Arcjet Propulsion System Study for NSSK

F. Scorteci*, L. d'Agostino†, F. d'Auria* and M. Andrenucci†

CENTROSPAZIO, Pisa 56014, Italy

The study of a low power arcjet propulsion system and its application to north-south stationkeeping of geostationary telecommunication satellites was carried out at Centrospazio in the framework of the arcjet propulsion system development programme conducted by BPD Difesa e Spazio under the Italian Space Agency sponsorship. In order to carry out this study, models were developed for all of the main components of the spacecraft. These included: arcjet thruster, propellant storage and feeding systems, power conditioning, solar arrays and batteries. The relevant characteristic parameters and variables, as well as their mutual-relationships, were defined in order to characterize each of the above components in terms of its performance, mass, electrical consumption and heat flux. The modules describing the different spacecraft systems were integrated in a code capable to simulate the system performance in a given mission and to optimize its design parameters on the basis of a properly chosen criterion.

Introduction

PROGRESS of the exploration and, in general, the utilization of space for scientific or industrial purposes is widely related to the availability of adequate propulsion and power systems. The use of electric thrusters for spacecraft propulsion can offer, on the medium term, the most significant increase of performance with respect to the current chemical systems. In particular, arcjet thrusters appear to have the most promising commercial application in the next few years1. Electric propulsion systems based on arcjet thrusters, with a power level ranging from 30kW to less than 1kW, are currently under development in the USA2, Europe3 and Japan4. The potential missions foreseen for these systems include interplanetary transfer5, orbit raising6, and orbit adjustment7.

In this framework, since 1988 Centrospazio has carried out the development of a 1 kW arcjet as part of an Advanced Space Technology Programme contract awarded to BPD Difesa e Spazio by ESA8. A follow-on programme, regarding the flight qualification of a 1.8 kW arcjet, is planned to start in fall 1993, with Centrospazio involved in the technology development and diagnostics application.

System activities started in spring 1992 under ASI sponsorship and included the development of the Power Conditioning Unit (PCU), the Gas Generator and the Diagnostic Package9. Within this programme Centrospazio has been involved in diagnostics definition10, PCU development11, and in modelling activities of the thruster12, the PCU13 and of the system as a whole14. As a part of this programme, Centrospazio has carried out a study of a low power arcjet propulsion system application for the north-south stationkeeping (NSSK) of a geostationary telecommunication satellite. The study was aimed at the characterization of the electric propulsion system (EPS) and its inter-relationships with the other spacecraft subsystems in order to provide information for the selection of the operational parameters of the various sub-systems and for trading-off the EPS design characteristics.

Reference Spacecraft and Mission

The reference mission and spacecraft configuration were defined according to the requirements and specifications approved by ASI within the arcjet system study definition programme, and are summarized in Table1. The NSSK of a 2600 kg-class geostationary telecommunications satellite was assumed as a reference mission. The satellite is configured as a 3-axis stabilized spacecraft with a dual mode propulsion system (DMPS), using hydrazine and nitrogen tetroxide (NTO) as propellants for the liquid apogee engine (LAE), plus catalytic and hydrazine-arcjet reaction control thrusters (RCT's).

The power system was assumed to comprise solar arrays and Ni-H2 batteries for eclipse period, which supply a shunt-regulated power bus. The power to the arcjets comes directly from the batteries.
### Table 1  Reference mission and spacecraft configuration

- **Launch Mass Range**: 2300-2900 kg
- **Mission Duration**: 15 years
- **Geostationary Orbit**
  - 3-axis stabilized
  - EWSK and NSSK Control Box from $\pm 0.01^\circ$ to $\pm 0.07^\circ$
- **Power System**
  - 5 to 8 kW during sun and eclipse (BOL and EOL)
  - Ni-H$_2$ Batteries for Eclipse Power
  - Bus Voltage $\geq$ 50V
- **Propulsion System**
  - Dual mode N$_2$H$_4$/NTO
  - Input Power to Arcjet System 1.5-2.2 kW
  - Arcjet System Power from Satellite Batteries
  - LAE is isolated by closing the N/O pyro valves of the system (Fig. 2). Two solar arrays supply a shunt-regulated 50 VDC bus with a power level ranging from 5 to 8 kW, throughout the whole operational life of the spacecraft. Packs of Ni-H$_2$ battery cells provide power continuity during eclipses. A partially regulated DC bus is assumed. This means that the solar array voltage is regulated during the sunlight period, while the battery discharge is unregulated.

