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Engineering Notes

ENGINEERING NOTES are short manuscripts describing new developments or important results of a preliminary nature. These Notes cannot exceed six manuscript pages and three figures; a page of text may be substituted for a figure and vice versa. After informal review by the editors, they may be published within a few months of the date of receipt. Style requirements are the same as for regular contributions (see inside back cover).

Cost-Optimum Electric Propulsion for Constellations

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Nomenclature

A = constant coefficient of thruster power cost function, W/N
B = linear coefficient of thruster power cost function, Wkg/N^{2}s
C_L = specific cost for launching and injecting the spacecraft, cost$/M_o$, cost expressed in contentional units (c.u.)
C_M = specific cost for spacecraft manufacturing, the cost$/M_{\mathrm{dry}}$, c.u.
D = total return from a constellation, dollars
F = thrust, N
g_{th}^W = thruster power cost per unit thrust $(W_{\mathrm{TU}} \cdot \eta_{\mathrm{PC}})/F$, W/N
I_{sp} = specific impulse, Ns/kg
K_T = constant of proportionality
L = $C_M/(C_M + C_L)$
M_{dry} = spacecraft dry mass, kg
M_{OPS} = mass of the onboard propulsion system, kg
M_o = spacecraft mass in parking or transfer orbit before startup of the onboard propulsion system, kg
M_P = mass of the propellant, kg
N = number of spacecraft per constellation
n_0 = spacecraft lifetime, years
R = profitability, D/Z
T_p = onboard propulsion system powered flight time, h
T_0 = solar power system lifetime, h
W_{TU} = onboard propulsion system power, W
$\bar{W}_{\mathrm{TU}}$ = onboard propulsion system specific power, W_{TU}/M_o, W/kg
W_0 = spacecraft nominal power, W
$\bar{W}_0$ = spacecraft specific power, W_0/M_o, W/kg
Z = total building cost of a constellation, dollars
α = tank factor, mass of tankages/mass of the propellant
γ_{th} = thruster specific mass, thruster mass/thruster power consumption, kg/W
γ_w = onboard propulsion system specific mass, $M_{\mathrm{OPS}}/W_{\mathrm{TU}}$, kg/W
ΔT_P = total hours of electric propulsion system operations in the year, h/year
ΔV = characteristic velocity increment per maneuver, m/s
η_o = overall electric propulsion system efficiency, $\eta_T\ \eta_{\mathrm{PC}}$
η_{PC} = power conversion subsystem efficiency, thruster power/propulsion system power consumption
η_T = electric thruster thrust efficiency, thrust power/thruster power consumption
λ = profit rate, $(D - Z)/Z$
μ_P = propellant mass fraction, M_P/M_o
μ_{TU} = onboard propulsion system mass fraction, M_{OPS}/M_o
ξ = coordination factor
ϕ = solar-array power degradation coefficient through the Van Allen belts

Subscripts

C = chemical propulsion
E = electric propulsion
e = electric absorption
i = index of the ith onboard propulsion system
opt = optimal
t = thermal dispersion

Superscripts

A = apogee
S = north–south stationkeeping

Introduction

THE accelerated growth of space telecommunication technologies and the availability of new advanced electric thrusters requires an update of the spacecraft propulsion systems optimization criteria. The traditional optimization, which maximizes the payload mass function, is replaced in this study by another criterion maximizing the profitability of the electric propulsion system for a given mission. This different approach will permit designers of commercial satellite constellations to consider electric propulsion system in terms of financial parameters, i.e., expenses, net profit, or extra profit.

With respect to traditional chemical propulsion, electric propulsion can offer a dramatic improvement of performance. The potential applications encompass essentially all in-space satellite propulsion requirements. Although the benefits from the application of xenon electric propulsion to north–south stationkeeping of satellites in geostationary Earth orbit (GEO) are already well known, the advantages coming from more demanding tasks like the satellite orbit raising from parking up to operational orbit and its end-of-life disposal now become viable.

This fact is especially true for satellite constellations in low Earth orbit (LEO), where the use of electric propulsion systems brings substantial advantages in terms of both the cost reduction of constellation displacement and the extension of the satellite economic life. The advantages referred to allow an extra profit comparable with the basic one. The optimization of electric propulsion system performance for these missions will significantly increase the extra profit.

