Operationally Enhanced Electric Propulsion Performance on Electrically Propelled Spacecraft

IEPC-2005-246

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Operational techniques are presented that allow performance enhancements to Electric Propulsion (EP) missions employing EP units capable of operating over varying power levels. Such techniques could be implemented by updates in hardware or software onboard, with respect to more traditional approaches. With such techniques, both thrust and specific impulse increases are shown to be possible. This is achieved by increased autonomy and intelligence onboard, and by integrating the electric propulsion and power subsystems control logic, within an electrically propelled spacecraft. This paper presents methods that could be employed by future EP spacecraft operators to maximise overall performance. The impact on ground operational complexity is also considered. Since both thrust and specific impulse enhancements are shown to be possible, the operator can choose which to implement, within the constraints dictated by the particular EP system used. The increased thrust option would be valuable for reducing thrust times, perhaps applicable to spacecraft operators wishing to reduce orbit transfer time, and therefore operational cost to station. Choosing to enhance specific impulse translates naturally into propellant savings, which could be used to increase lifetime. The paper will focus primarily on orbit raising spacecraft, but the enhancements could be employed on a wider range of electric propulsion missions. Some side effects of the techniques are also explored, in the context of cost efficient spacecraft operation.

I. Introduction

The highly successful completion of the Electric Propulsion phase of the SMART-1 mission[1] has paved the way for future orbit raising missions employing Solar Electric Propulsion (SEP). Over many years considerable effort has been invested in efficiency improvements at Electric Propulsion subsystem level by research and development organisations. As the technology begins to mature however, large efficiency gains become more difficult. As flight experience is gained, analysis of how the EPS is integrated, controlled and operated at system level can yield some possible improvements for future orbit raising missions. The focus of this paper is to explore safe and robust ways of integrating further, the way in which the power and electric propulsion subsystems are operated on spacecraft, to boost the overall propulsive efficiency. The methods described here are made possible by

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flexible EP subsystems that are able to be operated over a wide rage of input power levels, such as that used on SMART-1[2].

II. Operational considerations when orbit raising

The trend in modern spacecraft design is for increased on-board autonomy and reduced ground station contact. For Electric Propulsion orbit raising missions this implies long periods without ground station contact with the Electric Propulsion subsystem active and thrusting. Since the orbit is being continuously changed by the Electric Propulsion subsystem, accurate modelling of the effect of the engine is required. This information is propagated in ground orbital models, such that the ground station antenna pointing times and angles passed by the control centre to the ground stations, will be sufficiently accurate to achieve acquisition of signal (AOS) at the next programmed ground station pass. Any robust operations concept for an SEP spacecraft orbit raising from low Earth orbital altitudes (such as LEO or GTO) will be designed to cope with orbital uncertainties. Such orbital uncertainties occur from slight performance variations in the engine, which over time, can build into significant miss-pointing, reported by the ground station as Time Offset Values (TOVs). The beamwidth of a ground antenna depends upon its size, but is often only of the order of 1° solid angle, or even less. As such, depending upon the size of the performance variation, the time between ground station passes, and the orbital geometry, AOS may not occur at the predicted time. In such cases the ground station would report an early, or late, acquisition time in the form of a TOV.

The situation is more difficult when an unplanned shutdown has occurred outside of ground station contact. Depending upon time between ground station passes and the exact timing of the unplanned shutdown, a large cumulative impulse will effectively be 'missing' from the planned trajectory. In this situation, AOS will most likely require specialised search routines for the ground stations to acquire the spacecraft signal. It should be noted that the origin of the unplanned shutdown could come from various spacecraft modules through the system level FDIR (Failure Detection Isolation and Recovery) software. Since there are many onboard subsystems, it is likely that the origin is not the Electric Propulsion subsystem itself[d]. This type of robustness was present in the orbit determination and propagation developed by flight dynamics at ESOC for the SMART-1 mission, and is described by Mackenzie et al[3]. The effect can be schematically understood with reference to figure 1. Generally, missing impulse, either due to small thrust underperformance or an unplanned shutdown, implies that the spacecraft is in a lower orbit than expected, travelling faster, and thus giving an earlier than expected AOS at the ground station antenna. The opposite case is true if the thruster has 'over-performed', which has been observed in the SMART-1 mission. The magnitude of this effect is of course highly dependent on the orbital geometry and is most relevant to orbit raising around the Earth. Interplanetary missions, for example, would be largely unaffected.

