A Novel Concept of Cost-Effective Active Debris Removal Spacecraft System

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ABSTRACT

Post Mission Disposal (PMD) alone is an insufficient solution to stabilize the orbital debris environment, but combining Active Debris Removal (ADR) with PMD and other means of debris mitigation is considered a promising remediation method. The cost reduction of ADR is essential to make it a sustainable business. The major cost driver of ADR is the ADR spacecraft and its launcher, both of which are largely influenced by the spacecraft size. A trade-off study of the ADR spacecraft system has been conducted to minimize its size, resulting in the novel concept of a cost-effective ADR spacecraft system. The authors introduce the combination of electric propulsion (EP) and air drag utilization for the debris deorbit phase so as to lower the EP requirement, thereby leading to cost reduction of the ADR spacecraft. The minimum delta-V reentry method is also considered to reduce the propellant mass for final reentry. EP performance (e.g. thrust, specific impulse, mass) is modeled as a function of power consumption, and optimum EP performance has been explored to minimize overall mass of the ADR spacecraft, including its electrical power subsystem with this EP model. Given the minimum thrust limit to satisfy the mission duration requirement, a higher thrust-to-power ratio is preferable for debris removal electric propulsion. This paper discusses the optimal EP performance requirements in detail, and the results show that a roughly 500-kg ADR satellite using a 600-W-class Hall thruster can remove about three tons of a rocket body debris from an 800-km circular orbit within 600 days.

Nomenclature

μ	: Earth gravity coefficient	ρ	: Atmospheric density
a	: Semi-major axis	C_d	: Drag coefficient (2.0, fixed)
v	: Orbital velocity	A/M	: Area-to-Mass ratio
F	: Thrust of EP	f	: Acceleration by EP
LV	: Local vertical debris attitude	LVLH	: Local horizontal debris attitude

1 INTRODUCTION

Active Debris Removal (ADR) is considered a promising orbital debris remediation method together with Post Mission Disposal (PMD) and other means of debris mitigation [1, 2]. However, ADR using chemical propulsion entails a heavy propellant mass and a high cost for manufacturing, launch, and operation. Therefore, the cost reduction of ADR is essential [3, 4].

The major cost of ADR is the ADR spacecraft and its launcher, both of which are largely influenced by the spacecraft size. This study focuses on the deorbit phase because the propellant mass is generally the most dominant aspect of the ADR spacecraft size.

The total delta-v from LEO to the ground is around 300 to 400 m/s, and thus not so large compared to the required delta-v for geo-synchronous orbit insertion (i.e. apogee kick). The propellant mass for delta-v is proportional to the system mass including the target debris. Given this nature, the removal of a large debris object requires a large propellant mass.

The use of an efficient propulsion system such as electric propulsion (EP) or an electro-dynamic tether (EDT) is preferable to reduce the total propellant mass for ADR. An EDT with a novel cathode combination is apparently a

good candidate for both PMD and ADR [5-7]. However, verifying the EDT as a propulsion subsystem and increasing the technology readiness level (TRL) still require more on-orbit experiments. Plasma thrusters (e.g. Hall thruster, ion engine) are widely used in commercial spacecraft (communication satellite) [8] and exploration spacecraft (*Hayabusa*) [9]. The Hall effect thruster (HET) has a relatively shorter operation life and larger thrust. The ion engine system (IES) requires a relatively more complex power supply system. Both the HET and IES have TRL 9, and total system cost is the point of selecting the propulsion system. These plasma thrusters are candidates for the ADR propulsion subsystem in this study.

Another technology to produce the required delta-v is the utilization of air drag. The lower the altitude, the higher the air density, and thus air drag can be used as a primary source for reducing the orbital altitude.

The authors introduce the combination of electric propulsion and air drag utilization for a cost-effective ADR spacecraft.

2 NEW ADR CONCEPT

When assuming the removal of debris from a circular orbit at an altitude of 600 km, with a reentry interface altitude of 120 km, the required total delta-v for reducing the altitude is 274.2 m/s with low-thrust spiral orbit transfer. The delta-v from 350 km to 120 km is 135 m/s, or about a half the delta-v from 600-km reentry. If air drag is large enough from 350 km to the earth within the appropriate duration, the required delta-v and propellant mass will be reduced by half. Figure 1 shows this new ADR concept. At first, EP of the ADR spacecraft is used as the primary deceleration propulsion from the debris altitude. When the orbital altitude becomes low enough to utilize the air drag, EP operation is seized, and the natural deceleration of air drag is used to further reduce the altitude until reentry.

