the condition.

These results confirmed that the service and target satellites can descend simultaneously under respective constraints.

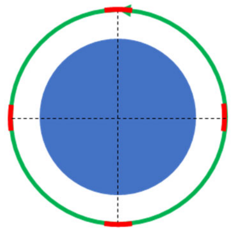

Figure 15. Thrust pattern during one orbit

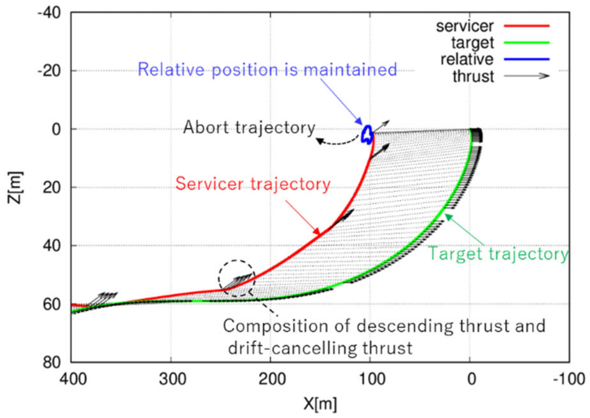

Figure 16. Simulation result of the deorbiting trajectory

In the deorbit simulation, descent altitude over one orbit is calculated to about 57 m. This is just a simulation of only one orbit, and about 10,000 orbits with the given thrust pattern will be required to descend 600 km. To achieve this maneuver, on/off control and the lifetime of the electric propulsion system are critical parameters for the mission's feasibility. These parameters will depend on the subsystem design of the satellite bus, and these will be studied in the next step.

5 CONCLUSIONS

This paper introduced a feasibility study of the EOL deorbit service using low-power laser ablation. An operational concept using low-thrust electric propulsion for the service satellite was designed to satisfy assumed mission requirements. Based on the concept, the relative orbit determination and control to approach the target satellite, the detumbling of the target satellite, and the deorbit maneuver control were designed, and their feasibility was demonstrated through numerical simulations. In addition, critical parameters for the feasibility of the system were identified. For future study, a detailed subsystem level design using the results is needed to realize the system.

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