Feasibility Study of All Electric Propulsion System for 3 ton class Satellite

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Abstract

Current satellite industry is heading towards High Throughput Satellite (HTS) to maximize the payload to Lift-Off-Mass (LOM). The propulsion system constitutes approximately 55\% of the total mass in the communication spacecraft with the conventional bi-propellant system. Although many subsystems are amenable for weight reductions, mass can be significantly reduced using Electric Propulsion System (EPS). In view of this, most of the worlds major telecommunications satellite manufacturers are implementing EPS and are able to bring down satellite mass from 5.5-6 tons to 3.5-4 tons for the same payload content.

Compared to conventional chemical propulsion, EPS ability to support more station-keeping and orbital maneuvers with the same propellant mass extends the on-orbit lifetime of the spacecraft and provides propellant margin for effectively increasing the spacecraft life.
A smaller and cheaper launch vehicle may be utilized for the mission if sufficient propellant mass and volume savings are achieved on the spacecraft bus, thereby permitting increased payload to orbit and decreasing the per unit launch cost. As onboard power availability continues to increase with developments in new solar panel materials and architectures, the higher thrust capability afforded to EP systems makes them more desirable for primary propulsion applications.

Today EPS is seen as future of Space Propulsion because of high specific impulse and ISRO is very keen to utilize this novel technology in current/future generation satellites on its platforms. ISRO’s I-3K bus with modifications provides feasible configuration for implementing Electric Propulsion for orbit raising, Station Keeping, inclination correction etc., and other satellite propulsion requirements.

This paper describes the feasibility of implementing the Electric propulsion to a 3-ton class satellite platform for various mission requirements.

**Key Words**: Electric Propulsion System (EPS); Chemical Propulsion; Spacecraft; Station Keeping; Orbital manoeuvres.

### 1 Introduction

Indian Space Research Organisation (ISRO) is playing a lead role in establishing a strong national communication infrastructure for societal, educational, industrial and entertainment sector applications. Presently ISRO has standardized various satellite platforms like I-1K, I-2K and I-3K bus to serve various payloads requirement. Also the next generation spacecraft bus like I-6K with Lift-off-Mass (LOM) up to 6000 kg is under development. For communication spacecraft, the prime focus always is on the maximization of the payload content i.e. to improve ratio of payload mass to LOM of the satellite.

Satellite propulsion system is very important part of the satellite system and is primarily intended to provide the required thrust (or, delta-V) to place the satellite into desired orbit and maintain the
satellite at the desired location/orientation. Also, once satellite’s useful/operational life is over, propulsion system is used for placing the spacecraft in graveyard orbit.

There are various types of spacecraft propulsion each having its own advantages and disadvantages. One of the most popular and widely used satellite propulsion systems today particularly for Geostationary spacecraft is Chemical Propulsion System (CPS), where the thrust is provided by chemical reaction, usually by burning (or oxidizing) a fuel. CPS is having a big share in total spacecraft mass almost 55% of LOM for 15 years of on-orbit life. Therefore it is a major contributor for the spacecraft launch cost. In all ISROs satellite mission, chemical propulsion is used extensively, has strong heritage and is easy to implement. A good alternate to CPS is Electric Propulsion System (EPS) which reduces overall LOM of satellite for the same operational life compared to CPS. Alternatively, for the same LOM, payload content can significantly be increased.

Electric propulsion system has come a long way since its first successful implementation in space propulsion in the year 1964 (by USA/NASA), thank to its multiple advantages over conventional chemical propulsion in space missions. In 1998 EPS was used in first deep space probe by NASA Dawn. Electric propulsion is now a mature and widely used technology on spacecraft. Russian satellites have used electric propulsion for decades.