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Methodology

The profitability R_E ensuing from the adoption of an electric propulsion system (EPS) for a satellite constellation can be defined as the ratio between the total return D_E and total cost Z_E of the constellation. In this study the function R_E is maximized by optimizing the EPS variables on the basis of the economic, mission, system, and EPS parameters of the constellation.[1]

Considering a spacecraft (S/C) equipped with different onboard propulsion systems, the total return from a constellation can be considered proportional to the number of spacecraft N, the net mass injected on parking or transfer orbit (for LEO or GEO constellations, respectively), the net energy available for S/C taking into account the consumption required by the EPS:

$$D_E = K_T N\left[M_o - \sum_i (M_{\mathrm{OPS}})_i\right]\left[\frac{W_0 T_0 - \sum_i (W_{\mathrm{TU}} T_p)_i}{W_0 T_0}\right] \quad (1)$$

The basic cost of a constellation includes the cost for launching and injecting the S/C in parking or transfer orbit and the cost of S/C manufacturing. The cost of propellant and that connected with recurring normal operations of the constellation can be neglected with respect to the basic one. Therefore, the following expression can be considered valid:

$$Z_E = N \cdot M_o\left[C_L + C_M \frac{M_o - \sum_i (M_P)_i}{M_o}\right]$$

On the basis of the available statistics for LEO constellations (costs expressed in conventional units, 1 c.u. $\approx$ 1 k\$/kg): $C_L = 6$ c.u. and $C_M = 16$ c.u. (Ref. 1). For GEO constellations using onboard chemical propulsion systems $C_L = 30$ c.u., and $C_M = 62.5$ c.u.

Therefore, the conditions determining the EPS profitability can be presented as follows:

$$R_E = \frac{K_T}{C_L + C_M} \cdot \frac{1 - \sum_i (M_{\mathrm{OPS}})_i / M_o}{(1 - C_M)/(C_M + C_L) \cdot \sum_i (M_P)_i / M_o} \times \left[1 - \frac{\sum_i (W_{\mathrm{TU}} T_p)_i}{W_0 T_0}\right] \quad (2)$$

When the EPS is the sole onboard propulsion system, Eq. (2) can be rearranged as follows:

$$R_E = \frac{K_T}{C_L + C_M} \frac{1 - (1+\alpha)\mu_P - \gamma_w \cdot g_{\mathrm{th}} \cdot (\mu_P \cdot I_{\mathrm{sp}}/\eta_{\mathrm{PC}} \cdot T_p)}{1 - L \cdot \mu_P} \times \left(1 - \frac{1}{2}\frac{\mu_P \cdot I_{\mathrm{sp}}^2}{\eta_T \eta_{\mathrm{PC}} \cdot \bar{W}_0 \cdot T_0}\right) \quad (3)$$

Concerning the powered flight time T_p, it should be noted that Eq. (3) does not take into account the effects on capital "freezing" and turnover. Without taking into account these factors, the powered flight time would be limited only by EPS lifetime. Otherwise, should a loan should be returned the parameter T_p has a prominent optimum depending on the fraction of the loan in the overall cost and the corresponding interest rate.

Optimization of EPS Performance for LEO Constellations

Figure 1 shows the relation between $R_E/R_E^{\max}$ vs T_p, where $R_E^{\max}$ corresponds to R_E for $T_p \to \infty$. Computations have been performed for EPS parameters $\alpha = 0.2$, $\gamma_w = 15$ kg/kW, $I_{\mathrm{sp}} = 16$ kNs/kg, $g_{\mathrm{th}}^W = 16$ kW/N, $\eta_{\mathrm{PC}} = 0.9$, and velocity increments representative of LEO constellations $\Delta V = 500$–1250 m/s. A powered flight time of 2000 h allows the achievement of 95–98% of $R_E^{\max}$ (considering not limitations caused by loans or other reasons). For EPS equipped with Hall effect thrusters, considering the thrust efficiency η_T, the EPS specific mass γ_w, and the specific impulse I_{sp} as independent parameters, Eq. (3) shows that η_T should be selected at the maximum level, γ_w at the minimum, and I_{sp} needs to be optimized. The optimum specific impulse can be approximated using the following relation:

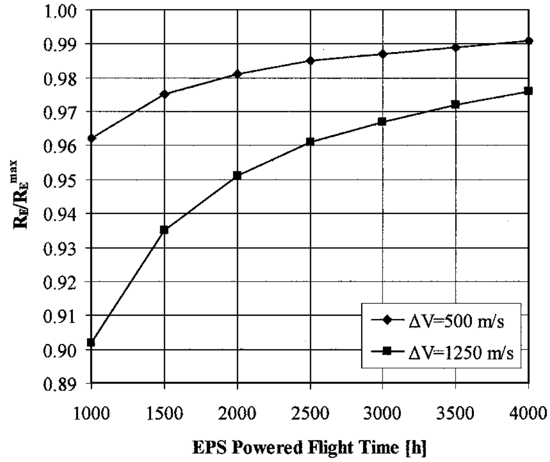

Fig. 1 EPS profitability vs EPS-powered flight time for LEO constellations.

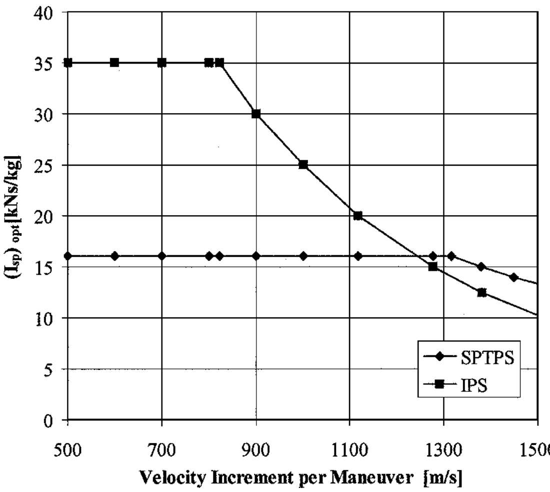

Fig. 2 Optimal EPS specific impulse vs characteristic velocity increment of LEO constellations.

$$(I_{\mathrm{sp}})_{\mathrm{opt}} = \xi\left\{\Delta V \cdot L + \sqrt{\frac{2(1+\alpha-L)}{\gamma_w}\eta_T \eta_{\mathrm{PC}} T_p}\right\} \quad (4)$$

The coordination factor ξ depends on the influence of EPS-power subsystem on the operational lifetime of the onboard solar power system and on power level employed by the payload. When these influences can be neglected, $\xi = 1$. However, from operational considerations it is possible to assume $\xi = 0.8$. In particular, for EPS with $\eta_o = \eta_T \eta_{\mathrm{PC}} = 0.45$, $\gamma_w = 15$ kg/kW, $T_p = 2000$ h, $\bar{W}_0 = 3$ W/kg, $T_0 = 10$ years, the optimal specific impulse for $\Delta V = 500$–1500 m/s would range between $(I_{\mathrm{sp}})_{\mathrm{opt}} = 12$–$12.5$ kNs/kg, respectively.

The second term of Eq. (4) in the square bracket is significantly greater than the first and differs from the optimal specific impulse obtained by mass criteria by the parameter L characterizing the constellation building cost. Thus, the optimal specific impulse that maximizes the constellation profitability is lower than that calculated only from mass criteria.

However, in a more detailed description, considering the experimental data shown in Refs. 2 and 3 for Hall-effect thrusters and ion thrusters (IonT), respectively, it is possible to assume the thruster power cost g_{th}^W as a linear function of the specific impulse $g_{\mathrm{th}}^W = A + B \cdot I_{\mathrm{sp}}$. Assuming a thrust assembly with a redundancy of 100%, the specific mass of a xenon EPS can be presented as $\gamma_w = 3.5\gamma_{\mathrm{th}} + 9.6$, where γ_{th}, γ_w in kilograms/kilowatts. For Hall-effect thrusters $\gamma_{\mathrm{th}} = 1.5$–$1.7$ kg/kW. Figure 2 shows the results of the optimization performed for Hall-effect thruster EPS (SPTPS) and ion thruster EPS (IPS) by using the following input parameters in Eq. (3): $\bar{W}_0 = 3$ W/kg, $T_0 = 10$ years, $T_p = 2000$ h, $\eta_{\mathrm{PC}} = 0.9$, $L = 0.727$, $\alpha = 0.2$.