---

**Spacecraft Configuration**

The DMPS schematic is shown in Fig. 1. The overall system can be divided into three main sub-systems:

- Pressurization sub-system,
- Propellant storage and distribution sub-system,
- Propulsion sub-system.

The first comprises three helium tanks pressurized at 17.58 MPa. A normally closed (N/C) pyrotechnic valve isolates these tanks from the rest of the system. A two-stage regulaor with a 0.69 MPa pressure drop is used to adjust the feed pressure.

Three tanks (two for hydrazine and one for NTO) store the propellants. A series of normally open (N/O) and non-return valves (NRV) separates these two sub-systems.

The propulsion sub-system comprises a 470N thruster for apogee maneuvers, 16 (redundant) liquid RCT's for EWSK and attitude control, 4 (redundant) electric RCT's (arcjets) for NSSK. The LAE is connected to the propellant feeding system both with N/C and N/O valves to allow separate operation with respect to the RCT sub-system. A series of latching valves allows to feed with pure hydrazine two redundant RCT's sub-systems. Tank and line heaters are used to maintain the propellants in the range of operating temperature.

The electric power system was considered as subdivided in three main components: the solar arrays, the battery system, and the power distribution and control sub-system (Fig. 2). Two solar arrays supply a shunt-regulated 50 VDC bus with a power level ranging from 5 to 8 kW, throughout the whole operational life of the spacecraft. Packs of Ni-H$_2$ battery cells provide power continuity during eclipses. A partially regulated DC bus is assumed. This means that the solar array voltage is regulated during the sunlight period, while the battery discharge is unregulated.

**Operating Concept**

The satellite operations can be broken into several distinct phases. During the launch phase the DMPS system is not active and the pressurant tank are isolated. After separation from the launch vehicle, the spacecraft performs various pre-apogee maneuvers to acquire a given orientation and a stable attitude. These operations are carried out by the catalytic RCT's operating in a blowdown mode. Just before the first apogee firing, the N/C pyrotechnic valve fires and the helium pressurizes the propellant tanks. The regulator maintains a given pressure in the propellant tanks during the LAE firings to establish the optimum pressure inside the LAE combustion chamber (about 1.6 MPa). By considering the various pressure drops inside the valves and lines, the propellant tank pressurization at BOL can vary from 1.8 to 2.2 MPa. During this phase the catalytic RCT's perform a series of attitude operations to control the spacecraft. After the completion of the apogee maneuvers the LAE is isolated by closing the N/O pyro valves of the feed lines, in order to prevent leaks for the remainder of the mission. The pressurization system is also insulated and a given propellant tank pressure is set. From this point on the propellant feeding system starts operating in a blowdown mode over the rest of the mission and supplies both the catalytic RCT's for 3-axis attitude control, contingency operations, EWSK and the arcjets for NSSK.

**Spacecraft System Models**

The spacecraft has been described in terms of different modules, each modeling a given system and sub-system.
IEPC-93-013

Kam discharges. A maximum DOD of 0.8 and a 100 Ah battery cell capacity was assumed. The maximum battery discharge voltage follows from the maximum discharge voltage of each battery cell (1.7 V) and the calculated number of cells in series. The battery system mass was calculated on the basis of the energy density data reported in literature ($E_d = 50$ Wh/kg).

$$M_{bat} = \frac{(N_b C V_{b,\text{max}})}{E_d}$$

The batteries can supply the arcjet system with a given power ($P_{PCLU}$) for a maximum time:

$$t_d = \frac{(N_b DOD (V_{b,\text{max}} - V_{b,\text{min}}))}{P_{PCLU}}$$

calculated by assuming the maximum discharge voltage range for the battery cells, a given DOD, the calculated cell capacity, and the effective number of thruster firings.

This result will be used to calculate the minimum number of ON/OFF cycles of the EPS. By knowing the number of payload cycles one can compare the calculated values with the maximum number of cycles allowed for the batteries.

$$N_{\text{max}} = 5000 \frac{(0.8/DOD)}{1.7^3}$$

The Power System module provides the PCU module with the calculated discharge voltage range and the mission module with the calculated discharge time.

**Propellant Feed System**

The PFS module is schematically subdivided in the pressurizing sub-system and the propellant storage and distribution sub-system.

