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SOLAR THERMAL PROPULSION FOR MICROSATELLITES END-OF-LIFE DE-ORBITING

Hironori Sahara
Japan Society for the Promotion of Science (JSPS)
NAL, 7-44-1, Jindaiji-Higashi-Machi, Chofu, Tokyo, 182-8522, Japan
sahamail@nal.go.jp
Morio Shimizu
National Aerospace Laboratory of Japan (NAL)
7-44-1, Jindaiji-higashimachi, Chofu, Tokyo, 182-8522, Japan
Keitaro Osa, Kai Watanabe, Yoshihiro Nakamura
Hosei University
3-7-2, Kajino-Cho, Koganei, Tokyo, 184-8584, Japan

Abstract

This paper shows solar thermal propulsion (STP) for satellite end-of-life de-orbiting investigation. It includes successful experimental results of about 4 mm and 10 mm cavity diameter windowless type of thrusters made of stainless steel, single crystal molybdenum or tungsten (SC-Mo, SC-W) with W-CVD coating. The propellant in the thrusters is heated up to about 2,300 K at the highest (800 s of Isp for hydrogen propellant) by concentrated solar energy. Also, it presents ultra-light weight solar concentrators of 640 mm in diameter and a plan of space proven test of STP for a Japanese 50 kg class microsatellite as end-of-life de-orbiting propulsion system from GTO or a 800 km altitude circular orbit.

1. Introduction

One potentially attractive propulsion concept offering significant payload gains for orbit transfer from LEO to higher orbits is thermal propulsion using light gas as propellant. Solar Thermal Propulsion (STP) is a typical thermal propulsion with high Isp (500-1,000 s) in an appropriate thrust magnitude range, and provides possibly much less space pollution than conventional chemical propulsion, so that USAF or NASA have devoted to STP development for doubling the GEO mission payload capability [1-3].

The authors have investigated STP as one of the most promising propulsion systems for upper stages or orbit transfer vehicles (OTV), and started to prepare a space proven test for a Japanese 50 kg class microsatellite, Micro-Lab-Sat, as end-of life de-orbiting propulsion.

2. Solar Thermal Thrusters in NAL

2.1 Small SC-Mo Thruster as the Targeted High Isp Thruster

In the late 1980's, the National Institute for Materials Science of Japan (NIMS), et al. succeeded in establishing a new technology to develop commercially available single crystal molybdenum (SC-Mo) and tungsten (SC-W) from hot-rolled sheet doped with a certain amount of CaO and/or MgO by means of the intentional secondary recrystallization [4,5]. We fabricated and tested STP thrusters from 10 to 50 mm cavity diameter of the windowless type [6-9]. The small thruster (the first SC-Mo thruster in NAL) had its outer (maximum) and cavity diameter of about 20 and 10 mm, respectively. The Mo/Ru highest temperature brazing (its melting point is about 2,300 K, much higher than other brazing materials) and tungsten chemical vapor deposition (W-CVD) successfully prevented the propellant gas leakage from the joint portions of the thruster. Both Mo/Ru and W-CVD were stable under high temperature N_2, H_2 or He atmospheres. Carbon felt surrounded the thruster for thermal insulation. The thruster was set in a vacuum chamber (Fig. 1) that was also installed on the solar concentrator, and then tested.

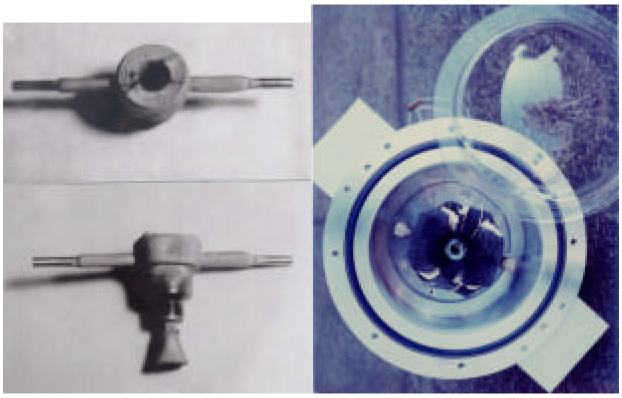

Fig. 1 Small STP Thruster Made of SC-Mo (Cavity size of 10 mm in Diameter) (left) and Set in the Vacuum Chamber, in which small STP was Surrounded with Carbon Felt (right)