The horizontal lines correspond to optimized specific impulse for EPS having a specific power lower of that of the S/C ($\bar{W}_{\mathrm{TU}} < \bar{W}_0$). The higher the ΔV the higher the EPS specific power. When $\bar{W}_{\mathrm{TU}} = \bar{W}_0$, an increase of ΔV is possible only if the EPS specific impulse decreases. In that case the following relationship has been found:

$$\Delta V = I_{\mathrm{sp}}\left[1 - \sqrt{1 - \frac{2\eta_{\mathrm{PC}}\bar{W}_0 T_p}{I_{\mathrm{sp}}(A + B \cdot I_{\mathrm{sp}})}}\right]$$

It is noticeable that, under the assumed input parameters, for $\Delta V = 1250$ m/s the maximum specific impulse does not exceed 16 kNs/kg.

It is also possible to calculate the profitability ensuing from the replacement of a chemical propulsion system (CPS) with an EPS. The profit rate coming from revenues of a telecommunication constellation using CPS can be defined as

$$\lambda_C = (D_C - Z_C)/Z_C = R_C - 1$$

the replacement of the CPS with a EPS would increase the profitability of the constellation up to $\Delta = (R_E/R_C)(1 + \lambda_C) - 1$, which would result in the following extra profit:

$$\Delta - \lambda_C = (1 + \lambda_C)(R_E/R_C - 1)$$

Figure 3 shows the parameter $\bar{R} = R_E/R_C$ calculated for optimized SPTPS and IPS in the range of $\Delta V = 500$–1500 m/s. In the computation of R_C, we have assumed $(I_{\mathrm{sp}})_C = 3.08$ kNs/kg and $(M_{\mathrm{TU}})_C = 1.12\ (M_P)_C$. From that figure we see that optimized xenon EPS allows increased profitability parameter of 3–10% with respect to CPS having the same cost.

We consider that the EPS will be used to raise the S/C from an initial circular parking orbit at 185-km altitude up to the operational orbit and then to displace back the S/C to the same parking orbit after its operational end of life. In this case Fig. 4 shows the value of the extra profit as a function of the operational orbital altitude and the value of the profit rate λ_C, which will exist on the market of telecommunication satellites at the moment one substitutes the CPS with the EPS. From the same Fig. 4 we can see that for altitudes over 1000 km the extra profit goes up to 13% of the costs.

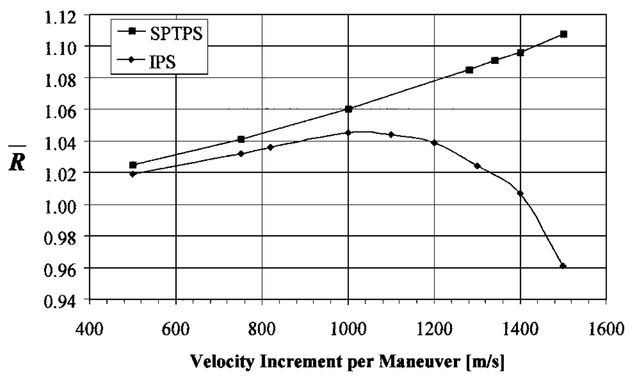

Fig. 3 EPS profitability vs characteristic velocity increment of LEO constellations.

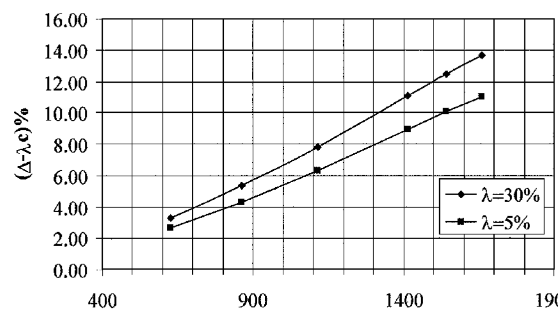

Fig. 4 SPTPS extra profit vs LEO altitude and chemical profit rate.

