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An Engineering-model PPT Using Self-inductor Coupling Element

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Keywords: PPT, Electric Propulsion, Stationkeeping, Small Satellite

Abstract

Pulsed plasma thruster is one of the best propulsion options for small satellites, which is capable of performing a variety of propulsive tasks from stationkeeping to precise attitude control. More recently, following the growing interest for smaller satellites, low power μPPT has been accounted as an ideal option for micro- and nano-satellites (e.g. CubeSats) to perform both primary and attitude control propulsive tasks. It offers the many benefits of simplicity, high specific impulse, variable thrust level, fine impulse bits, and high reliability as a small size thruster with low mass and power which benefits from a solid propellant, i.e. PTFE, compatible with vacuum condition and space adverse temperature gradients.

Following the design and development of a laboratory PPT in 2009, a compact engineering model was designed and developed with total mass of about 1.6 kg including power processing unit, capacitor, and case. The PPT characteristics are listed in Table 1. The PPT employed a self-inductor coupling element to connect the PPT cathode to ignitor plug cathode. Coupling elements were already utilized as a method to reduce ignitor plug electrode erosion which is caused by arc attachment to electrodes. However, the results of the current research suggested that coupling elements may have some influence on PPT discharge behavior and performance. This proposes that the method can be used to improve PPT performance. However, the way it can affect the performance has not been investigated and is not known yet. Therefore, further research is necessary to clarify it.

The PPT employs a 35 μF metallized-film capacitor charged at 700 V. The PPT successfully operated near 20,000 pulses but experienced charring and irregular pulsing after that as a result of deposition of carbon on propellant surface. It subsequently prevented capacitor charging by shorting the PPT electrodes and intermittent arcing occurred. Figure 2 indicates collection of carbon deposits on PTFE surface. The engineering-model produced an impulse bit of 80 μNs at 8.5 J of energy and specific impulse of 1000 s and thrust efficiency of 4.6 %.

Introduction

Pulsed plasma thruster is one of the earliest electric propulsion technologies developed with a successful flight history starting over four decades ago and culminating in the successful TIP/NOVA flights of the 1980's. Another recent focus has been put on μPPTs as an ideal option for micro- and nano-satellites (e.g. CubeSats) to perform both primary and attitude control propulsion tasks. Basically, an arc in PPT is used to ablate the propellant, typically PTFE, and accelerate it to high velocities. The thruster itself consists of a propellant bar positioned between two electrodes which are connected to an energy storage capacitor charged with high voltage to a desirable energy coming from the power processing unit which uses spacecraft bus power. The main discharge of capacitor is initiated by a small plasma puff created by an igniter plug excited by the PPU. The details of PPT physics, configurations, and the effect of essential parameters on PPT discharge and performance were studied and reviewed elsewhere.[1-3) PPTs are among the most promising propulsion systems to perform various propulsive tasks on small satellites. Some of the many advantages they offer compared to other propulsion systems are simplicity, small size, low mass, low power, solid propellant, high reliability, high specific impulse and variable thrust level. Therefore, to perform an orbital maneuver of drag compensation for a small satellite with a mass of less than 60 kg to maintain its repeat ground track orbit throughout its life time, pulsed plasma thruster was chosen, designed and developed. The satellite is planned to have a circular orbit at an altitude near 760 km, where the orbit decay caused mostly by aerodynamic drag can be compensated by the order of thrust that PPT is capable of producing. The paper briefly describes the design and development of an engineering model pulsed plasma thruster of nominal 8.5 W power level with a specific impulse level of 1000 seconds and impulse bit of 80 μN-s. The PPT uses a 35 μF, 800 V metalized-film capacitor, a semiconductor igniter plug, copper electrodes, and an insulator structure with a total mass of about 1.5 kg including power processing unit (PPU). A self-inductor coupling element is employed to connect the PPT cathode to ignitor plug cathode. In addition to the PPT design and development, the planned satellite mission will be briefly discussed.

PPT Mission Capability

A quite wide range of propulsive tasks are proposed for PPTs with main focus on low thrust missions requiring fine impulse bits. Generally, it is desirable for the PPT to perform multiple functions in a satellite in order to increase its appeal as an alternative for conventional propulsion systems.[4)]

PPTs offer small mass while they can produce high level of total impulse which make them an ideal option where high ratio of total impulse-to-system mass is desired.