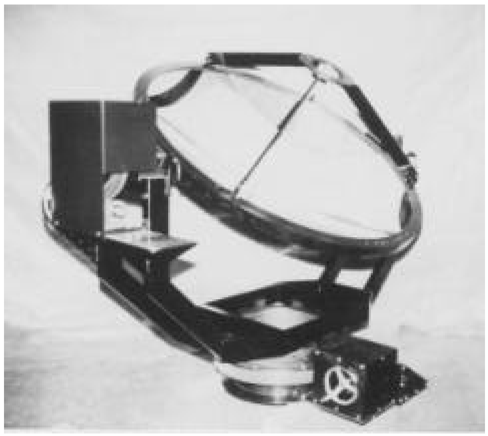

Fig. 2 1.6 m Diameter Precise Solar Concentrator of NAL with Solar Image of 15 mm in Diameter

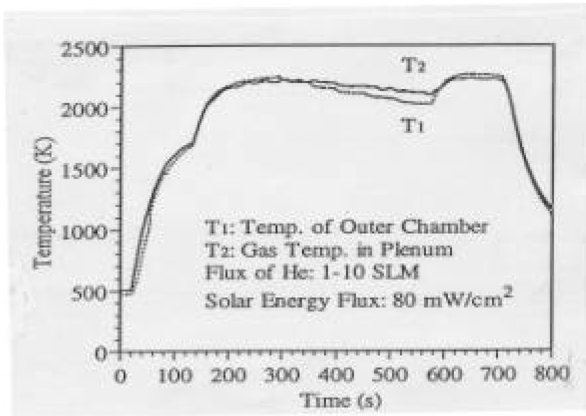

Fig. 3 Typical Results of Small Solar Thruster Test with He Propellant

The diameter of the precise paraboloidal concentrator (Fig. 2) of NAL is 1.6 m and its solar image diameter is ideally 7 mm (really about 15 mm) that is suitable for the small STP thruster. The concentrated solar light heated nitrogen or helium propellant up to about 2,300 K (Fig. 3), equivalent to about 800 s of Isp for hydrogen propellant. These are the champion data of NAL and the target of the mini thrusters mentioned later. Furthermore, the STP thruster operates in space of much higher vacuum condition, the higher thermal insulation and higher thermal efficiency. In addition, the solar light intensity in space is about twice as that on the earth. These can double or even triple the propellant flow rate (thrust) at the same Isp (800 s) as that of the tests.

2.2 Mini SC-Mo and SC-W Thruster as the Ultimate High Isp Thruster

For the highest Isp of 1,000 s at higher propellant temperature than 3,000 K, our group fabricated mini thrusters (8 mm in outer diameter and 4 mm in cavity diameter) of not only SC-Mo but also single crystal tungsten (SC-W, Fig. 4) on trial for micro/nanosatellites. Figure 5 shows the preliminary experimental results that the propellant gas temperature Tg reached about 2,000K at 1.0 SLM propellant flow rate, corresponding to 750 s Isp for H_2. As the flow rate increases, Tg decreases inversely.

The melting point of W is 3,700 K, so that the propellant gas temperature in the SC-W thruster can be proof against over 3,400 K at high reliability. SC-W has no problem with high temperature hydrogen atmosphere up to the melting point. Therefore, the SC-W thruster could have worked over 3,400 K as a 950 to 1,000 s-Isp class STP thruster if an appropriate solar concentrator had heated the thruster.