The pressurization sub-system generates the propellant flux from the tanks by the displacement of a diaphragm under the action of a high pressure gas. The initial storage pressure inside the tanks must be high enough to allow for the spacecraft attitude control using catalytic RCT’s before the apogee injection. The pressurization of the propellant tanks starts just before the first apogee firing. Helium is usually chosen as a pressurant because of its low molecular weight. A simplified analysis of the pressurization system can be performed on the basis of the energy conservation principle. Assuming that the initial amount of gas in the feedline is small and that the gas evolution in tank is isothermal, the total mass of helium can be evaluated using

**Table 2. Specific mass of power sub-system with partially regulated DC bus**

<table>
<thead>
<tr>
<th>Module</th>
<th>Specific Mass (kg/kW)</th>
</tr>
</thead>
<tbody>
<tr>
<td>SOLAR ARRAY</td>
<td>25.0</td>
</tr>
<tr>
<td>CHARGE ARRAY</td>
<td>3.9</td>
</tr>
<tr>
<td>SHUNT</td>
<td>7.5</td>
</tr>
<tr>
<td>CHARGE CONTROL</td>
<td>1.5</td>
</tr>
<tr>
<td>BATTERY</td>
<td>47.3</td>
</tr>
<tr>
<td>DISCHARGE REGULATOR</td>
<td>0.2</td>
</tr>
</tbody>
</table>
the following equation:

\[
M_h = \frac{P_{BOL} V_p}{RT_o \left(1 - |P_{BOL} + P_r| / P_h\right)}
\]  

(1)

where \(P_{BOL}\), \(P_r\), and \(P_h\) respectively represent the propellant storage pressure at BOL, the pressure drop across the regulator, and the pressurant storage pressure, \(T_o\) is the storage temperature, \(V_p\) is the volume occupied by the propellant, and \(m_h\) is the molecular weight of the pressurant. The blowdown mode of operation starts after the last LAE firing. It has been modeled according to the following assumptions:

- the liquid hydrazine is an incompressible ideal fluid
- the helium gas inside the tank undergoes an isothermal expansion
- helium is an ideal gas
- no mixture between helium and hydrazine takes place
- the velocity of the boundary between gas and liquid is negligible.

The simplified Bernoulli equation for the hydrazine writes:

\[
P_{g} = \frac{\rho_o u^2}{2}
\]

and the hydrazine mass flow rate can be computed as follows:

\[
m' = \rho_o A u = A \sqrt{2 \rho_o P_g}
\]  

(2)

For an isothermal expansion the following expression holds:

\[
P_g = P_{eq} \left(\frac{V_{eq}}{V_g}\right)
\]  

(3)

and, since

\[
m' = \rho_o \frac{dV_g}{dt}
\]  

(4)

the expression for the gas volume evolution as a function of time writes:

\[
V_g(t) = \left(V_{eq}^{\frac{\gamma + 2}{2}} + \frac{\gamma + 2}{2} \frac{A t}{\rho_o} \sqrt{2 \frac{P_{BOL} \rho_o V_{eq}}{V_g}} t^{\gamma + 2}\right)^{\frac{2}{\gamma + 2}}
\]  

(5)

By substituting the eq. (5) into eq. (3) and then into eq. (2) one can estimate the mass flow rate and pressure blowdown throughout the mission. The hydrazine passage area \(A\) and the initial pressurant volume inside the propellant tank \(V_{eq}\) were evaluated to fit blowdown profile properly chosen according to the arcjet design criteria established by ASI's programme (Table 3). The area \(A\) was evaluated from eq. (3) for an initial 45 mg/s mass flow at 18 bar feed pressure. The volume \(V_{eq}\) was evaluated as a function of the initial and final pressures, \(P_{BOL}=1.8\) MPa and \(P_{eq}=0.8\) MPa, in order to have 30 mg/s mass flow rate after a total firing time of 1087 h. The calculated blowdown mass flow rate and pressure profiles are shown in Figs. 3 and 4. The hydrazine flow rate to the arcjet thruster is then used for the determination of the total impulse, as shown in the following.

### Electric Propulsion System

The EPS module includes the PCU and the arcjet thruster assembly (AJT). Four EPS systems are installed on the spacecraft; two of them are simultaneously fired during NSSK operations. The overall EPS module schematic and
its interrelationships with internal and external modules are illustrated in Fig. 5.