One of the most extensively analyzed functions for PPTs has been attitude control.[4-7)] Stationkeeping is also a satellite function which PPTs are efficiently capable of and have successful heritage in. Stationkeeping includes drag compensation, negation of orbit perturbations, formation flying, and constellation maintenance.[4,8-10)] Drag make-up, like attitude control, also is a function that requires only lowest levels of thrust for many spacecraft. NOVA required less than 1 mN of thrust for the mission. However, at altitudes below 500 km, drag make-up requires more thrust.[4)]

For stationkeeping, PPTs have long been considered a preferred alternative to low performance cold gas thrusters, or hydrazine thrusters as they require lower mass. Moreover, their high specific impulse makes the propellant mass quite insignificant for such small velocity change maneuvers. The maximum velocity change requirement for station keeping, assuming a duration of 6 to 12 months is 300 ms^{-1}, within the expected performance range of PPTs.[4,8-10)]

All things considered, the pulsed nature of the thrust produced makes PPTs ideal for propulsive maneuvers that require small impulse bits such as drag compensation, stationkeeping and attitude control. The low power requirements of PPTs compared to other electric thrusters and the simplicity of its mechanical design with relatively few moving parts presents the ideal solution to providing active on board propulsion for nano- and pico-satellites.

Therefore, to perform a series of drag compensation maneuvers to maintain the repeat ground track of a small 60-kg satellite within the band of ±5 km at an altitude of about 760 km and an orbit inclination of 55º, PPTs have been proposed and a preliminary analysis was conducted on the feasibility of their utilization on the satellite which is briefly reviewed in the present paper.

Design and Development of the PPT

Following the design and development of a laboratory benchmark pulsed plasma thruster in Iran[11)] in 2010 and successive studies[3,12)], and with the intention of developing a smaller model operating at lower power to fit a small satellite requirements; an engineering model PPT was designed and developed with total mass of about 1.5 kg working at 8.5 W and 1 Hz. Figure 1 shows a picture of the lab-PPT developed and Table 1 presents the PPT performance at various discharge energies and power inputs tested.

Table 1 The lab-PPT performance at various discharge energies.

V_o	E (J)	I_{bit} (μN-s)	I_{sp} (s)	M_{bit} (μg)	T/P	η	C_e (km/s)
750	9.84	476	200	242	48.4	5%	1.96
1000	17.5	663	366	184	37.9	7%	3.60
1250	27.3	943	525	183	34.5	9%	5.15
1500	39.3	1118	800	142	28.4	11%	7.87
1750	54	1323	1100	122	24.5	13%	10.84

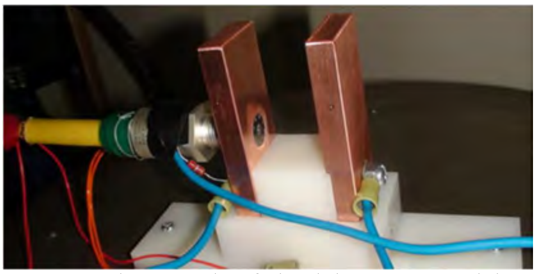

Fig. 1 A photograph of the laboratory model PPT developed.

The engineering-model PPT type is similar to the previous lab model, a breech-fed rectangular parallel-plate PPT. The electrode dimension of this model are scaled down with a factor of 2/3 compared to the earlier model[11)] to achieve the smaller size and lower mass in accordance with the needs of microsatellites. The main constraint which confined further miniaturization of the model was the size of available igniter plug at the time of development as developing a smaller igniter plug was not planned.

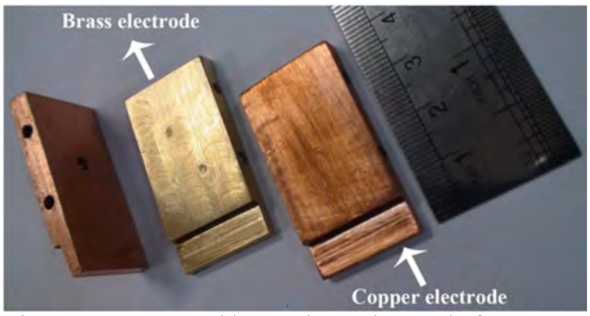

Fig. 2 Copper and brass electrodes made for PPT.