Figure 1: Predicted growth of Electric propulsion satellites (Source NSR, APSCC, Q1 2015 Newsletter)
The most common application of EPS in communication satellites is Station Keeping (SK) of Geostationary Earth Orbit (GEO) satellites such as Boeing 702, SSL 1300, Astriums Eurostar 3000, Thales space bus 4000 and OHB, SGEO etc. EPS is also used extensively in various scientific missions like NASAs DAWN, Artemis, ESAs SMART-1, GOCE and JAXAs HAYABUSA etc. In 70% of commercial communication satellites Electric propulsion is used mainly for SK operations. Boeing has flown two All Electric Satellite, (ABS-3A and Eutelsat-115 WEST-B) weighing around 2200 kg (which otherwise would have launch mass of about 3500 kg) using Falcon-9 launcher in 2015. The EPS of these satellites is based on Xenon Ion Propulsion System. Thales-Alcatel and Astrium are jointly developing next generation European GEO platform called NEOSAT. ELECTRA will be the first European all EP telecom satellite to be launched in 2018-19.

By the end of the decade, Northern Sky Research (NSR) forecasts that 16% of commercial GEO communications satellites will employ full EPS and 24% will use hybrid propulsion for orbit rising.

2 Electric Propulsion System v/s Chemical Propulsion System

Chemical propulsion is said to be Energy Limited because the propellant have a fixed amount of stored energy per unit mass, which limits the achievable exhaust velocity. Electric propulsion systems are termed as Power limited because the rate at which energy from the external source is delivered to the propellant is limited by the capacity of on board power generation.

Electric thrusters in general are about one and a half times more efficient compared to chemical propulsion system. The advantages of EPS include higher $I_{sp}$, high propellant efficiency, precise on-orbit and transfer orbit maneuvers etc. EP devices operate at low thrust and hence they are more easily able to balance exact disturbance forces experienced by spacecraft in orbit. As a result, they can be utilized efficiently for tasks such as drag compensation on spacecraft requiring precise pointing. Since EPS is highly efficient system, a significantly smaller mass of propellant as compared to CPS is sufficient to provide same spacecraft on-orbit life.
Though EPS has many attractive advantages there are few disadvantages too. EPS generate relatively lower thrust (because of limited on-orbit power) and as a result it takes more time for orbit transfer than conventional CPS. Electric propulsion thrusters for spacecraft may be classified into electrostatic, electro-thermal and electromagnetic based on method used to propel the ions of the plasma. Of late thrusters working on the electro static principle namely stationary plasma thrusters and Gridded Ion thrusters is primarily used for EPS applications. Figure 2, clearly depicts that for the same LOM the EPS will provide greater Delta-V (m/s) compared to CPS.

2.1 Relationship between Thrust and Required Power

Although the $I_{sp}$ value gained using electric propulsion are significantly higher than the chemical propulsion, the available on-board power of a spacecraft limits the amount of thrust that electric thrusters are able to deliver. The relationship given in equation 1 shows that for a given unit of thrust, the required power is proportional to $I_{sp}$.

$$\frac{P}{T} = \frac{1}{2\eta} \times I_{sp} \times g_0$$

Where:
- $P$ = Power required (Watts)
- $T$ = Thrust (N)
- $g_0$ = Gravitational parameter (9.81 m/s$^2$)
- $I_{sp}$ = Specific impulse (s)
- $\eta$ = Efficiency
3 Elements of Electric Propulsion Systems

EPS comprises of various complex components which include storage (typically Xenon tank) and feed system that stores and feeds the propellant to the thrusters to generate thrust, the valves, piping which connects the propellant storage system with the thruster, and Power Processing & Control Unit (PPCU) to operate electrically the valves and thrusters. The power system supplies D.C power to PPCU and other auxiliary units such as valves, heaters, temperature sensors, telemetry monitoring channels etc. The PPCU processes the raw power into the specific form required by the thrusters (sometimes as high as 5 kV) and it is one of the most crucial elements. Figure 3 shows the simplified block diagram of EPS.
4 Propellant requirement for All electric satellite

All-electric satellite uses electric propulsion for all of its propulsion needs. To calculate the propellant required for the satellite to successfully complete its life (15 years in this case), primarily ∆V requirements for orbit raising, station keeping, attitude control and de-orbiting are required to be calculated. Here ∆V represent change in velocity of the spacecraft which also determines the consumption of propellant.