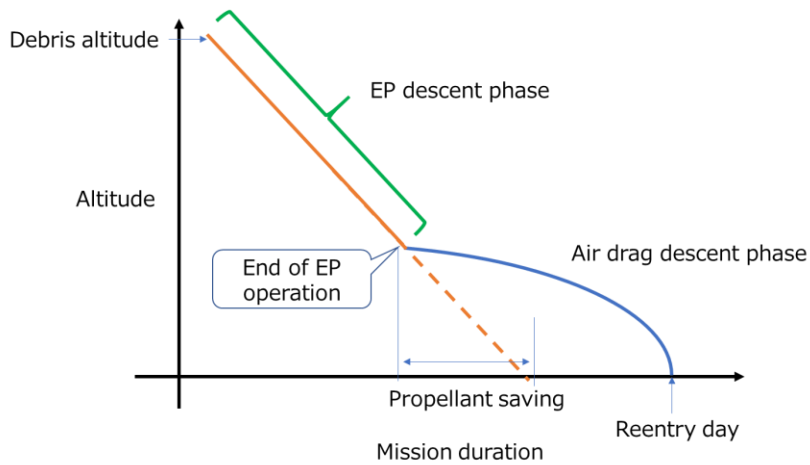

Fig.1. ADR descent profile schematics.

The debris considered in this study is a large debris object (i.e. large rocket body) having a significant effect on the debris environment. In some cases, the natural reentry of such a large debris object may exceed the casualty risk probability of 10^{-4}. In cases where ADR must fulfill the casualty risk limit, controlled reentry is required. Controlled reentry is typically conducted from an altitude of around 300 km with delta-v of 100 m/s. This reentry maneuver requires a large propellant mass with high impulse thrust. Bacon [10] proposed reducing the delta-v requirement for controlled reentry (called minimum-dv reentry), and this system study adopts said method for the final reentry maneuver at delta-v of about 10 m/s for controlled reentry.

As a result, the proposed ADR concept consists of three propellant reducing technologies: electric propulsion, air drag utilization, and minimum-dv reentry. The reasonable ADR spacecraft size (relative to mass and cost) will be achieved with these propellant reducing technologies.

3 MODELING AND NUMERICAL CALCULATION

3.1 Descending profile calculation

In order to simplify the problem, the debris orbit is assumed to be a circular orbit. The governing equation is derived from the orbital dynamics equation.

The relationship between orbital velocity v and semi major axis a is $\frac{\mu}{a} = v^2$, and those time derivatives are as follows:

$$\frac{da}{dt} = -\frac{2a}{v}\frac{dv}{dt} \tag{1}$$

dv/dt in each technologies are as follows, respectively.

$$\frac{dv}{dt} = \frac{\mathrm{F_{EP}}}{\mathrm{M_{debris}} + \mathrm{M_{satellite}}} \text{ (for EP)} \tag{2}$$

$$\frac{\mathrm{dv}}{\mathrm{dt}} = \frac{1}{2}\rho C_{\mathrm{d}}\mathrm{v}^2\mathrm{A/M}(\text{for atmospheric drag}) \tag{3}$$

By substituting Eq. 2 into Eq. 1, the low thrust change in the semi-major axis is obtained as the circular orbit derived in [11] as follows:

$$\frac{da}{dt} = -\frac{2a}{v}\mathrm{f},\ \text{where f is acceleration of EP} \tag{4}$$

And by adding the atmospheric drag into Eq. 4, the total change in the semi-major axis is obtained as follows:

$$\frac{da}{dt} = -\frac{2a}{v}\mathrm{f} - \rho\mathrm{C_d}\mathrm{avA/M} \tag{5}$$

The descent profile during the mission duration is obtained through the integration of Eq. 5. Since atmospheric density is a complex function of altitude, the numerical integration method is used for this system study.

A rough numerical calculation scheme is preferable for a better understanding of the system parameters; thus, the first order integration has been conducted with fixed atmospheric density in the small altitude bin. The air density model for the air drag descent phase calculation is taken from the NRL-MSISE-00 model [12]. For simplicity, orbital average density is calculated and tabulated in advance. The duration time in the altitude bin is calculated as follows:

$$\Delta T = \frac{\Delta a}{\frac{2a}{v}\mathrm{f} + \rho\mathrm{C_d}\mathrm{avA/M}} \tag{6}$$

where Δa is the altitude bin. Equation 6 is calculated repeatedly until reentry.

Drag coefficient C_d is fixed at 2.0 for any attitude. The target debris object is assumed to be cylindrical in shape, 4 m in diameter x 10 m in length (i.e. simplified H-2A rocket body shape), and the mass of the debris object with the ADR spacecraft is fixed at 3500 kg.

The GMAT [13] simulation and rough estimation results are compared to check the rough estimation accuracy. Figure 2 shows the decent profile of this ADR method from the 600-km circular orbit with 30 mN electric propulsion. The upper figure shows the GMAT simulation results; the lower figure shows the rough calculation results. Both figures show a similar descent profile (i.e. EP operation seized at 400-km altitude, reentry occurring about 420 days after the start of descent). A few days of the numerical integration accuracy is sufficient for this system study, and the figure shows that the rough calculation is adequate.