The PCU sub-system

The PCU's are supplied directly by the satellite battery buses. In this study, the input power was assumed to vary from 1.5 to 2.2 kW for each PCU. The voltage input ranges depending on the discharge characteristics calculated for the battery system ($V_{d_{, max}}$ and $V_{d_{, min}}$). The current input profile during discharge was then calculated by assuming the power input constant.

The PCU efficiency was assumed to vary linearly as a function of the input current. The literature reports values of 0.945 maximum and 0.911 minimum efficiency for a 1.8kW system with a minimum of 18.7 A and a maximum of 27.7 A input currents\textsuperscript{23}. The power output to the arcjet thruster is then calculated as a function of the battery voltage discharge range.

The PCU loss is considered as heat conducted from the baseplate to the spacecraft thermal control:

$$Q_{PCU} = P_{PCU,m} (1 - \eta_{PCU})$$

The PCU mass is considered a linear function of the power input. A 2.22 kg/kW specific mass was assumed\textsuperscript{23}.

The AJT sub-system

The arcjet performance has been evaluated by interpolating the experimental data measured during the testing activity at Centrospazio\textsuperscript{24} on a typical geometrical configuration capable to meet the assumed ASI goal performance given in Table 4\textsuperscript{25}. The voltage/current characteristics has been calculated as a function of the input current $I_{aj}$ and propellant mass flow rate:

$$V_{aj} = -32.67 + 6.74 m' - 0.054 m'^2 + (4.16 - 0.25 m' + 0.0025 m'^2) I_{aj}$$

The efficiency is assumed to be a linear function of the specific power ($P' = P_{aj,m} / m'$)

$$\eta_{aj} = 0.495 - 0.0042 P'$$

Thrust, specific impulse and arc current have been calculated from their definitions:

$$T = \sqrt{2 m' P_{aj} \eta_{aj}}$$

$$I_{aj} = T / m' \dot{m}$$

$$I_{aj} = P_{aj} / V_{aj}$$

<table>
<thead>
<tr>
<th>Table 4 Arcjet system performance</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>AVERAGE THRUST</strong></td>
</tr>
<tr>
<td><strong>MINIMUM SPECIFIC IMPULSE</strong></td>
</tr>
<tr>
<td><strong>INPUT POWER</strong></td>
</tr>
<tr>
<td><strong>LIFE</strong></td>
</tr>
<tr>
<td><strong>ON/OFF CYCLES</strong></td>
</tr>
<tr>
<td><strong>HYDRAZINE THROUGHPUT</strong></td>
</tr>
<tr>
<td><strong>SINGLE SYSTEM TOTAL IMPULSE</strong></td>
</tr>
</tbody>
</table>
The above system is determined when the power input $P_{\text{thr}}$ to the thruster and $m'$ are given. $P_{\text{thr}}$ is equal to the PCU output power minus a 20 W assumed as the power loss in the connection cable. An additional equivalent input power due to the enthalpy of the flow at 658 K coming from the gas generator is accounted for. The power absorption of the propellant valve and the gas generator heaters are not accounted for in the calculations due to their low contributions. The propellant mass flow rate is given by eq. (2) as a function of the feed pressure.

The total impulse generated by the arcjet has then been calculated by solving the following integral of the total propellant mass consumed for the NSSK operations ($M_{\text{arc}}$):

$$I_{\text{arc}} = \int \frac{T}{m'} dM_{\text{arc}}$$

From continuity considerations $dM_{\text{arc}} = \rho_{\text{arc}} dV_{\text{arc}}$ and, assuming an isothermal expansion, the following holds:

$$dM_{\text{arc}} = -\rho_{\text{arc}} V_{\text{arc}} (P_{\text{BOL}} / P) dP.$$ 

By substituting eq. (2), (6) and (8) into eq. (7), the total impulse and the arcjet average specific impulse

$$I_{\text{arc}} = \frac{I_{\text{arc}}}{g_{\text{ref}} M_{\text{arc}}},$$

can be calculated as functions of the blowdown pressure and the thruster power input.

After the optimization of the system has been performed, the BOL and EOL values of all of the propulsive parameters are calculated as functions of the optimized variables.