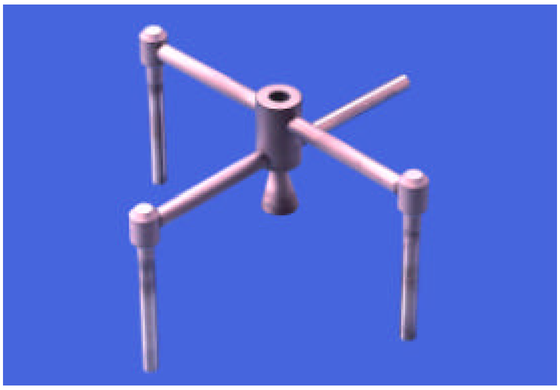

Fig. 4 Mini Thruster with Cavity of 4 mm in Cavity and 8 mm in Outer Diameter, Made of SC-W

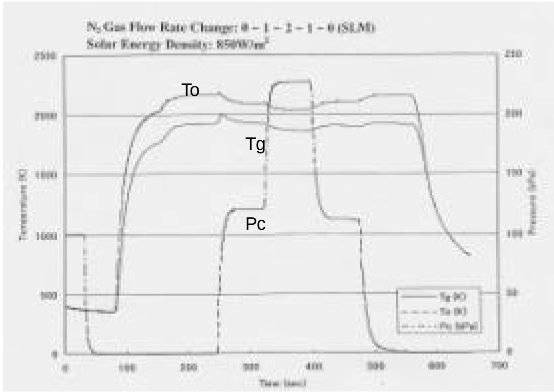

Fig. 5 Typical Results of Mini Thruster with 1.05 m in Diameter Solar Concentrator

2.3 Mini-Thruster for a Japanese 50kg Class Microsatellite

A preliminary study of NAL started for a STP space proven test using a Japanese 50 kg class microsatellite, Micro-Lab-Sat. The required thruster's outer diameter and cavity diameter are only 5 and 3 to 4 mm, respectively, because the suitable STP thruster diameter is about one hundredth of the solar concentrator one. The satellite dimensions are 50 by 50 by 50 cm cubic shape. We plan to use a one-piece of solar concentrator whose dimensions are less than 50 cm (really 43 cm) in diameter because of difficulty in using a segmented type of solar concentrator in such a micro satellite.

Figures 6 and 7 show the two fabricated mini-thrusters. One is a preliminary thruster of stainless steel (SUS 316) and the other is of SC-Mo. The W-CVD coating sealed the connection portions of the SC-Mo thruster to prevent the propellant from leaking, and the molybdenum wall of the thruster from vaporizing under high temperature (over 2,500 K) and high vacuum condition in space.

Fig. 6 Mini-Thruster Made of Stainless Steel SUS 316 with Outer Dia. of 5 mm and Dia. of 4 mm

Fig. 7 Mini-Thruster Made of SC-Mo with Outer Dia of 5 mm and Cavity Dia. of 3 mm