Both figures show a bend profile of about 320 days or 320 km, where the debris attitude changes from local vertical (LV) to local horizontal (LVLH). Given the low air density above 320 km and stable debris attitude with the gravity gradient, LV attitude can get higher air drag due to the larger cross-sectional area. Below 320 km, the air density becomes higher and the weathercock effect begins. As maintaining LV attitude against the air drag makes no sense, LVLH attitude will be taken.

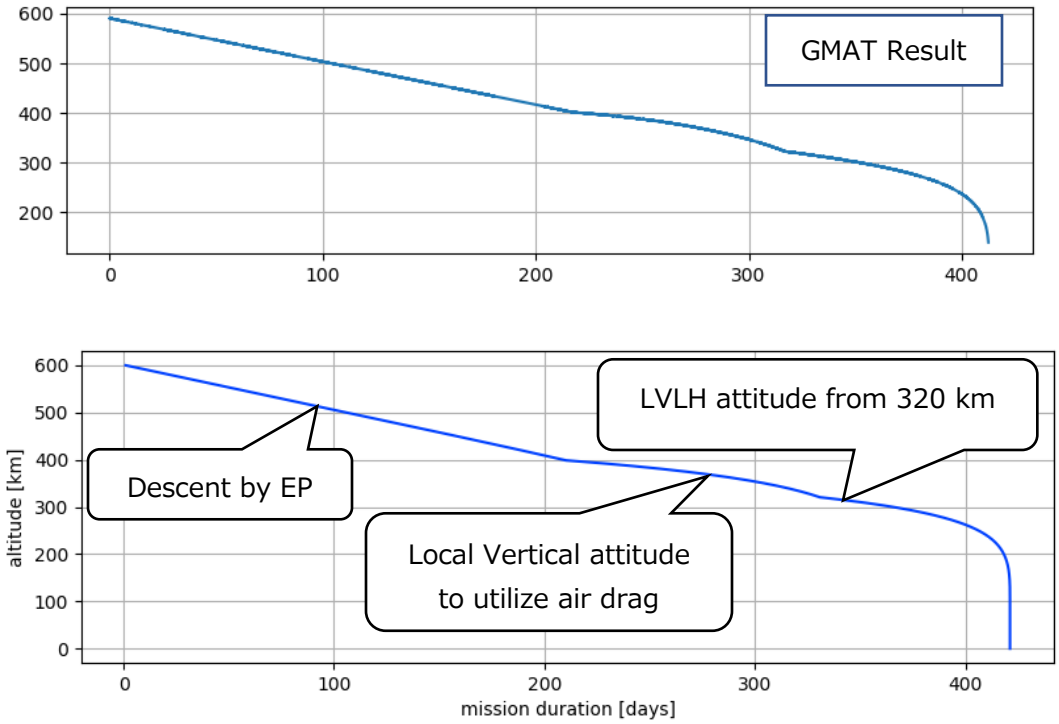

Fig. 2. Comparison of descent profile calculation (upper GMAT, lower Rough calculation)

3.2 EP modeling:

For this system study, it is important to model the EP and the spacecraft. The spacecraft is modeled as a function of total power demand using the SMAD theoretical first unit (TFU) model [14].

The EP performance is assumed to be a function of the required electrical power. The thruster performances (i.e. thrust, specific impulse, thruster head mass) are functions of the power requirement derived from the linear fitting of the actual data found in [15-17]. Power Processing Unit (PPU) performance is also a function of power, and the mass and cost of PPU are derived from the linear fitting of the actual data found in [18, 19].

The spacecraft's electric power consumption excluding EP power is fixed at 700 W; thus, the spacecraft's total power consumption is a function of EP power.

The mass of the electric power subsystem (EPS) consisting of the solar array, battery, and distributors is also function of total power modeled in the SMAD EPS. The overall spacecraft mass is derived from the mass fraction of EPS to the total spacecraft mass described in the SMAD model. Figure 3 shows the functions that describe these models used in this study, where "mp" is the mass of required propellant and "me" is the mass of the thruster and the PPU. "tfu" is the cost of the spacecraft TFU and "bus mass" is the total spacecraft mass. The total cost is the sum of tfu, PPU cost, and operation cost, as derived from SMAD (160k USD/year per person assuming three-person operation).

Figure 3 also shows the propellant tank and support structure mass assumed to be a function of the propellant mass.

```
function eps_mass(p):                                  # EPS mass calculation function to input power p
    sap = p/25                                         # sap mass (SMAD 10.4.6 Table 10-27)
    eta = 0.9                                          # LiB charge-discharge coefficient (assumed)
    dod = 0.5                                          # LiB DOD for LEO 3 year (assumed)
    bat = p*0.6/eta/dod/45                             # 45 W hr/kg and 0.6 for eclipse duration (hr) (SMAD 10.4.6 Table 10-27)
    pcu = 0.02*p                                       # PCU mass (SMAD 10.4.6 Table 10-27)
    reg = 0.025*p                                      # Converter mass (SMAD 10.4.6 Table 10-27)
    return sap+bat+pcu+reg

function sat_tfu_eps(p, mp, me):                       # TFU and bus mass function of power p, propellant mass mp and thruster mass me
    epsmass=eps_mass(p+700)                            # EPS mass calculated by function with EP input power p and the bus power 700W
    bus_mass = 73.3/27.9*epsmass                       # bus mass from fraction (Table A-2)
    mp2tank = 1.1                                      # tank mass ratio to the propellant mass (assumed from the catalogue data)
    mstr = 1.2                                         # tank support structure ratio to the propellant and tank mass (assumed)
    bus_mass = bus_mass + mp*mp2tank*mstr + me*mstr    # Total mass
    tfu = 43.0 * bus_mass                              # Bus TFU from bus mass (Table 20-5)
    tfu = 1.341 * tfu + (10.4+4.9)* bus_mass           # TFU adjustment for IA&T, Program Level, and LOOS (Table 20-5)
    return tfu, bus_mass
```