Optimization of EPS Performance for GEO Constellations

The benefits from the application of EPS to north–south stationkeeping (NSSK) of GEO satellites are already well known. The adoption of optimized EPS also for orbit raising from transfer orbit to GEO would allow a significant reduction of the propellant mass required with respect to traditional CPS, therefore reducing dramatically costs associated with on-orbit displacement of GEO constellations and allowing an extra profit regardless of the increase in manufacturing cost. Several authors have already investigated the possibility of using EPS for this operation.[2,4]

In terms of profitability, an EPS power degradation ϕ factor must be taken into account in the analysis as a term D_E to be included in Eq. (1). In case the CPS is separated from the S/C before the startup of the apogee EPS and considering separated EPS for NSSK and orbit transfer, we can obtain[1]

$$R_E = \frac{K_T}{C_L + C_M}\cdot\left[1 - \frac{\left(\bar{W}_{\mathrm{TU}}^A \cdot T_p^A\right)_E}{\bar{W}_0 \cdot T_0} + \frac{\bar{W}_{\mathrm{TU}}^S \cdot \Delta T_p \cdot n_0}{\bar{W}_0 \cdot T_0}\cdot\left(1 - \mu_p^A\right)_E\right]$$
$$\times\frac{\phi\left[1 - \left(\mu_{\mathrm{TU}}^A\right)_E - \left(1 - \mu_p^A\right)_E\cdot\left(\mu_{\mathrm{TU}}^S\right)_E\right]}{1 - L\cdot\left(\mu_p^A\right)_c / 1 - \left(\mu_{\mathrm{TU}}^A\right)_c - L\left[\left(\mu_p^A\right)_E + \left(1 - \mu_p^A\right)_E\cdot\left(\mu_p^S\right)_E\right]} \tag{5}$$

The square bracket in the numerator of Eq. (5) represents the final net mass after the full utilization of the integrated EPS. From the analysis in Ref. 4, it follows that the numerator reaches its maximum for $\Delta V_C^A = 1000$ m/s. For xenon EPS this corresponds to $\Delta V_E^A = 1250$ m/s and $\phi = 0.97$. As can be seen from Fig. 3, with $\Delta V_E = 1250$ m/s, $\bar{W}_0 = 3$ W/kg, $(T_p^A)_E = 2000$ h the profit rate $(\bar{R} - 1)$ of SPTPS is about three times higher than that allowed by IPS. Therefore, the advantage of SPTPS with respect to IPS for orbit transfer operations to GEO is evident. For NSSK operations it is possible to use both SPTPS or IPS. However, a fully integrated EPS based only on SPTPS has a number of advantages with respect to a combination of Hall-effect thrusters and ion thrusters. As for instance the power processing unit of integrated SPTPS can be common to both stationkeeping and orbit transfer thrusters.

Conclusions

The optimization of electric propulsion system performance for low-Earth-orbit and geostationary-Earth-orbit constellations according to the profitability criterion leads to a decrease of the specific impulse of Hall-effect thrusters to 12–15 kNs/kg. Furthermore, as a consequence of the optimization of the powered flight time, the electric propulsion lifetime would be shortened to 2000–3000 h, which would make flight qualification easier and less expensive. Electric propulsion system based on Hall-effect thruster optimized towards low specific impulses reduces power converter output voltage leading to an increase of the power converter efficiency and to a decrease in the specific mass.

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Spacecraft Thermal Analysis and Control by Thermal Energy Optimum Distribution

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Nomenclature

E_{IN} = energy for battery charge
E_{OUT} = energy needed from the battery
P_{BCR} = charge power
P_{SP} = power produced by solar panel
T_{eq} = equilibrium temperature
T_{req} = temperature requirement

Subscripts

min, max = minimum, maximum value

Superscript

r = rth orbit

Introduction

THIS Note presents a new thermal design and control technique based on the concept of an optimum global thermal energy distribution, achieved by a successive model size increase through a critical substructuring method, using MATLAB® software codes, until NASTRAN finite element analysis and verification. Figure 1 shows the procedure flowchart, which consists of three main phases: system definition, thermal design, and detailed modeling and verification. The technique has been applied to the design of the SMART microsatellite, developed at the Universities of Naples under the sponsorship of the Italian Space Agency. SMART ($450 \times 450 \times 360$ mm stowed; 33 kg; 64-W average power) is a three-axis stabilized multimission microsatellite aimed at carrying remote sensing payloads on sun-synchronous orbits[1] with altitudes ranging from 400 to 1000 km. The in-orbit configuration of SMART is shown in Fig. 2.