The PPT electrode dimensions are (width-length-thickness) 21 × 40 × 8 mm for cathode and 21 × 40 × 4 mm for anode. Copper and brass have been investigated for use as electrode material but copper was finally employed (Fig. 2). The interelectrode spacing is 21 mm making the propellant face area be 4 cm^2 and aspect ratio (interelectrode spacing/electrode width) equal to 1. The propellant for the PPT is PTFE as for the earlier model. Figure 3 shows a picture of the engineering model PPT.

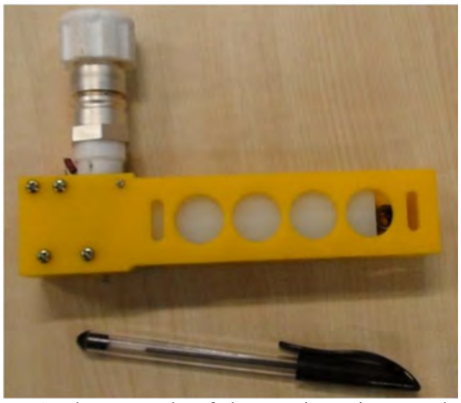

Fig. 3 A photograph of the engineering model PPT developed.

The PPT structure was chosen from temperature-resistant fluorocarbon thermoplastics and the available options were PTFE, polyether-Imide (PEI) also known as Ultem®, and polyamide-imide (PAI) also known as Torlon®. Eventually, Ultem was chosen with respect to lower mass, better mechanical characteristics[13], and flight heritage[14]. PTFE was used for the same purpose on the earlier model.[11] Figure 4 indicates Ultem®, Torlon®, and PTFE propellant holding structures developed.

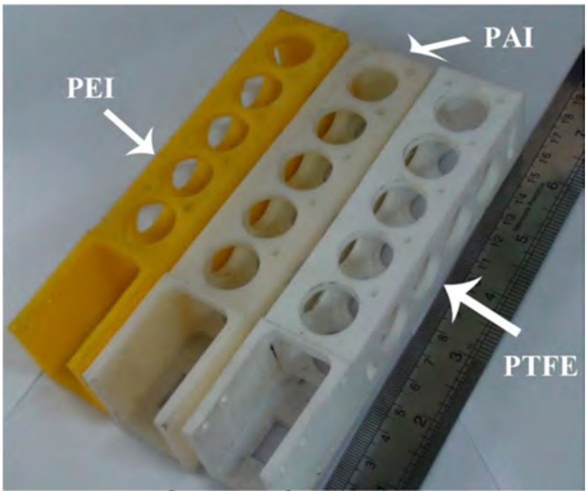

Fig. 4 Ultem®, Torlon®, and PTFE propellant holding structures for PPT.

A metalized-film, 40 µF, 800 V capacitor (Fig. 5) was initially chosen to be employed as the main capacitor for the PPT discharge although there are still doubts about its life time. However, it has accomplished to operate at 10,000 PPT discharges in vacuum chamber during the PPT functional tests but it is preferred to replace it with a mica-paper capacitor bank.

Fig. 5 The metalized-film capacitor selected for the engineering model PPT.

The PPU for this model was redesigned to supply the discharge initiation circuit with a pulse of 1400 V to fire the igniter plug and also to charge the main capacitor with 700 V from a regulated power bus at 12 V with maximum current limited to 1.5 A. Figure 6 indicates a schematic of the designed power processing unit with two outputs, output 1 for main capacitor and output 2 for igniter plug. The PPU efficiency is estimated to be about 80%. The PPU was designed in two configurations when both have the same schematic but different position for parts on the boards in the hope of smaller size; one on a one-floor board and the other on a two-floor board (Fig. 7) but the one-floor design was finalized and utilized on the final model. To get a feedback that the PPU was working properly, two signals were recorded. The first one was the voltage sent to charge the capacitor and the second one was the voltage sent to the step-up pulse transformer which steps up the voltage by a factor of 2:1 and sends a 1400 V pulse to igniter plug.

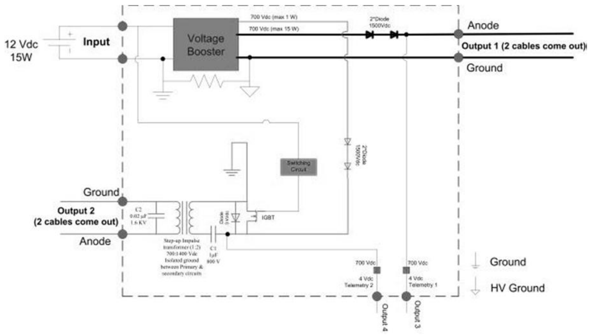

Fig. 6 Schematic of the power processing unit.