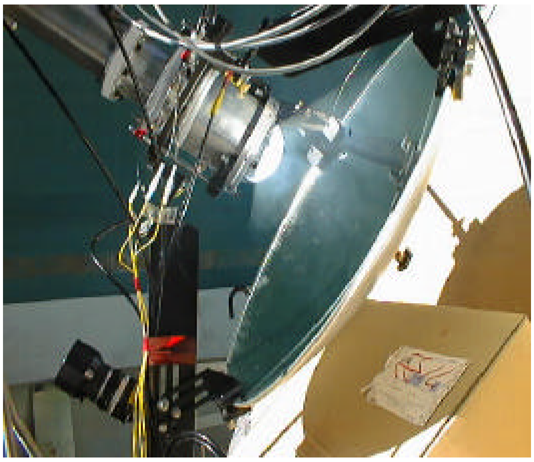

Fig. 8 Vacuum Chamber and 640 mm Diameter Solar Concentrator

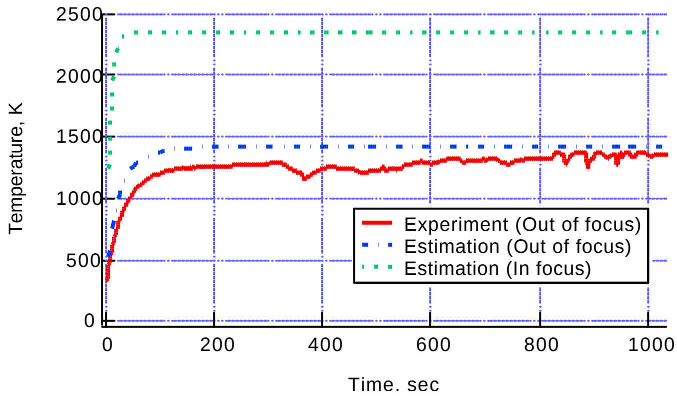

Fig. 9 Solar Heating Test Results together with Thermal Analysis with ANSYS

Figures 8 and 9 depict the preliminary test situation and results of the SUS thruster together with a thermal analysis using ANSYS. The outer thruster wall temperature rose at only 1,400 K with a 640 mm diameter glass paraboloidal mirror. Its focal length is 250mm, so that the concentrated solar image diameter is ideally 3 mm and really 5 mm, equivalent to the thruster outer diameter. In fact, the thruster was intentionally out of focus by 7-10 mm, and resulting solar image diameter is 20-30 mm for not over-heating the thruster. If in focus, the thruster temperature would have risen up to ANSYS-estimated 2,350 K, over the melting point of SUS 316 (about 1,700 K). The test results are shown in Table 1. In the near future we will perform the in-focus tests with the SC-Mo thruster, targeting 2,300 K.

An in-house ultra light film solar concentrator was fabricated, using the above-mentioned glass mirror as a mold. Hence, the dimensions are almost same as the mold mirror. But the concentrated solar image diameter is over 10 mm (Table 1), so that the input energy to the thruster was much lower (about 20 %) than the in-focus concentrated energy. This is the reason why the thruster temperature is about 1,300 K, much lower than the aimed temperature of 2,300 K (at the lowest, 1,700 K).

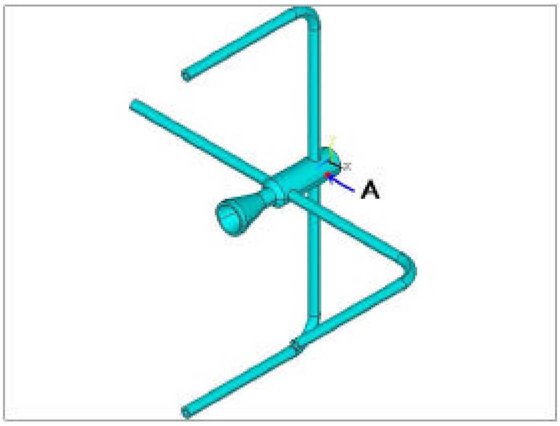

Fig. 10 Model of Thermal Analysis with ANSYS

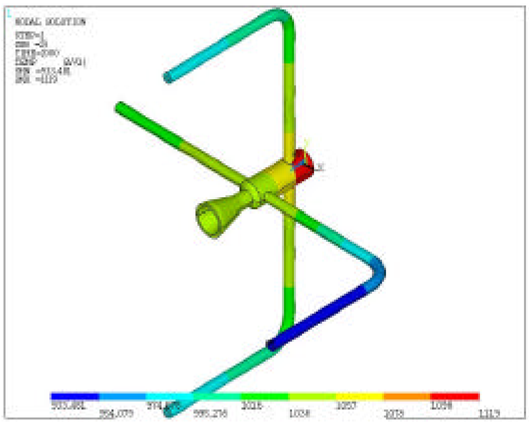

Fig.11 Temp. Distribution of Mini-Thruster

Figures 10 and 11 depict the model and the whole temperature distribution of our ANSYS thermal analysis. The saturated temperature agrees well with the test results, that are useful to the design and performance estimation in space of the thruster. Table 2 tabulates the data used in the analysis.