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Technique Description

The different procedure subphases are described in the following subsections.

System Analysis

The first step is aimed at designing a suitable configuration. SMART design, mass, and dimensions were determined in order to guarantee compatibility with the majority of launch opportunity. The simplest solution was retained: bus of cubic shape with solar panels deployment consisting in a rigid rotation of the lateral surfaces around the deployment axis (Fig. 2). The flight configuration selection consists in computing the optimum deploying angles; the optimum condition is given by referring to both power and thermal requirements.

Environmental Modeling

To simulate satellite-environment interactions, a MATLAB code was developed.[2] Thermal inputs and electric energy can be computed for each configuration under analysis (i.e., for each couple of deployment angles θ_1 and θ_2).

Global Thermal Energy Estimation

At this level of design, thermal analysis is based on the preceding assumptions. Power analysis is carried out considering solar panel working at their maximum power point at the end of life and at worst condition, as it is assumed in electric power subsystem (EPS) design. Three orbits, during summer and winter solstices and spring equinox, corresponding to the extreme values of the sun angle, are selected for the analysis. The optimization procedure is based on the comparison of the electric power generated along the orbit with the power requirement. In particular, for each orbit and for each configuration two energies must be computed, as well as the minimum and maximum temperatures. A given configuration can be considered preliminarily acceptable if

$$\exists\left[P^r_{\mathrm{BCR,min}} \quad P^r_{\mathrm{BCR,max}}\right] \subset \left]0 \quad P^r_{\mathrm{SP,MAX}} - P_L\right] : \forall P_{\mathrm{BCR}} \in \left[P^r_{\mathrm{BCR,min}} \quad P^r_{\mathrm{BCR,max}}\right] \Rightarrow E_{\mathrm{IN}}/E_{\mathrm{OUT}} \geq 1 \tag{1}$$

and for each subsystem

$$\left[T^r_{\mathrm{eq\,min}} \quad T^r_{\mathrm{eq\,max}}\right] \subseteq \left[T_{\mathrm{req\,min}} \quad T_{\mathrm{req\,max}}\right] \tag{2}$$

Then, considering orbits in different periods of the year, the configuration can be selected only if

$$\bigcap_i \left[P^r_{\mathrm{BCR,min}} \quad P^r_{\mathrm{BCR,max}}\right] \neq \emptyset \tag{3}$$

The configuration (in the worst orbit case) giving the maximum value of $E_{\mathrm{IN,max}}/E_{\mathrm{OUT}}$ is chosen.

Results

The analysis has been based on SMART microsatellite power budget[3] with orbital data derived by the Polar Satellite Launch Vehicle (PSLV) assuming a descending node local time of 1100 (Ref. 1). Solar panel performance are computed considering data of space qualified GaAs solar cells.[3] Figure 3 shows three surfaces representing the value of the ratio E_{IN}/E_{OUT} for each configuration (θ_1, θ_2). Each surface has been computed for one of the three orbital cases considered.

Moreover, assuming a margin of 15% on E_{IN}, the 1.15 level curves for the three surfaces show that the number of acceptable configurations is minimum at the winter solstice and that the configurations which are acceptable during the winter solstice are also acceptable in the other cases. Among the acceptable configurations, the one that gives the maximum energy ratio is selected ($\theta_1 = 16$ deg and $\theta_2 = 90$ deg).

Spacecraft Thermal Substructuring

Once the geometrical configuration of SMART is set, a critical substructuring is performed. The exposed area-to-mass ratio can be assumed as a distinguishing parameter between the different parts of the spacecraft. Note that, at this stage, the thermal paths between