Fig. 3. Function of EPS and TFU

4 SIMULATION RESULTS and DISCUSSIONS

4.1 Decay Simulation

Figure 4-1 shows the orbital altitude decay profile from the circular orbit at an altitude of 600 km. The target mission duration is one year (365 days).

Because a lower power HET has a shorter life, the operation life for 400 W and smaller EP is set to 200 days. Then the 400-W case exceeds the HET operation life before reaching the EP off altitude, and cannot fulfil the required duration of 365 days. And it takes about 450 days until reentry. EP with power greater than 400 W can fulfil the 365-day requirement. From this result, the HET with a operation life of 200 days may be sufficient. The HET with power greater than 600 W thus has an adequate life margin, so that one with 600 W to 1 kW of power can be a candidate thruster for this ADR spacecraft.

The IES has a much longer operation life, but a lower thrust-to-power ratio means that the descent rate is slower than that of the HET. Power of 800 W and less IES thrust cannot fulfil the 365-day requirement due to the slower descent rate. Therefore, the IES has too much operation life and smaller thrust than that of the HET for the ADR mission.

Figure 4-2 shows the mass and cost of the ADR spacecraft. Figure 4-3 shows the propulsion system performance relative to fulfilling the one-year mission duration requirement. Even though the IES has better specific impulse (Isp) than the HET, the HET is still better in terms of the total mass and cost of the spacecraft. If Isp exceeds 1000 sec., the difference in propellant mass has less effect on the total spacecraft mass than the EP power demand. And as cost is closely related on the spacecraft mass, the cost difference between the IES and HET spacecraft systems stems from the thrust-to-power ratio. The HET has less power to generate adequate thrust for an appropriate descending rate, and less power means less mass of the overall EPS and spacecraft, which also effects the overall cost including the launch.

In this case, lower power means a lower mass and cost of the spacecraft. The propulsion system requirement for the ADR mission is to apply EP at a high thrust-to-power ratio, which means that the HET is the most preferable candidate to fulfil appropriate ADR mission duration.

If the conventional chemical propulsion is used from 600 km to re-entry of 3000 kg debris, total required propellant mass is about 200 kg and the dry mass of spacecraft is about 850 kg. Then the spacecraft total mass at launch is 1050 kg. The proposed three propellant reduction technologies, i.e. EP, air drag utilization and minimum-dv re-entry, can reduce total spacecraft mass is about 600kg or 60%, and the cost is saved about 47 % with this model.