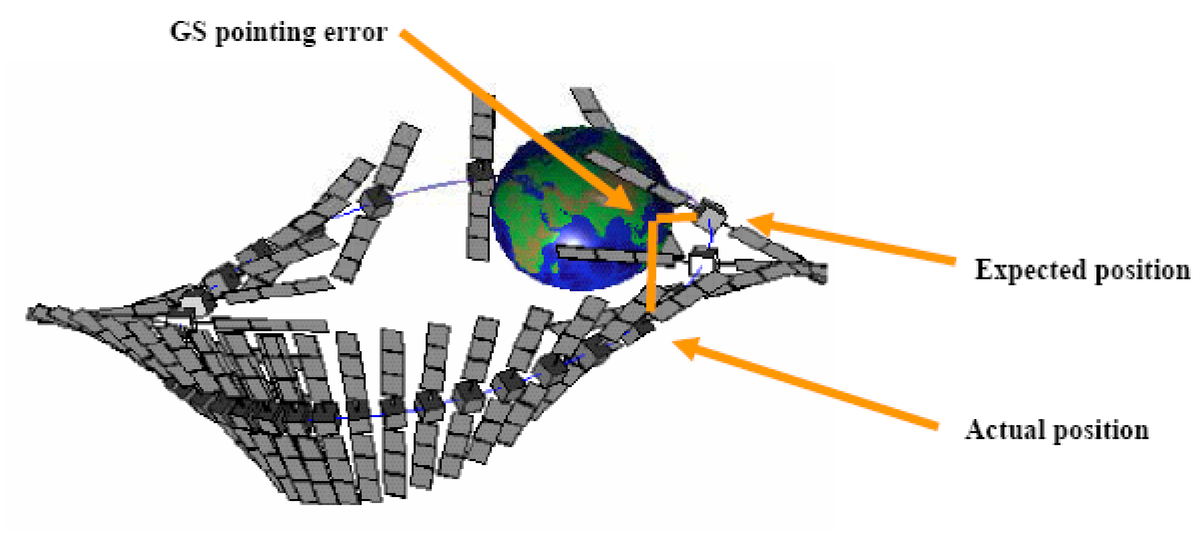

Figure 1 Schematic of the effect of EP performance variations and unplanned shutdowns on ground station contact (missing impulse shown)

III. Maximised operational use of on-board resources

Any spacecraft employing an EP system can benefit from maximising the power available to that subsystem. The benefit gained can be understood with reference to equation 1, which shows the electrical input power to an EP system, assuming ~50% electrical efficiency[7].

$$\frac{P_e}{T} = g_0 I_{SP} \qquad \textbf{\textit{equation 1}}$$

P_e = Electrical input power
g_0 = Acceleration due to gravity (at Earth's surface)
T = Thrust
I_{SP} = Specific Impulse

The important factor is that increased power utilisation by the electric propulsion system implies increased thrust and/or specific impulse. For an electrically propelled spacecraft this implies faster transfer times for a given mission and/or more efficient propellant utilisation.

A. Spacecraft Power Budget

Spacecraft power generation subsystems typically generate more power than is used by the sum of all spacecraft subsystems. The spacecraft power conditioning and regulation function (usually performed by analogue electronics) is designed to dissipate the unused generated power by the use of dissipative shunts/radiators. Examples of such shunting techniques include the use of dissipative loads (e.g. dump resistors that thermally dissipate the excess electrical power) or simple short-circuiting of the Solar Array sections (e.g. the currents generated by solar array sections, that are not needed, are short circuited across the generating solar array section, rather than fed into the spacecraft).