Table 1 Comparison between Experimental and Analytical Plenum Chamber Temperature

	Experimental Results		Analytical Results (ANSYS)	
	Glass Mirror Reflection: 90%	Film Mirror Reflection: 80%	Glass Mirror Reflection: 90%	Film Mirror Reflection: 80%
Solar Energy Density [W/m]	900	870	900	870
Plenum Temp. [K]	1,379	1,276	1,392	1,326
Solar Image Dia. [mm]	20	11	20	11

Table 2 Data used in Thermal Analysis

Thermal conductivity [cal/cm/s/K]	0.039
Mass Density [g/cm^3]	8.03
Specific Heat [cal/g/K]	0.12
Coefficient of Heat Transfer [W/m^{2}K]	4
Boltzmann's Constant [W/m^{2}K^4]	5.67e-8

3. Ultra-Light Weight Solar Concentrator

The other most important subsystem is an ultra-light weight precise solar concentrator that heats up the thruster and propellant over 2,000 K for 750 s of Isp for hydrogen. Usually, a honeycomb sandwich structure supported the reflection surface of conventional rigid lightweight concentrators that required 5 to 10 kg/m^2 for enough rigidity. But it is too heavy for space use, especially for STP.

The group of USAF/Boeing tried to adopt inflatable solar concentrator made of aluminized and transparent films for a solar orbit transfer vehicle (SOTV). The inflatable structure is ultra-light weight, but it is very vulnerable to space debris impact. The debris over 0.1 mm in diameter can easily penetrate the films and the inflation gas leaks from the penetrated holes. This requires additional gas supply that means Isp decrease. And more seriously, the gas leak generates some amount of thrust that causes attitude disturbance, so that additional propellant ejection is necessary to compensate the disturbance.

In NAL, a sheet of aluminized paraboloidal film concentrator was formed by plastic deformation due to stress relaxation under high temperature condition. Using a paraboloidal glass mirror as a mold, aluminized film was set on the glass mirror and evacuated between the film and mirror. After heating at about 420 K for polyester film (about 550 K for polyimide) in a temperature controlled chamber, the air between the film and glass mirror was evacuated so that the film was fit to the mold and formed into the paraboloidal solar concentrator for five hours to five days by plastic deformation due to stress relaxation.

This method fabricated several axisymmetric film mirrors successfully. One is 640 mm in diameter and 250 mm in nominal focal length, quasi-paraboloidal polyester film mirror (Fig. 12). But the paraboloidal shape accuracy is not enough. Our real solar concentration tests measured the solar images of these glass and film mirrors, taking photos of the solar images on a target metal plate. The diameter of the solar images is defined as the FWHM (full with at half maximum) of the concentrated solar light intensity distribution (Fig. 13). The solar image diameters of the film mirror are about double the mold glass mirror and about five times the ideal (theoretical) solar image diameter (about 2.5mm) (Table 3). The solar concentration ratio of the film mirror results in about 3,000. A solar concentration ratio of at least 5,000 is necessary and will be available in the near future in NAL to heat the propellant in the thruster up to 1,750 K, about 700 s of Isp for H_2 and 230 s for water propellant.