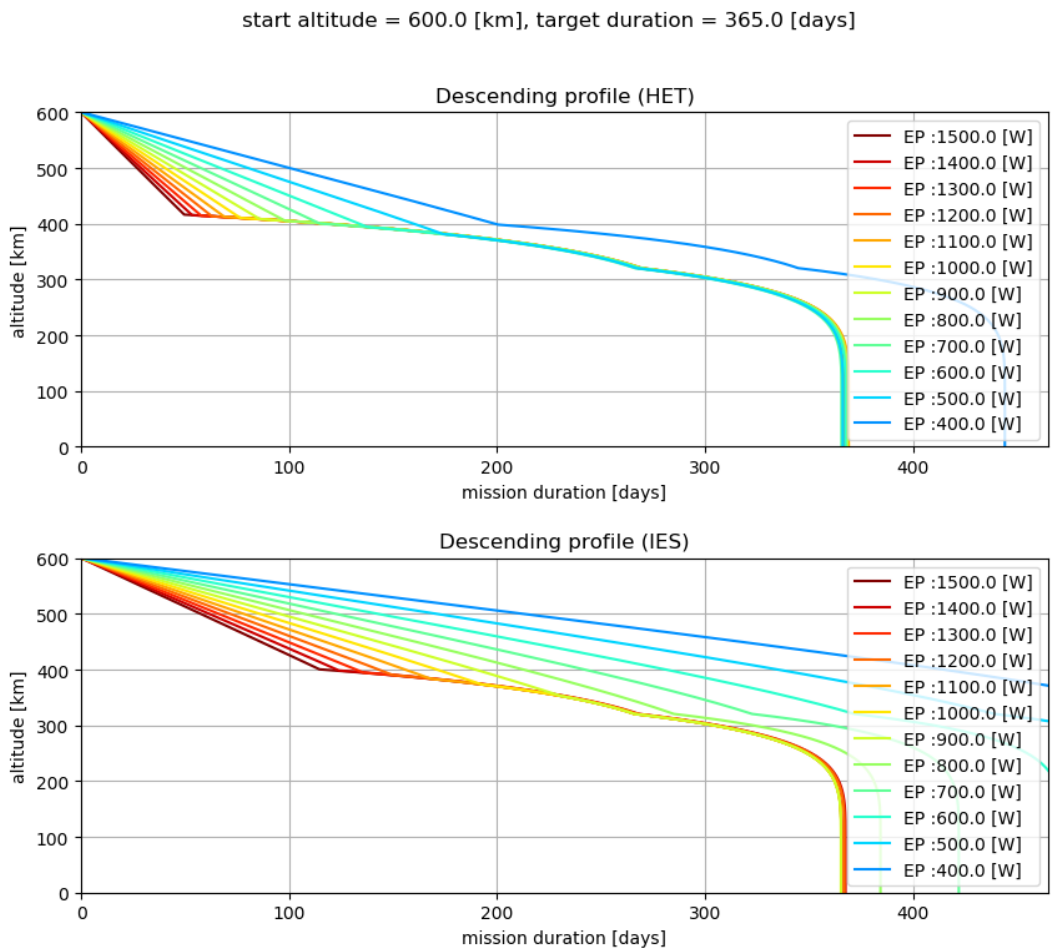

Fig. 4-1. Orbital altitude decay profile of each EP power (upper: HET; lower: IES)