When a spacecraft power subsystem is designed, a power budget is drawn up which includes margins. After the launch, in practice, this usually means there is considerable excess power available that will not be used by any of the spacecraft subsystems. For the case of a spacecraft using an EP system, this extra power could be fed into the EP system, if the EP system is designed to operate over a variable power range. The power budget on an electrically propelled spacecraft is quite different to that of a conventional spacecraft, in that it is dominated by one load, and that load can be designed to benefit from extra available power.

Apart from power budget design margins, it is useful to consider the instantaneous power budget on-board a flying spacecraft. Typical spacecraft power consumption is characterised by a number of static and dynamic loads. Static loads are all those loads that are switched ON and stay ON for very long periods, with power consumptions that do not alter appreciably. Examples of static loads include on-board computers and data-handling equipment, receivers, demodulators, gyros and star trackers. Dynamic loads are loads that either frequently change their consumption, or are actively switched ON and OFF. Examples of dynamic loads include reaction wheels (used for attitude changes, and the thermal subsystem, where spacecraft heaters are regularly switched ON and OFF as spacecraft unit temperatures change. Typically, the thermal subsystem exhibits the largest dynamic power consumption range. The effect is illustrated in figure 2 below, taken from SMART-1, which at this stage of the mission was using a 100W margin to allow for periodic autonomous reaction wheel offloadings (not visible in this graph).

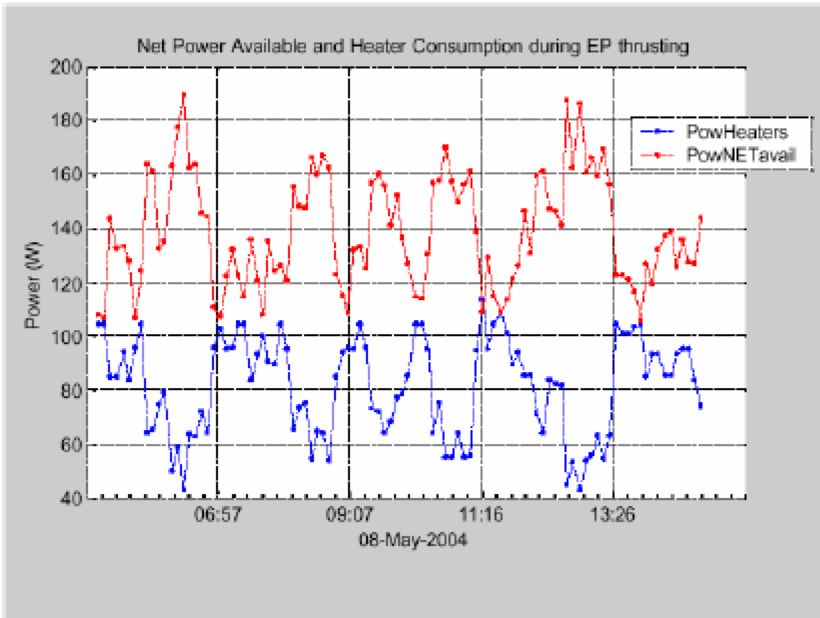

Figure 2 Total Heater Power Consumption, and overall instantaneous net available onboard power

B. Autonomous Sunlight EP throttling

By integrating the spacecraft power regulation and conditioning function with the operation of the EP subsystem, it is possible to design an autonomous control loop to automatically channel excess power into the EP system, which is able to operate at variable power levels[4]. As more and less power becomes available (e.g. the switching ON/OFF of spacecraft heaters) the EP is automatically adjusted to use whatever power is available. In this way a greater proportion of spacecraft generated power can be propulsively utilised. The excess power is thus transformed into greater thrust, and/or greater specific impulse, depending upon the implementation. Such a control loop could be performed by hardware using the MEA (Main Error Amplifier) voltage signal, which is used in the power electronics and is proportional to the instantaneous power budget (see figure 3).