In all requirements of ∆V mentioned above, the maximum ∆V needed for orbit transfer from Transfer Orbit (TO) to GEO is the major contribution towards propellant consumption. The calculation of propellant required for transfer orbit maneuver (once the satellite is placed in the TO by launcher) is a complex procedure. Since this is a non-coplanar orbit transfer and also the thrust of EP thruster is very less as compared to chemical thruster, the process is completed in multiple corrections/maneuvers. The number of orbital maneuver required for maximum efficiency could be in order of 100 or more. This is certainly a time consuming process. Also
it puts the spacecraft through multiple crossing of the Van-Allen radiation belt. Hence, orbit maneuvers from T.O to GEO may take about 6-8 months.

Propellant consumption can be calculated using rocket equation. The ideal rocket equation is given by:

\[ \frac{M_p}{M_i} = 1 - e^{-\frac{\Delta V_{Tot}/g_0}{I_{sp}}} \]  

Using this equation mass of xenon/propellant consumed can be computed.

Burn duration, can be computed as

\[ T = \frac{(M_i - M_f)}{\text{Flow rate}} \]  

Where,
- \( M_p \): Mass of propellant consumed
- \( M_i \): Initial mass of the satellite
- \( M_f \): Final mass of the satellite
- \( \Delta V_{Tot} \): Total V (Change in velocity) required
- \( g_0 \): Acceleration due to gravity at the Earth’s surface
- \( I_{sp} \): Specific impulse of the engine
- \( T \): Burn duration

Assuming a GSLV which places the communication spacecraft in T.O with perigee of 180 km and apogee of 36000 km with inclination of 19.3 deg, \( \Delta V \) needed to reach GSO is about 1650 m/s. Station keeping maneuver needs \( \Delta V = 750 \) m/s for 15 years mission life. Considering orbit raising and station keeping and other maneuvers, total \( \Delta V \) required about 2500 m/s.

5 Spacecraft Configuration and Mission strategy

ISRO is designing, building and launching the spacecrafts over 4 decades. Over a period of time, ISRO has standardised its satellite platform buses, namely viz. I-1K, I-2K, I-3K, I-6K bus. Out of these platforms, I-3K platform with suitable augmentation is considered for all-electric-spacecraft.
Generic I-3K is having a cuboid bus structure of size 2000 mm x 1770 mm x 3100 mm and lift of mass of around 3.2 ton. Power generation is around 6500 W which is generated by using two winged solar array with three panels each on North and South wing, with highly efficient multi junction solar cells. Eclipse support is provided by having 2 numbers of 180 Ah Li-Ion batteries. Attitude and Control System (AOCS) uses 3 axes body stabilized control system with sensors and actuators. Propulsion system uses Unified Bipropellant system with 16 thruster configuration and one 440N Liquid Apogee Motor (LAM).