The working frequency of the PPT is 1 Hz and PPU is designed to charge the capacitor within 950 msec. The charging time has been extended within the whole available time to prevent any peak current above 1.5 A to keep the PPU compatible with the requirements of a sample microsatellite. Fig. 8 shows the PPT capacitor charging history. Experiments have shown that the period of time the capacitor is charged has great influence on its life time.

Fig. 7 PPU designed on a two-floor board.

All in all, considering the PPU efficiency, the whole system can operate at less than 11 W at 1 Hz. The PPT has been successfully tested for about 20,000 pulses and has passed several environmental tests including thermal vacuum, vacuum cycling, sinusoidal and vibrational tests. Figure 9 indicates the compact engineering model PPT, together with its PPU inside the case and capacitor mounted on the case. Several logs have been built on the case to be mounted on a possible satellite. Figure 10 shows a picture of the PPT during the test inside the vacuum chamber.

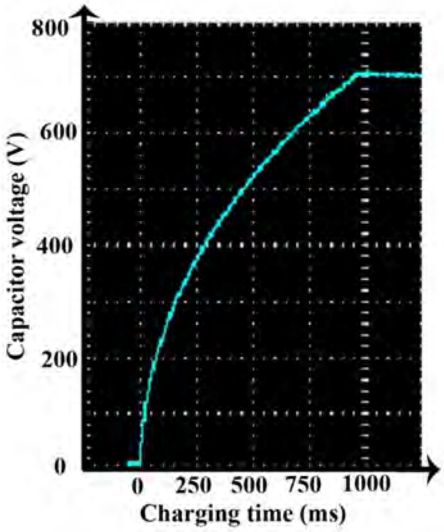

Fig. 8 PPT capacitor charging history.

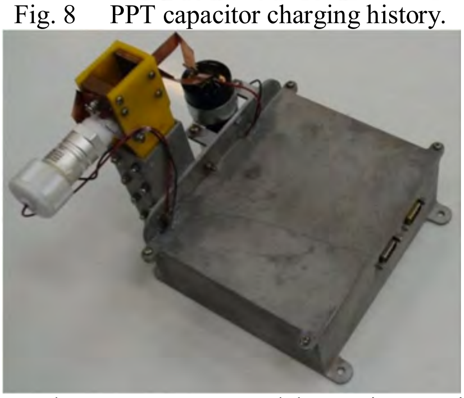

Fig. 9 The compact Eng-model PPT, its PPU inside and capacitor mounted on the case.

Table 2 presents the characteristics of the engineering model PPT developed. During the PPT life cycle tests, it was observed that after about 20,000 pulses, charring and irregular pulsing occurred as a result of deposition of carbon on propellant surface which is required to be prevented by optimizing the design. It subsequently prevented capacitor charging by shorting the PPT electrodes and intermittent arcing occurred. Figure 11 indicates collection of carbon deposits on PTFE surface.

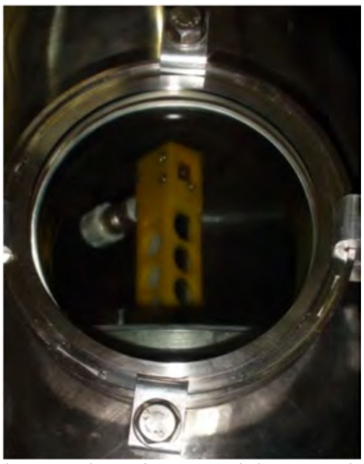

Fig. 10 The engineering model PPT in vacuum chamber under life cycle tests.

Table 2 Engineering model PPT characteristics.

Cathode dimensions	21 × 40 × 8 mm
Anode dimensions	21 × 40 × 4 mm
Electrodes' material	copper
Electrode spacing	21 mm
Aspect ratio (h/w)	1
Propellant face	4 cm2
Propellant type	PTFE
Ignition voltage	1500 V
Igniter plug mass	75 gr
Circuit switching	IGBT
Capacitor	metalized-film, 40 µF, 800 V
Capacitor mass	90 gr
PPU mass	< 400 gr
Input voltage	12 V
Power	8.5 W
PPU efficiency	80 %
Frequency	1 Hz
Total mass	~ 1.5 kg
Strip lines	Copper sheet

Deposition on ignitor plug electrodes can be another restriction for PPT life time. Coupling elements which connect ignitor plug cathode to PPT cathode were proposed[15] in PPTs to prevent this. Coupling elements can be resistive or inductive. A self-inductor coupling element was used in our PPT for the same reason, however, initial results proved that it can have influence on PPT performance by affecting the discharge behavior as well.