A given fraction of the arcjet heat losses is assumed to be conducted to the spacecraft. Similarly, a fraction of the radiated energy is assumed to return to the spacecraft structure. These quantities are taken to be linear functions of the total power input to the thruster, with the coefficients estimated from ref. (25):

$$Q_{\text{rad}} = 0.037 P_{\text{PCU,th}}$$
$$Q_{\text{cond}} = 0.00176 P_{\text{PCU,th}}$$

The total amount of the AJT and PCU heat losses affecting the spacecraft T/C has then been computed:

$$Q_{\text{tot}} = Q_{\text{cond}} + Q_{\text{rad}} + Q_{\text{cable}} + Q_{\text{PCU}}$$

A passive thermal control is assumed, with a specific mass of 40 kg/kW. For the mass budget the actual number of engines firing simultaneously ($N_{\text{thr}}$) is considered.

$$M_{\text{arc}} = Q_{\text{rad}} M_{\text{arc}} N_{\text{thr}} / T_{\text{arc}}$$

The overall AJT mass, $M_{\text{arc}}$, (thruster, gas generator and mounting structure) is assumed to be 1 kg over the entire power range from 1.5 to 2.2 kW considered in this study. The power cable mass ($M_{\text{cable}}$) is assumed 0.8 kg. Four EPS's are considered in the mass calculations. According to contract specifications, no gimbals are foreseen for this mission. The masses of the interface structures between the EPS and the spacecraft are assumed to be included in those of the PCU's and AJT's. All of the EPS control and housekeeping functions are assumed to be performed by the PCU's, so that no additional mass is considered. A mass ($M_{\text{str}}$) of 8 kg is assumed to account for the extra battery circuitry and other contingency masses.

$$M_{\text{str}} = N_{\text{eps}} M_{\text{PCU}} + N_{\text{eps}} M_{\text{cable}} + N_{\text{eps}} M_{\text{th}} + M_{\text{str}} + M_{\text{str}} + M_{\text{str}}$$

Structure Module

This module uses all of the computed masses of the various systems and calculates the overall mass budget of the satellite. The spacecraft dry mass is assumed to be subdivided in the following main components:

$$M_{\text{dry}} = M_{\text{arc}} + M_{\text{eps}} + M_{\text{ps}} + M_{\text{ps}}$$

where $M_{\text{dry}}$ and $M_{\text{arc}}$ are the masses of the EPS and Power system calculated by their own modules, $M_{\text{arc}}$ is the mass of the propellant feeding system, and $M_{\text{arc}}$ accounts for the dry masses (chemical propulsion system, payload, spacecraft structure etc.). The $M_{\text{str}}$ has been calculated using appropriate tankage fractions desumed from the literature.

$$M_{\text{str}} = M_{\text{str}} + M_{\text{str}} + M_{\text{str}} / (r + 1)$$

The helium mass can be calculated by means of the eq. (1) as a function of $M_{\text{hyd}}$ and $M_{\text{str}}$. The total hydrazine on board of the spacecraft is the sum of that required by the arcjets for the NSSK operations $M_{\text{arc}}$, that required by the catalytic thruster for the on orbit non-NSSK maneuvers $M_{\text{arc}}$, and that for the apogee kick-off, $M_{\text{arc}}$. If $r$ is the mixture ratio (oxidizer to fuel mass), the hydrazine mass used for apogee maneuvers is $M_{\text{str}} / (r + 1)$. The total hydrazine mass is then:

$$M_{\text{hyd}} = M_{\text{str}} + M_{\text{str}} / (r + 1)$$

For non-NSSK in orbit maneuvers, the propellant consumption has been calculated according to ref. (26):

$$M_{\text{ps}} = M_{\text{dry}} (M_{\text{ps}} + M_{\text{ps}} Y)$$

where $M_{\text{ps}} = 30$ kg and $M_{\text{ps}} = 4$ kg/yr respectively are the time independent and dependent contributions; 1640 kg is a spacecraft dry mass used for normalization purposes.

The geostationary satellite mass gains is given by:

$$M_{\text{geo}} = M_{\text{dry}} + M_{\text{ps}} + M_{\text{ps}} + M_{\text{helium}}$$

Finally, the satellite launch mass ($M_{\text{geo}}$) is:

$$M_{\text{geo}} = M_{\text{geo}} + M_{\text{str}}$$
Mission Model

The apogee and NSSK maneuvers were considered in the model. These missions have been calculated on the basis of assumed \( \Delta V \)'s. For the apogee maneuver a \( \Delta V \) of 1514 m/s has been assumed. The Royal Ordnance LEROS I with a 310 s \( I_e \)\(^{27}\) has been considered for the apogee kick-off maneuver. A reserve fraction \( P_f \) of 0.042 has been assumed\(^{26}\).

\[
M_a = M_{GEO} \left( \exp \left( \frac{\Delta V_{apogee}}{I_e} g \right) - 1 \right) (1 + P_f)
\]

\[
M_s = (M_{GEO} - M_a) \left( \exp \left( \frac{\Delta V_{NSSK}}{I_e} g \right) - 1 \right) (1 + P_f)
\]

The arcjet average specific impulse to be substituted in the last equation has been computed by eq. (9) of the EPS module. The above equations, plus those of the structure module, can be considered as an algebraic system of 15 equations in 15 unknowns (see Table 5) depending on four optimization (decision) variables and four parametric data (see Table 6).