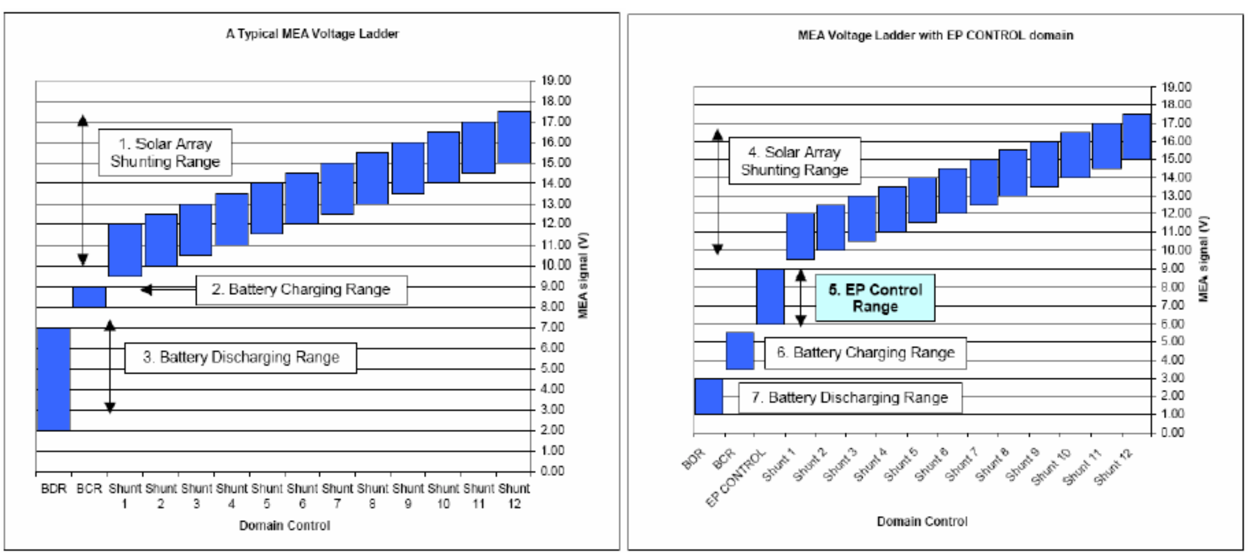

Figure 3 A typical MEA signal for a regulated power bus (left) and with the addition of an EP control domain for throttling the EP power (right)

A typical spacecraft power subsystem consists of solar arrays and batteries, and is controlled and regulated by analogue electronics. When the generated power exceeds the sum of all loads, the shunt regulator takes control of spacecraft power management. Any excess generated power is automatically dissipated in resistive shunts and/or across solar array sections. The signal driving the decision on how much power is shunted is the MEA (Main Error Amplifier) voltage. The MEA voltage is derived from the bus voltage as compared to a stable reference. The position within the output signal range is proportional to the excess generated power. Figure 3 shows a typical implementation, which is explained below.

- In figure 3 (left), when the output of the MEA is below 7V the BDR (Battery Discharge Regulator) uses the MEA voltage signal to supply the correct amount of power from the batteries to the loads (point 3 figure 3).
- When the generated power is higher than the sum of all the loads, excluding battery charging, but not greater than the sum of all the loads plus the maximum battery charge power (when the batteries are not fully charged), the BCR (Battery Charge Regulator) uses the MEA voltage signal to control how much charge current is supplied to the batteries. This is the voltage range 8-9V, point 2, in figure 3.
- The third domain used is the Solar Array shunting range (MEA voltage signal > 10.5V, point 1, in figure 3). In this range, the sum of all spacecraft loads (including battery charging, if operative) is less than the total generated power. This 'unnecessary' power needs to be removed from the bus in some way, and this is achieved either by adding new loads (e.g. resistive shunts that thermally dissipate heat) or by removing input power (e.g. some solar array sections are short circuited, such that their power does not enter the spacecraft). The MEA signal is used to control how many solar array sections are shunted and/or how many resistive shunts are operating.
- Thus, depending upon the excess generated power, the three domains maintain the spacecraft power balance by use of the MEA voltage signal.