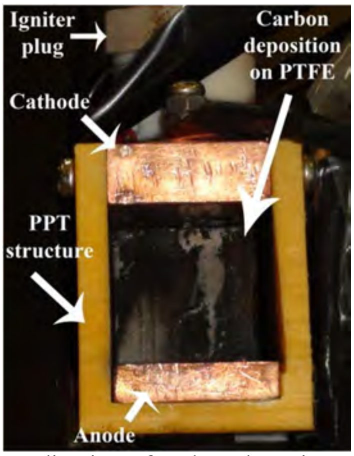

Fig. 11 Indication of carbon deposits on PTFE.

The performance data of the thruster where investigated in PPT functional tests and are presented Table3.

Table 3 Eng-PPT performance parameters.

V_o (V)	E (J)	I_{bit} (µN-s)	I_{sp} (s)	M_{bit} (µg)	T/P	η (%)	C_e (km/s)
700	8.5	80	1000	8.1	9.3	4.6	9.87

The input data for our orbital analysis (next section) were taken from our PPT performance in three different modes. The first two were two modes of the earlier PPT at 9.84 and 25 W while the third one is the operating mode of the engineering-model PPT at 8.5 W. All cases were analyzed for the orbital analysis to perform a drag compensation maneuver to maintain the repeat ground track orbit of a 60-kg sample satellite with an effective area of 0.55 m^2 within the band of ±5 km, considered for imaging purposes. It requires a maneuver with a ΔV (velocity change) of 0.04 m/s each 138 days to raise the satellite semi-major axis (a) for 69 m per maneuver. Table 4 proposes the results of this analysis and presents the number of maneuvers per life time, total ΔV, propellant mass per maneuver (Mp), total propellant mass, time of each maneuver (Tm), and time of maneuver-to- orbit time ratio for all the three cases for different life times of 1, 3, 5, and 7 years of operational life in orbit.

Satellite Characteristics and Mission

The sample satellite is a remote sensing satellite which is imagined to be designed for the Earth observation mission. The satellite final mass is supposed to be about 60 kg, capable of generating 100 W of power using its body-mounted solar panels.
The satellite is designed to be at a repeat ground track orbit with a band of ±5 km for imaging missions. As a result of the fact that the satellite will be launched from Iran into an orbit with an inclination of 55°, thus its altitude needs to be approximately 760.7 km to have a repeat ground track orbit.
Based on orbital analysis conducted by the commercial software STK, the satellite needs to be launched to an orbit with an altitude 24 m (semi-major axes (a) =7138.844 km) above the target orbit which is 760.7 km (a[1] = 7138.82 km) to automatically initiate a repeat ground track cycle in orbit.
Based on the STK results, the satellite with an effective area of 0.55 m^2 and mass of 60 kg will decay about 0.5 m per day at orbit altitude of 760.7 km, considering a launch date in 2014. Therefore, to keep the ground track error within ±5 km, the first maneuver needs to be done after 118 days to raise the satellite altitude for 69 m. The required velocity change for this maneuver is about 0.04 m/s.
After the first maneuver, it needs to repeat each 138 days throughout the satellite life time to keep the satellite ground track within ±5 km. Consequently, the satellite will have to perform this maneuver 2.645 times per year and total required velocity change per year is 0.1 m/s. Moreover, the satellite will have nearly 14.4 orbits each day and each orbit will lasts about 6000 seconds.
PPT has been chosen as the satellite propulsion system to perform the maneuvers as it is within the capabilities of the thruster to perform this drag make-up maneuver (more generally a stationkeeping and orbit maintenance maneuver). Figure 12 depicts a proposed configuration for 2 PPTs on the satellite.

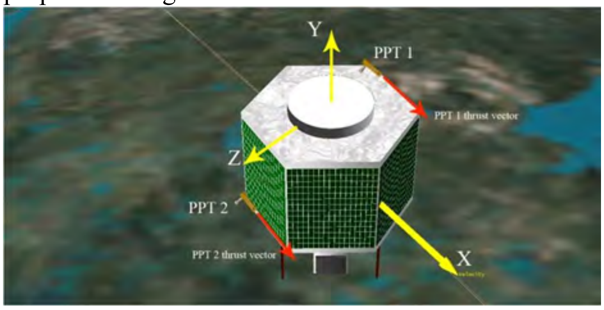

Fig. 12 Schematic of the satellite body frame showing PPTs proposed configuration to perform the drag make-up maneuver.