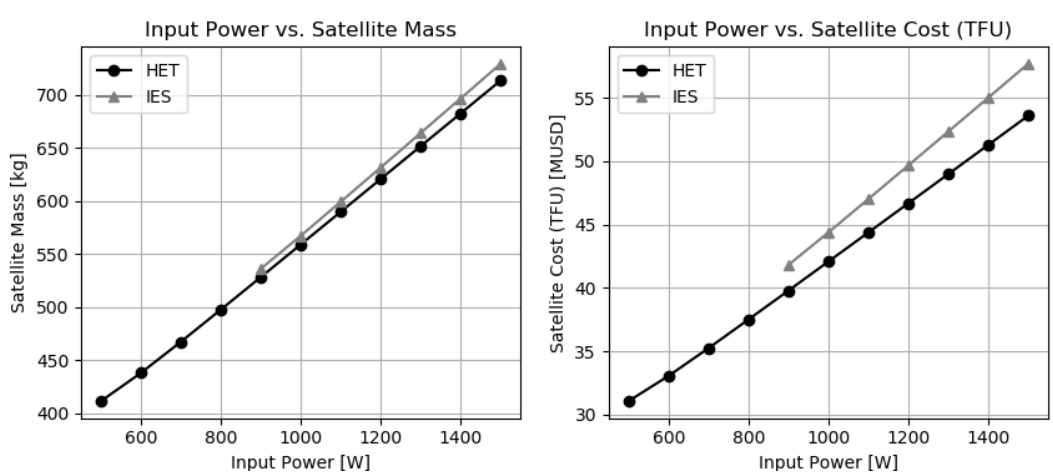

Fig. 4-2 Satellite mass and cost vs. EP input power

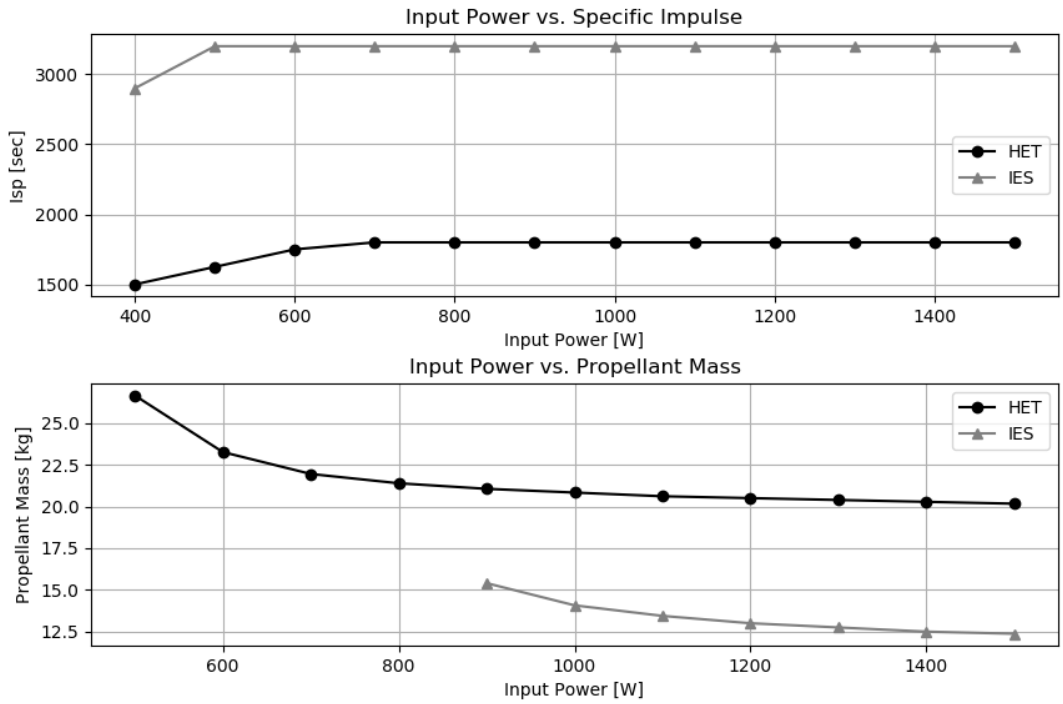

Fig. 4-3 Thruster performances

Figure 5-1 shows the descending profile with different mission duration requirements with fixed EP power of 600 W. The EP descent phase starts from the 800-km circular orbit.

The different EP seized altitudes are obtained according to the different mission duration requirements. The HET has a variety of mission durations ranging from 400 to 950 days in this calculation with the same HET performance and the same spacecraft system design.

Figure 5-2 shows the mass and cost of the ADR spacecraft. In this case, the IES has smaller mass than the HET spacecraft, but the cost is different. This feature stems from the relatively high cost of the IES PPU. There seems to be an optimum mission duration in terms of HET cost, due to the operation cost.

4.2 Discussion

The IES has better performance from the standpoint of specific impulse, and the spacecraft mass is lighter in general high delta-v missions. But in the case of ADR, the mission duration is also important, particularly in terms of reliability, and the thrust-to-power ratio is the key feature of this concept to reduce the cost of ADR. Because high thrust with a smaller thruster aperture requires high density plasma, the HET is better suited for ADR than the IES.

The ADR spacecraft equipped with the HET can meet different mission duration requirements, and the same common bus system design could possibly be applied to various ADR missions from LEO.

There is another ADR concept, especially in the GEO region; consequently, the propulsion system requirement will be different.