An addition of a fourth domain, is shown as EPC (EP Control domain), illustrated as point 5 in the right hand plot of figure 3 (6-8V MEA). When EP thrusting is desired, the MEA voltage signal is used to control how much power is used by the EP system. This function could be enabled/disabled by an uplinked telecommand (TC). The EPC domain could function somewhat similarly to the BCR domain. If automatic EP power consumption control is not desired, the EP would draw no load, or insufficient load, to prevent shunting taking place. The MEA voltage signal would automatically climb above the EP Control domain and control would pass to the Solar Array Shunting domain. This feature allows for periods when the EP is off and for the EP to be operated in conventional ways, where power consumption is constant, or less than the excess generated power.

Automatic control could also be performed by software on-board the spacecraft. If this approach were chosen, it would offer the operators greater flexibility. Such an implementation could use tuneable (changeable by simple TC) hysteresis levels and power margin. This would allow maximum robust power utilisation by the electric propulsion subsystem without any impact on the batteries. It would also be possible to allow the batteries to absorb some of the peaks of the dynamic loads, but this would add a large number of shallow cycles to the batteries, which could complicate the battery sizing design process considerably, especially for orbit raising spacecraft that are designed for a high number of eclipses.

C. Use of SEP through eclipses during orbit raising

Orbit raising missions experience regular eclipses, which vary slightly depending upon season and the exact orbit. Usually the batteries are sized to ensure safe spacecraft operation through the longest predicted eclipse. Since the EP load is typically a large fraction of the overall power budget, it may be decided to exclude EP usage in eclipses due to the mass penalty incurred by a large energy storage system. Also, since EP systems typically have a very high power demand, thrusting through eclipses can be considered dangerous. Careful consideration of the operating principle is necessary to produce a robust system capable of thrusting through eclipses. If battery sizing for the longest eclipses of a mission at full power (EP thrusting) is considered not possible due to battery mass penalty at design phase, a subset of smaller eclipses could still be possible, at a reduced EP power level.

It is also useful to note that if the spacecraft is designed such that it is unable to support any EP operation through eclipses at all, then it must be operationally ensured that the EP system is off at eclipse entry. When this constraint is applied in conjunction with thrust level uncertainties, and relatively infrequent spacecraft commanding, the actual time the EP system is off around an eclipse can be several times the eclipse length (e.g. a 20 minute eclipse length can require a 100 minute EP off time, see figure 4).