Fig. 12 Glass Mirror as the Mold (640 mm in Dia. and 250 mm in Focal Length) and Film Mirror

Table 3 Solar Image Dias. and Focal Lengths of 640 mm in Dia. Glass and Film Concentrators

		Solar Image Dia.(mm)	Focal Length (mm)
Glass Mirror	Vertical	4.6 ± 0.001	245.0
	Horizontal	5.4 ± 0.001	
Film Mirror	Vertical	11.1 ± 0.003	255.2
	Horizontal	11.5 ± 0.005	

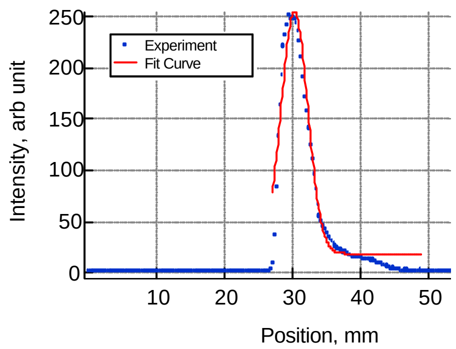

Fig. 13 Solar Light Intensity Distribution of 640 mm Dia. Glass Concentrator

Table 4 Specs. of Micro-Lab-Sat #2 Mother Satellite

Item	Specs.
Dimensions	50 by 50 by 30 cm
Weight	30 kg
Available Electric Power	150 W
Attitude Control	3 Axes, 0.1 Degree
Orbit Altitude	800 kg, Circular
Life Time	3 Months

4. STP for End-of-Life De-Orbiting

The STP upper stage or STP-OTV will be very useful to the Japanese launch vehicle capability augmentation [8,9]. Also, the STP-OTV is useful to de-orbiting satellites after the end-of-life. It is important to perform STP space proven tests for its development. But, as a matter of fact, there are few opportunities for any space proven tests. Fortunately, NASDA has developed a 50 kg class micro-satellite, Micro-Lab-Sat, for easy, inexpensive and more space proven tests opportunities. Last December, the H-IIA #4 successfully launched Micro-Lab-Sat 1 as one of piggyback payloads. Therefore, NAL started to prepare a micro STP space proven test on the Micro-Lab-Sat series. We plan to perform the micro satellite end-of-life de-orbiting propulsion as a useful mission for the microsatellite and STP space proven test. This requires not only a mini-STP thruster made of SC-Mo of 5 mm in diameter, but a small solar concentrator of 430 or 640 mm in diameter. Table 4 tabulates the specifications of the Micro-Lab-Sat #2 that has four 5 kg class daughter nanosatellites, so that the mother satellite weighs only 30kg and its dimensions are 50 by 50 by 30 cm.

When the all missions of the mother satellite have completed, the micro STP starts de-orbiting. STP decelerates the satellite at an appropriate point of the circular orbit, and then repeats the deceleration at the same point, because the point has become the apogee. Accordingly, the repeating of the deceleration at the apogee lower the perigee attitude, and finally the satellite dives into the earth atmosphere and reaches at 100 km of altitude. At this altitude, air drug is enough large to decelerate the satellite to lower the perigee, and then satellite reaches the 100 km circular orbit and burns out within one circulation. Therefore, the stage of 100 km altitude is defined as the satellite abandonment. Table 5 shows the assumptions of the de-orbiting estimation and the results. The required water propellant mass and period for the satellite abandonment from an 800 km circular orbit are about 3.3 kg and 25 days, respectively, if STP operates to decelerate the satellite during plus/minus 20 degrees from before to after the apogee passing. The case of GTO requires the propellant mass of only about 0.7 kg and the period less than one day, much easier than 800 km circular orbit. Figure 14 shows the orbit change preliminary simulations from the 800 km circular orbit and GTO to the 100 km abandonment perigee altitude.