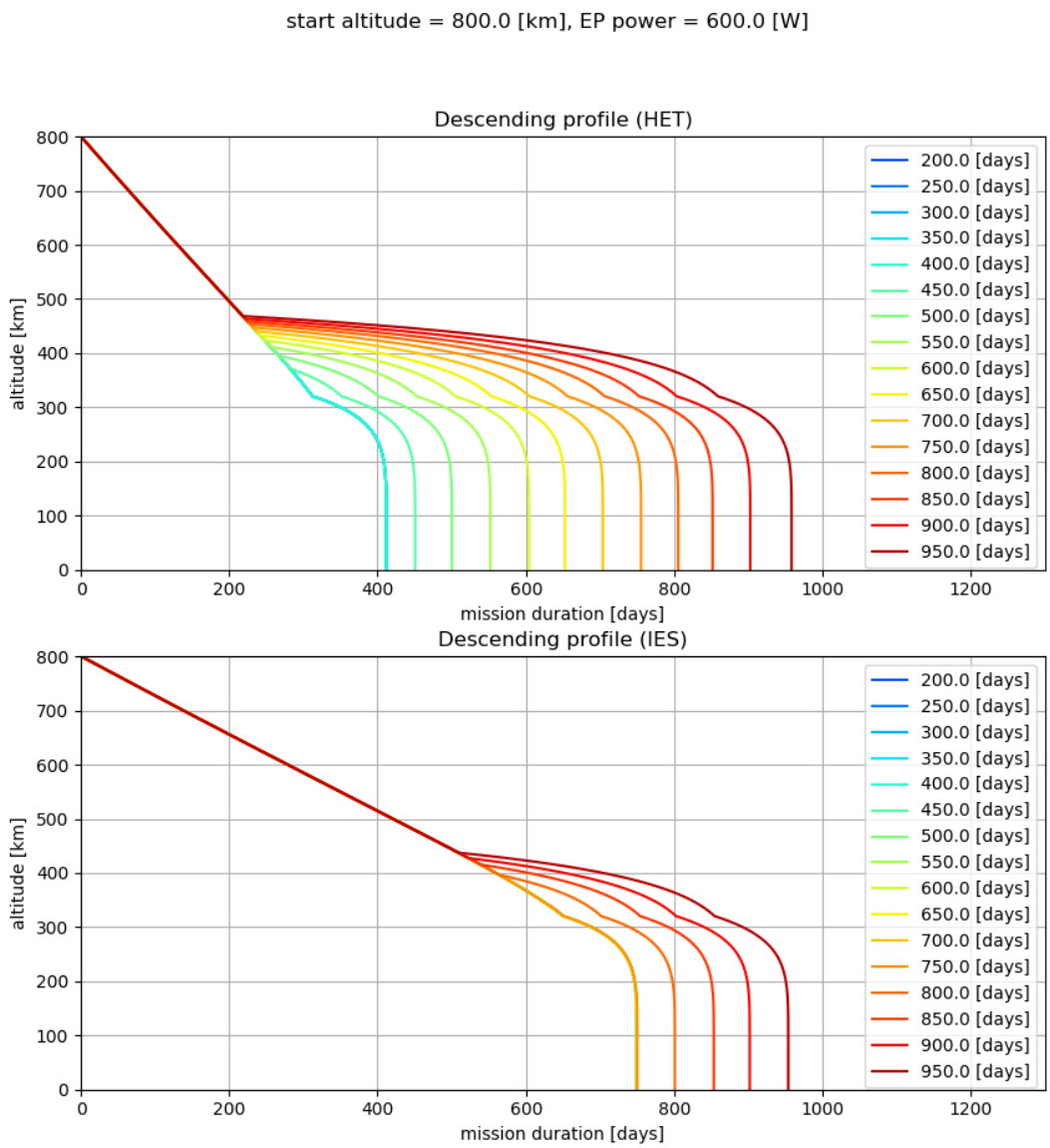

Fig. 5-1 Descending profile of each mission duration requirement

Figure 5-2 shows there is an optimum mission duration to reduce the total cost. But because the difference in total cost due to the mission duration is not large, the lower power requirement is preferable for the spacecraft.

There is another issue of redundancy. As deorbit manoeuvre will continue through a crowded altitude region, the system must have the appropriate reliability. This study does not consider EP entailing a single thruster or the clustering of small thrusters. From the standpoint of reliability, the clustering of small thrusters seems better, although small thrusters have a shorter operation life. Therefore, another trade-off study is needed to address this issue.

Sun-lit only operation of EP could further reduce the spacecraft EPS mass, and this concept will be examined in the future.

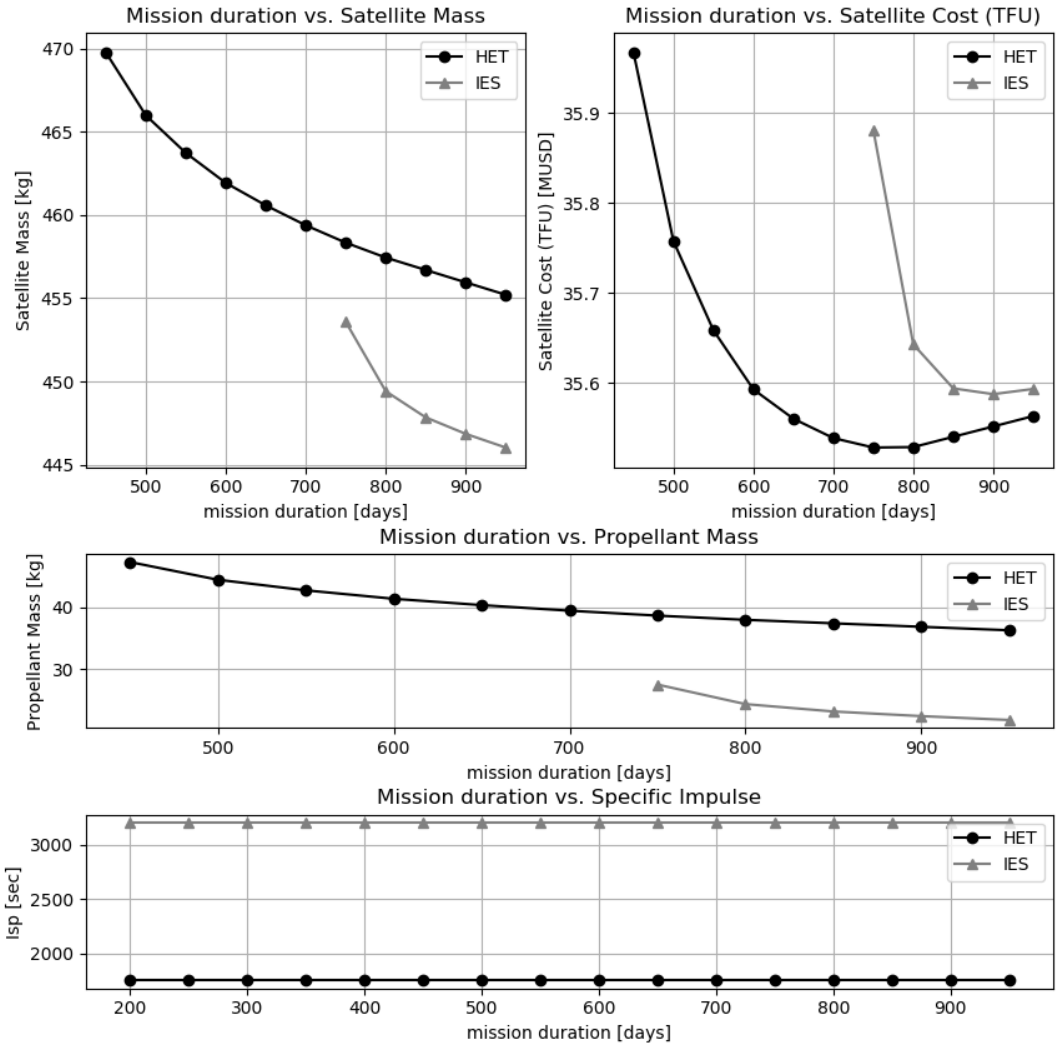

Fig. 5-2 Satellite mass and cost vs. mission duration, and Propellant mass vs. mission duration (EP power is 600 W fixed case)