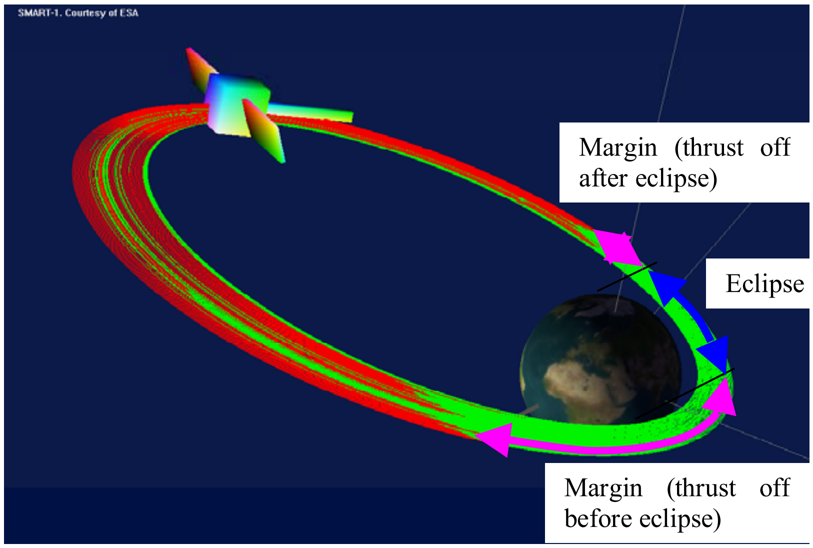

Figure 4 Early SMART-1 trajectory showing eclipse and EP off margins

A similar principle as that explained in section B above can be applied for EP thrusting through eclipses[5]. This time the goal is to safely maximise the available power for Electric Propulsion through the eclipse, such that this goal includes both efficiency and robustness with respect to failure modes. The technique is to automatically control EP power consumption, by targeting a certain battery capacity at the eclipse exit. This could be implemented using a software algorithm on-board, which computes the total power consumption of the spacecraft at any one time. The algorithm also simultaneously estimates the total battery capacity. The software combines the current battery capacity estimate with the current power consumption (including the EP load), to estimate the battery capacity at the end of the eclipse. The eclipse length is also known by the software, either because the software has an orbit model, or because the ground uplinks this data periodically. Finally, the end of eclipse battery capacity estimate is used to change and control the EP power consumption dynamically through the eclipse, such that a safe battery level is ensured at eclipse exit. Since the level is automatically targeted by on-board software (OBSW), it is robust against failures, and at the same time, more efficient than could be done by manual ground control. If no OBSW control is used, then this could only be implemented by ground prediction, which has to include much larger margins to cover failures that in most cases will not occur.

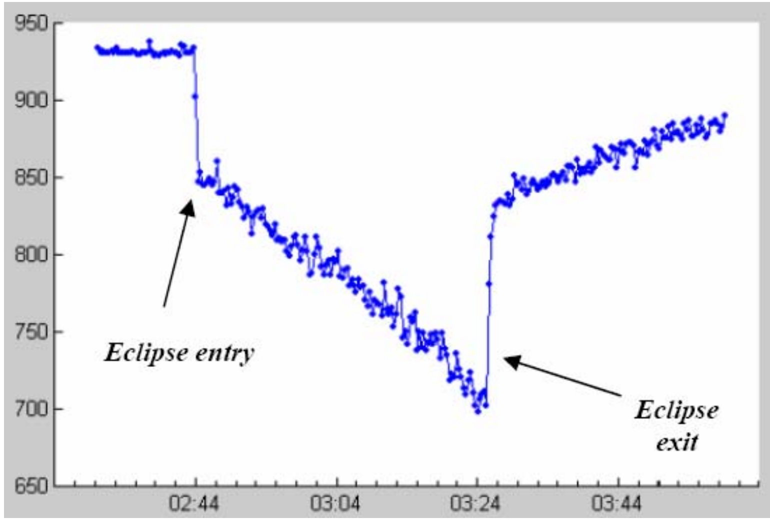

Figure 5 An example Battery Charge Status (Whrs) through an eclipse (SMART-1 data)

Figure 5 above shows the output of a battery charge status algorithm in the OBSW. Excluding the transients at eclipse entry and exit, the important part is the quasi-linear section within the eclipse itself. Such an algorithm was developed by SSC and was used as part of the FDIR of the SMART-1 spacecraft[6]. This type of algorithm would be used by the control loop to automatically throttle the EP subsystem. For example, if it were decided that a battery charge status of 200Whrs was acceptable at the end of the eclipse, this would mean that in the above example figure, a total of 500Whrs energy, spread over the eclipse length of around 40 minutes, is available for thrusting. This would equate to a mean power available for EP of 750W in eclipse. The eclipse length of 40 minutes is known by the software, so that as heaters switch ON and OFF during the eclipse, the power consumed by the EP is adjusted, so that, in this example, 200Whrs is available at the end of the eclipse. In practice, 200Whrs may be considered too low, but the principle is illustrated. An important aspect of this technique is that it is intrinsically failure tolerant. If for some reason multiple unexpected loads come up, the EP consumption is automatically throttled down, or even switched OFF, such that whatever battery charge status needs to be guaranteed at eclipse exit, is ensured.