As the power requirement of the all EPS satellite is much higher, of the order of 15kW, I-3K bus need to be suitably modified for all EPS satellite. The LOM of EPS satellite is around 2200 kg including 450 kg of Xenon propellant. Figure 4 shows the typical view of All EPS Spacecraft. Propulsion system consists of four 75 mN thrusters which are mounted using Thrusters Pointing Mechanism (TPM) on the AEV side of the spacecraft with an angle of 45 deg. Two numbers of 300 mN thrusters are mounted on the AEV side of the spacecraft for orbit raising. 4 Nos of PPCU are used for converting raw power into the specific form required by the thrusters, 2 to 3 Xenon tank of 150 litres each for propellant storage, 2 sets of pressure regulation system main and redundant. Power requirement is primarily increased in order to support the power requirement for two numbers of Space plasma thrusters (SPT) each of 300 mN which is used for orbit raising operations. This apart, four numbers of thrusters of 75 mN each is considered for the station keeping operations which requires power of 2 kW each. Solar array is configured with two solar arrays, each having 5 solar panels with increased panel dimensions. As power generation being higher power distribution and control is suitably modified. It is assumed to deploy the solar array in transfer orbit followed by the orbit raising manoeuvres; power requirement can easily be met in transfer orbit as payloads are off. Also with the generation of higher power, it is possible to support 24 Ku transponders in saturation. For station keeping one thruster is fired at the ascending node followed by the firing of the paired thrusters at the descending node. Attitude and Orbit Control System (AOCS) uses zero momentum system with Gyros in loop, updated by Star Sensors.
PPCU and thrusters being high dissipating elements possess challenge for maintaining the desired temperature, apart from standard thermal design of Bus and Payload systems. As the orbit raising will take few months, it calls for spacecraft onboard autonomy and automation of EPS operations, also multiple crossing of the Van Allen radiation belt necessitates the suitable radiation hardening of the electronic systems.

Case studies
ISRO I-3K bus is having 3000 to 3500 kg LOM capacity with around 10% of LOM of the payload. By implementing EPS in ISRO
I-3K bus, it is possible to reduce LOM significantly and hence the launch cost. A mass budget in terms of percentage contribution of different subsystem of ISRO satellite bus with both all chemical and all electric propulsion is shown in Figure 5.

Here, two case studies are carried out to show the advantage of all-electric propulsion satellite over chemical propulsion satellite for mission requirements.

Case 1: In this case chemical propulsion and all-electric propulsion satellite platforms are compared in terms of subsystem mass keeping same payload mass content in both satellites platforms.

A typical I-3K bus has a payload mass of around 300 kg. Now by considering the mass contributions as shown in Figure-5 for all chemical propulsion based satellite LOM would be around 3100 kg, where as it is around 2200 kg for all EPS satellite. It can be noted that the all EPS satellite dry mass is more than that of CPS satellite. The difference in LOM is huge approximately 1000 kg. It is a significant amount considering the launch cost. International launch cost per kg of LOM is around USD $24000 (equivalent to INR 15,60,000/kg) and when launched with Indian launch vehicle it will be USD $15,500 (equivalent to INR 10,07,500/kg). Hence launch cost saving comes around 24 million USD (150 Crore Indian rupees) which is a huge amount and is almost one third of the cost of a 3-tones class of satellite itself.

Case 2: In this case chemical propulsion and an all-electric propulsion satellite platforms are compared in terms of subsystem mass keeping the LOM same. For comparison, LOM capacity of I-3K bus is considered as 3100 kg.

A CPS satellite with LOM of 3100 kg can accommodate payload of approximately 300 kg. For all electric satellite with LOM of 3100 kg, we can have payload capacity over 440 kg. The effective increase in payload capacity is 140 kg. This results into increase in the number of communication transponders/ units and hence overall revenue increases accordingly.

These calculations clearly shows that EPS is a cost efficient and cost effective alternate of CPS and that is why it is seen as future of spacecraft propulsion.
6 Conclusion

EPS makes it possible to increase/maximize the usable payload capacity for a given LOM or to decrease LOM. This mass saving can be utilized to equip more payloads in spacecraft hence generating more revenue in case of commercial spacecrafts or alternatively can reduce launch cost. Considering the lower mass of All-electric satellite, it allows to have a dual launch on various launchers available, optimizing the use of capacity of launch vehicle. In general, small satellites traditionally result in a higher cost per on-orbit transponder than the more efficient large satellites, but EP is one method of sufficiently reducing expenditure to achieve a competitive per transponder price.

This paper brings out the spacecraft configuration of all EPS satellite. Case studies have been carried out to show the advantage of all EPS satellite over the satellite using chemical propulsion in terms of increase in payload considering same LOM and also the decrease in overall LOM of satellite. For an all EPS satellite LOM is about 1000 kg less, resulting into saving in launch cost of around 150 Crore.

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