5 CONCLUSIONS

This study introduced the novel concept of ADR utilizing electric propulsion and air drag. A relatively small spacecraft (of about 500 kg) can use EP as the primary propulsion for descending debris objects with the assist of air drag.

A higher thrust-to-power ratio is preferable for this concept, and the Hall effect thruster (HET) is better suited than the ion engine system (IES). The minimum-dv reentry should be studied in order to apply this concept in an actual ADR spacecraft system.

6 REFERENCES

1. Liou, J.-C., "An active debris removal parametric study for LEO environment remediation," *Adv. Space Res*., **47**, 2011, 1865-1876.
2. Bonnal, C., "Active debris removal: Recent progress and current trends," *Acta Astronautica*, **85**, 2013, 51–60.
3. Wiedemann, C. et al., "The Cost-Effectiveness of Post-Mission Disposal Manoeuvres," IAC-15.A6.8.4, 66th International Astronautical Congress, Jerusalem, Israel, 12-16 October 2015.
4. Yamamoto, T. et al., "Cost Analysis of Active Debris Removal Scenarios and System Architectures," 7th European Conference on Space Debris, 18-21 April 2017; Darmstadt, Germany.
5. Ohkawa, Y. et al., "Electrodynamic Tether propulsion for Orbital Debris Deorbit," *J. Space Tech. Sci*., **26**, 2012, 33-46, https://doi.org/10.11230/jsts.26.1_33.
6. Kawamoto, S. et al., "Current Status of Research and Development on Active Debris Removal at JAXA," 7th European Conference on Space Debris, Darmstadt, Germany, 18-21 April 2017.
7. Ohkawa, Y. et al., "Review of KITE - Electrodynamic Tether Experiment on HTV-6," 6th International Conference on Tethers in Space, Madrid, Spain, 12-14 June 2019.
8. Corey, R. L. et al., "Electric Propulsion at Space Systems/Loral," IEPC-2009-270, 31st International Electric Propulsion Conference, Michigan, USA, 20-24 September 2009.
9. Nishiyama, K. et al., "Development and Testing of the Hayabusa-2 Ion Engine System," IEPC-2015-333, 34th International Electric Propulsion Conference, Kobe-Hyogo, Japan, 4-10 July 2015.
10. Bacon, J. B., "Minimum dV for Targeted Spacecraft Disposal," 7th European Conference on Space Debris, 18-21 April 2017, Darmstadt, Germany.
11. Pollard, J. E., "Simplified Approach for Assessment of Low-Thrust Elliptical Orbit Transfers," 25th International Electric Propulsion Conference, Cleveland, OH, 1997, pp. 97-160.
12. Picone, J. M. et al., "NRLMSISE-00 empirical model of the atmosphere: Statistical comparisons and scientific issues," *J. of Geophysical Research: Space Physics*, **107** (A12), 1468. doi:10.1029/2002JA009430.
13. Hughes, S. P., "General Mission Analysis Tool (GMAT)," (2016).
14. Werz, J. R. et al., "Space Mission Analysis and Design," Third Edition, (1999).
15. Ozaki, T. et al., "In Orbit Operation of 20mN Class Xenon Ion Engine for ETS-VIII," IEPC-2007-084, 30th International Electric Propulsion Conference, Florence, Italy, 17-20 September 2007.
16. Monheiser, J. et al., "Development and Testing of a Low-Power Hall Thruster System," in Micropropulsion for Small Spacecraft, M. Micci and A. Ketsdever (eds.), AIAA Progress Series in Astronautics and Aeronautics, Reston, VA., Vol. 187, 2000, pp. 244-269.
17. Szabo, J. J. et al., "High Throughput 600 Watt Hall Effect Thruster for Space Exploration," 52nd AIAA/SAE/ASEE Joint Propulsion Conference, 25-27 July 2016, UT, doi:10.2514/6.2016-4830.
18. Bourguignon, E. et al., "Power Processing Unit Activities at Thales Alenia Space Belgium (ETCA)," IEPC-2015-224, 34th International Electric Propulsion Conference, Kobe-Hyogo, Japan, 4-10 July 2015.
19. Gollor, M. et al., "Power Processing Units - Activities in Europe 2015," IEPC-2015-225, 34th International Electric Propulsion Conference, Kobe-Hyogo, Japan, 4-10 July 2015.