Figure 13 shows the satellite ground track in km versus its number of orbits since its launch, indicating how the satellite ground track moves within the band of ±5 km, when the satellite maneuvers are performed correctly by its thrusters (pulsed plasma thrusters) during the orbits after launch. The graphs are generated by STK software. The point that the thrusters start to fire is the peak of the curves where ground track error reaches its maximum +5 km.
Figure 14 shows the satellite altitude variations in km versus the satellite ground track in km. It indicates that when the satellite altitude decays, its ground track error reaches its maximum +5 km. At this point, the thrusters begin to operate and they raise the satellite altitude for 69 meters and this causes the satellite ground track error become its minimum -5 km. It takes another 138 days until the disturbances decay it and this cycle repeats throughout the satellite life time.

[1] Semi-major axes

Table 4 Results of the analysis on utilization of PPT for the drag make-up maneuver.

First case	**Thruster**	**I_{sp}**	**Mass shot**	**Power (W)**	**Ibit (µN-s)**	**η (%)**		
	Lab-PPT	200 s	238 µg	9.8	476	4.85		
	Life Time	**No. of maneuvers/Life**	**Total Δv**	**Mp/maneuver**	**Total Mp**	**Tm (s)**	**Tm (min)**	**Tm/Torbit**
	1 yrs	2.64	0.105 m/s	1.224	3.239	5145	85.7	0.86
	3 yrs	7.93	0.317 m/s	1.224	9.716	5145	85.7	0.86
	5 yrs	13.22	0.528 m/s	1.224	16.193	5145	85.7	0.86
	7 yrs	18.51	0.740 m/s	1.224	22.671	5145	85.7	0.86
Second case	**Thruster**	**I_{sp}**	**Mass shot**	**Power (W)**	**Ibit (µN-s)**	**η (%)**		
	Lab-PPT	500 s	180 µg	25	900	9.00		
	Life Time	**No. of maneuvers/Life**	**Total Δv**	**Mp/maneuver**	**Total Mp**	**Tm (s)**	**Tm (min)**	**Torbit/Tm**
	1 yrs	2.64	0.105 m/s	0. 49	1.295	2721	45.3	0.45
	3 yrs	7.93	0.317 m/s	0. 49	3.886	2721	45.3	0.45
	5 yrs	13.22	0.528 m/s	0. 49	6.477	2721	45.3	0.45
	7 yrs	18.51	0.740 m/s	0. 49	9.068	2721	45.3	0.45
Third Case	**Thruster**	**I_{sp}**	**Mass shot**	**Power (W)**	**Ibit (µN-s)**	**η (%)**		
	Eng-PPT	1000 s	8.1 µg	8.5	80	4.50		
	Life Time	**No. of maneuvers/Life**	**Total Δv**	**Mp/maneuver**	**Total Mp**	**Tm (s)**	**Tm (min)**	**Torbit/Tm**
	1 yrs	2.64	0.105 m/s	0. 245	0.648	30612	510.2	5.1
	3 yrs	7.93	0.317 m/s	0. 245	1.943	30612	510.2	5.1
	5 yrs	13.22	0.528 m/s	0. 245	3.239	30612	510.2	5.1
	7 yrs	18.51	0.740 m/s	0. 245	4.534	30612	510.2	5.1

Figure 15 shows the variations of the satellite semi-major axis (a) in km versus the number of orbits it revolves. It can be apparently seen how the satellite altitude decays during 138 days and the drag compensation maneuver raises the satellite altitude to maintain its ground track error within ±5 km.

The attitude control system of the satellite is supposed to employ one reaction wheel and two magnetorquers to maintain its 3-axis stabilization so that thrust vector of the PPTs can be kept aligned with the satellite velocity vector in this circular orbit. This gives the PPTs the capability to fire throughout the orbit as much as needed to in an unlimited time to perform the orbit maintenance maneuver, if not constrained by other factors like power or satellite imaging missions.