IV. Operational Impact of Autonomous EP throttling

Section III above detailed autonomous techniques potentially enabling operators to dynamically boost the available power for Electric Propulsion. This has the positive impact of reducing transfer times for orbit raising missions, and increasing specific impulse. The specific impulse gain translates into a reduced propellant cost of a given manoeuvre. Reduced transfer times are interesting for operators wishing to reduce their time (and hence cost) to being on-station. The exact gains that could be achieved are dependent upon the type of mission and the type of EP subsystem (i.e. the EP input power equation 1, is in practice affected by the thruster efficiency changes over the input power range). There are however, highly developed EP subsystems available that are capable of efficiently throttling over wide power ranges, such as that provided by Snecma for the SMART-1 mission[2]. A SMART-1 type mission could potentially have its thrust performance boosted, especially in the early Van Allen belt escape phase, when there are frequent eclipses, which also has the benefit of reducing radiation exposure. Propellant and transfer time saving predictions are relatively complex functions of the exact transfer trajectory strategy used. A simple analysis of the effect of allowing EP thrust during the eclipse period only yields gains of a few percent in terms of transfer duration. However, when the required margins involved for infrequent ground contact are applied, this figure can climb, up to a 10% benefit (see figure 4). This projection does not include the application of the sunlight throttling gains. As such, it is not unrealistic to predict that the combined application of both throttling loops could yield benefits in the 10%-20% range. These figures are at present indicative, and could rise or fall with more detailed analysis.

It is interesting to consider the effect of autonomous EP throttling, unseen by ground (i.e. between ground station passes). This would produce effects similar to those described in section II. The main source of autonomous thruster throttling would be from the dynamic load changes in the thermal subsystem, and this would not change dramatically on an orbit by orbit basis. As such, it is expected that the orbit could be modelled and propagated using a similar approach as that developed by SMART-1[3], with the effect of the autonomous throttling being akin to the small thrust performance variations.

If the operators are designing the operations concept of the orbit raising mission from scratch, some options could be considered to help alleviate the effects of impulse variations out of coverage. These would be applicable to all types of thrust variations out of coverage, whether their origin is in autonomous EP power control or unplanned shutdowns. Some ground stations could be used of reduced complexity, which would just track the carrier signal from the spacecraft. Such an antenna would not require the telemetry (TM) and TC encoding/decoding chains. The antenna would follow the spacecraft in auto-track and the antenna pointing angles could be compared with those expected. If an unexpected shutdown of the EP had occurred then an angular offset would start to build up. This information would then be used to request an antenna with full TM and TC capabilities to investigate. Otherwise, the time between the use of full TM/TC ground station passes could be safely increased, relying in-between on the feedback of the smaller less complex antennas.

Ultimately, the problem could be completely solved by use of an on-board orbit model on the SEP spacecraft and the use of a data relay satellite to communicate with ground. The SEP spacecraft is itself always capable of knowing its position, since it would know if an unplanned (FDIR initiated) shutdown had occurred, or could log the autonomous throttling. If a relay satellite were used, the SEP spacecraft would be able to communicate with it at the designated time. The ground antenna would only need to point at the relay satellite, which does not alter its position. This would eliminate the need for antenna search routines capable of searching over wide angular ranges. Such a system should be considered on balance in the frame of cost effective operations. Even though such a system offers an effective technical solution to antenna pointing errors, it may not be the most cost effective approach when the complexity of the on-board orbit model and relay satellite is included.