Table 5 Performances of Micro STP for the 30kg Mother Satellite of Micro-Lab-Sat De-orbiting from 800 km Circular Orbit and GTO to 100 km Perigee Altitude

Original Orbit	800 km Circular	800 km Circular	GTO (Perigee: 250 km)
Propellant / Temp./ Isp	Water /1,500 K / 200 s	Water / 1,500 K / 200 s	Water / 1,500 K / 200 s
Concentrator Dia.	430 mm	640 mm	430 mm
Solar Energy Density	1.4 kW/m^2	1.4 kW/m^2	1.4 kW/m^2
Mirror Reflection	80%	80 %	80 %
Thermal Efficiency	70%	70 %	70 %
Effective Input Energy	0.48 kJ	1.05 kJ	0.48 kJ
Max. Prop. Flow Rate	1.0 g/min	2.2 g/min	1.0 g/min
Required Prop. Mass	3.3 kg	3.3 kg	0.68 kg
Required Period	25 days	11 days	21 hours

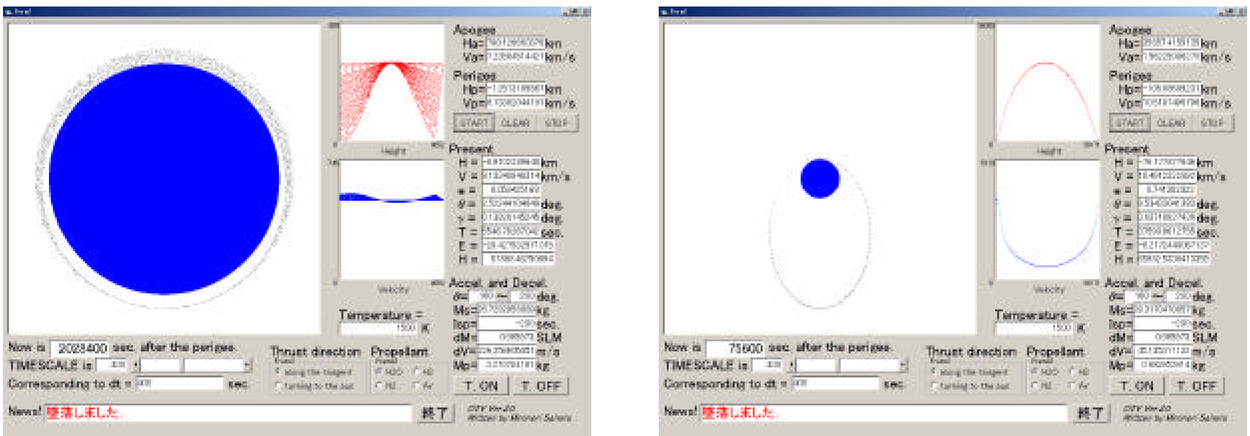

Fig. 14 De-orbiting Simulation of Micro-Lab-Sat Apogee Deceleration with Micro STP from 800 km Circular Orbit and GTO to 100 km Perigee Altitude

5. Considerations

The micro satellite de-orbiting from an 800km circular orbit requires 3.3 kg water propellant (11% of total satellite mass), as mentioned above. It is not so light load. Using not water but H_2, the Isp is triple, but it is difficult to develop such a small and light liquid H_2 tank, so that water or NH_3 propellant selection is realistic. But how about GTO, another popular orbit where space debris population is large? In fact, it requires only about 0.7 kg water propellant (about 2 % of satellite mass), so that H_2 can be realistic using a light small high-pressure tank. As a space proven test H_2 is really desirable for future OTV development.

The all above-mentioned STP thrusters are of the axisymmetric type that is suitable for only apogee or perigee propulsion. If orbit transfers need the Hohmann type procedure, they require the opposed type STP thruster and the appropriate opposed type off-axis paraboloidal solar concentrators. We have preliminarily tested these types of thrusters made of SC-W and ultra-light weight concentrators successfully [10].

6. Conclusion

NAL has preliminarily studied a micro STP for the 50 kg class Japanese Micro-Lab-Sat end-of-life de-orbiting propulsion as space proven tests successfully. The thruster was made of Single crystal molybdenum NIMS patented in Japan and USA. The expected propellant temperature is over 1,500 K, 200 s of Isp for water propellant. The preliminary work gave promising results of required propellant and period for satellite end-of life de-orbiting from 800 km circular orbit and GTO to 100km perigee altitude.

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