The results of the investigation to employ the PPT to perform this specific repeat ground track orbit maneuver shows that the performance data from both lab model and engineering model are capable of performing the maneuver within either a fraction of the orbit time (for cases 1 and 2) or in the order of orbit time (for case 3). Considering one year lifetime for the sample satellite with the specified maneuver previously discussed, the lab PPT at 9.8 W (case 1) will need to perform 2.64 maneuvers per life time, each taking 5145 s, consuming 1.224 g of PTFE per maneuver which makes the total propellant mass equal 3.239 g. Each maneuver for case 1 takes 0.86 times of orbit maneuver (6000 s). If the same situation is applied using the lab PPT at 25 W (case 2), the PPT will need to perform 2.64 maneuvers per life time, each taking 2721 s, consuming 0.49 g of PTFE per maneuver which makes the total propellant mass equal 1.295 g. Each maneuver for case 2 takes 0.45 times of orbit maneuver. Considering that the engineering model PPT has been utilized at 8.5 W (case 3), the PPT will need to perform 2.64 maneuvers per life time, each taking 30622 s, consuming 0.245 g of PTFE per maneuver which makes the total propellant mass equal 0.648 g. Each maneuver for case 3 takes 5.1 times of orbit maneuver. However, the design target is to develop PPTs with higher efficiency and higher thrust-to-power ratio (T/P) at lowest power to minimize the maneuver time.

Conclusion

Following a laboratory PPT, a scaled-down engineering-model PPT was designed and developed. The PPT was tested in vacuum chamber and its performance data were reported. The PPT used a coupling element which showed to have more influence on PPT performance rather than only increasing the PPT ignitor plug lifetime.

The performance data of both models of the PPT were employed to perform an orbital maneuver of drag make-up to maintain the repeat ground track orbit of a small 60-kg satellite within the band of ±5 km at an altitude of 760.7 km for imaging purposes and three cases were analyzed based on the stated results.