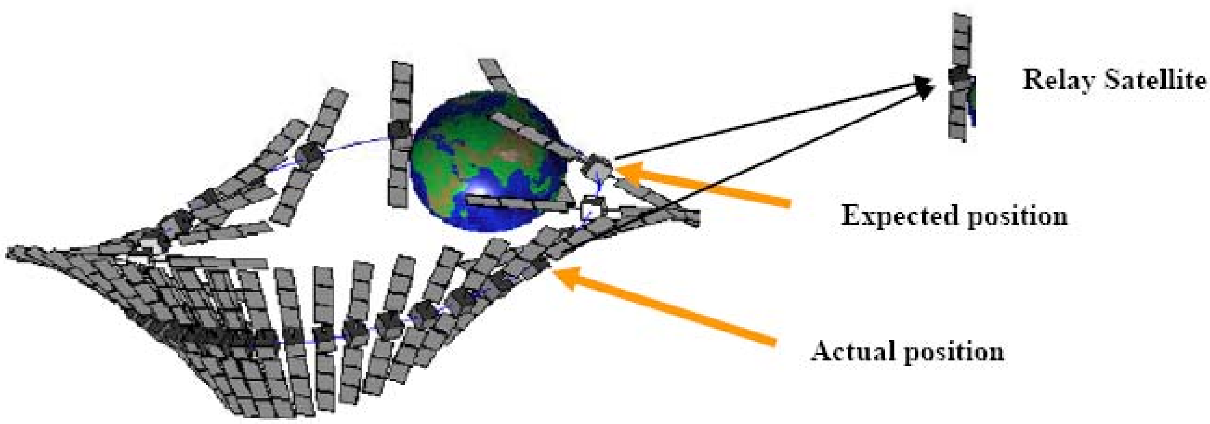

Figure 6 The use of a relay satellite to eliminate antenna pointing errors

V. Conclusion

Operational techniques are described that allow potential performance enhancements to Electric Propulsion orbit raising missions, employing EP units capable of operating over varying power levels. These techniques involve the autonomous control of the Electric Propulsion power consumption onboard, with inherent failure tolerance included. This is achieved by increased autonomy and intelligence in the integration of the electric propulsion and power subsystems control logic. In sunlight, the power consumption is automatically varied with the use of dynamic loads, such that power utilisation by the EP is maximised. In eclipse, a similar approach is applied with an on-board algorithm computing the estimated end of eclipse battery status. An indicative analysis suggests that the potential performance gains are around 10%, when compared to a more conventional implementation. The impact of unexpected impulse changes, whether induced by unplanned (FDIR initiated) shutdowns, or by the use of autonomous EP throttling, is explored in terms of operational complexity and impact on the ground segment. Possible changes in terms of the ground segment design are suggested for future SEP orbit raising systems, including the use of a relay satellite that would in effect eliminate antenna pointing errors. Such a system however, such be considered on balance with a more traditional approach in terms of overall cost effectiveness.

Acknowledgments

The experience of operating the SMART-1 spacecraft has been instrumental in developing the ideas contained in this paper, and so the authors would like to thank all those involved in the SMART-1 mission.

References

[1]Milligan et al. "SMART-1 Electric Propulsion Operational Experience". IEPC-2005-245.

[2]Koppel, C., Lyszyk, M., Valentian, D., Saccoccia, G., Estublier, D."PPS 1350 With Variable power Feature for SMART-1" 36th AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit, 17-19 July 2000, Huntsville, Alabama.

[3]R. Mackenzie et al. "Orbit determination of the SMART-1 mission". 18th International Symposium on Space Flight Dynamics. Munich, 2004.

[4] VEGA Gmbh. (D. Milligan) Patent Application GB0409164.1, 20th March 2004 "Automatic Control of Electric Propulsion Power Consumption".

[5] VEGA Gmbh. (D. Milligan) Patent Application GB0410056.6, 6th May 2004 "Enhanced use of Spacecraft Electric Propulsion through eclipses"

[6] Racca, G. D. et al, "SMART-1 missions description and development status." Planetary and Space Science, 50 (2002), pp. 1323, 1337.

[7] C. Koppel, " Advantages of a continuous thrust strategy from a geosynchronous transfer orbit, using high specific impulse thrusters", 14th International Symposium on Space Flight Dynamics, 08-12 February 1999, Foz do Iguaçu - Brazil.