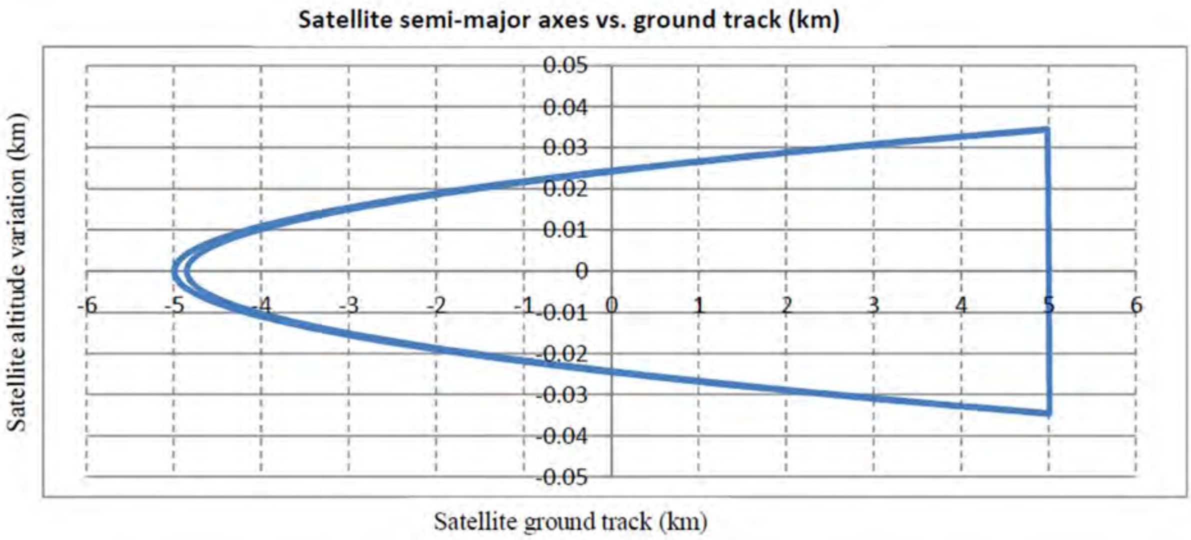

Fig. 13 Satellite altitude variations versus ground track

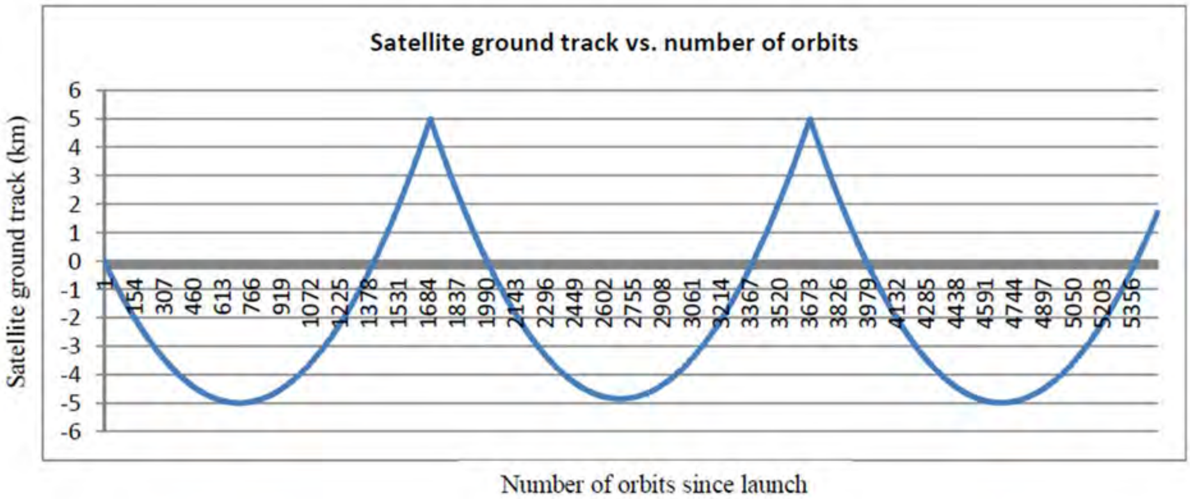

Fig. 14 Satellite ground track versus number of orbits since launch.

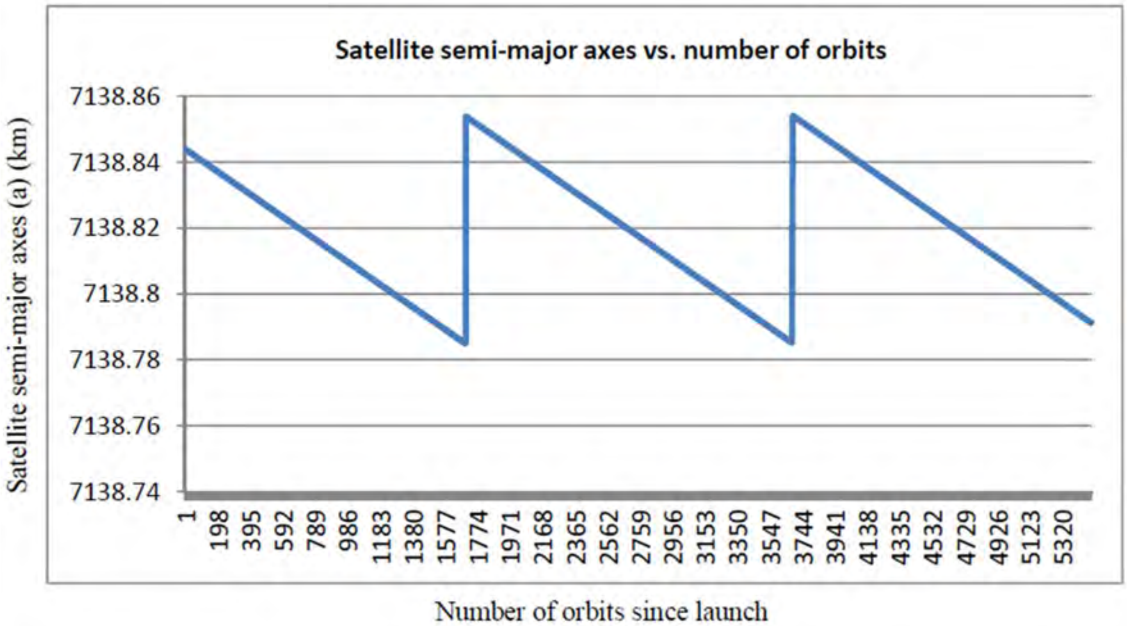

Fig. 15 Satellite semi-major axes versus number of orbits since launch

In each case, PPT needs to fire for either a fraction (0.86 for case 1, 0.45 for case 2) or at order of (5.1 for case 3) a period of the satellite in orbit after each 138 days to raise its altitude for 69 m with a velocity change of 0.04 s per maneuver. The amount of propellant required to perform this maneuver, alongside with the maneuver time, and the number of maneuvers per life time have been calculated for the three cases and four different life times of 1, 3, 5, and 7 years. The results show that PPT is an appropriate option for this mission.

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