When the NSSK propellant mass is known, the total firing time can easily be calculated from:

\[
M_a = \int_{t_a}^{t} m(t) \, dt
\]

after substituting eq. (2), (3) and (5).

The minimum number of ON/OFF cycles can also be calculated once the maximum discharge time of the battery system under EPS operation \( t_e \) is known or assigned. By considering two thrusters firing simultaneously with a given cant angle \( \phi = 15^\circ \), the average thrust during the mission can be calculated in first approximation from the total impulse:

\[
I_{TOT} = I_{q, \phi} g, \quad M_a = 2 \cos \phi I_{TOT}
\]

Knowing the average thrust and the average specific impulse, the average mass flow rate during the NSSK mission can also be calculated.

Optimization Procedure

The modules describing the different spacecraft systems were integrated in a code which provides the capability of simulating the system performance in a given mission defined by the parametric data and of optimizing the system variables on the basis of a properly chosen criterion. The code includes a procedure that allows one to assign the configuration of the systems to be analyzed, as well as the values of those parameters that are assumed as given.

The optimization was performed by means of Powell's quadratically convergent method\(^ {28}\). This is a conjugate directions method, with proper provisions in order to avoid that the minimization along one direction be spoiled by minimizations along the next ones. Powell's routine intrinsically assumes the variables to be unconstrained. On the other hand the present problem deals with constrained decision variables and parameters. Furthermore, these constraints are highly nonlinear. In order to account for this aspect in a simple but reliable way, the "penalty function method" has been chosen\(^ {29}\).

A set of fixed parameters and decision variables were defined in order to optimize the EPS for the NSSK mission and to carry out parametric studies. The parameters and variables, with their assigned ranges of variation, are shown in Table 6. The satellite launch mass was chosen as the function to be minimized. Indeed the objective function

<table>
<thead>
<tr>
<th>Table 6</th>
<th>Parametric data and decision variables</th>
</tr>
</thead>
<tbody>
<tr>
<td>Parametric Data</td>
<td></td>
</tr>
<tr>
<td>Power System</td>
<td>5 - 8 kW</td>
</tr>
<tr>
<td>Satellite dry mass</td>
<td>976 - 1872 kg</td>
</tr>
<tr>
<td>Yearly NSSK ( \Delta V )</td>
<td>42 - 50 m/s</td>
</tr>
<tr>
<td>Mission duration</td>
<td>10 - 15 years</td>
</tr>
<tr>
<td>Decision Variables</td>
<td></td>
</tr>
<tr>
<td>Power to EPS</td>
<td>1.5 - 2.2 kW</td>
</tr>
<tr>
<td>Propellant feed pressure, BOL</td>
<td>18 - 22 bar</td>
</tr>
<tr>
<td>Propellant feed pressure, EOL</td>
<td>8 - 12 bar</td>
</tr>
<tr>
<td>Battery DOD</td>
<td>0.4 - 0.8</td>
</tr>
</tbody>
</table>
was set as:

\[ M_{\text{GTO}} - M_{\text{GEO}} - M_e = 0 \]

## Results and Discussion

The code was run at first on a reference satellite configuration in order to acquire useful data on the general behavior of the system. This information is especially relevant to better understand the mutual relations of the variables, to acquire the necessary feeling for their orders of magnitude, and for orientation in subsequent parametric studies. Table 7 shows the results of the detailed analysis for the reference configuration.

The result obtained shows that the optimized decision variables \( DOD, P_{\text{BOL}}, P_{\text{EOL}} \) assume a limit value. \( DOD \) is maximum, while \( P_{\text{BOL}} \) and \( P_{\text{EOL}} \) are minimum. The power input to EPS is well within the assumed range.

A sensitivity analysis has been performed in order to provide information on the general behavior of the optimal solution. This analysis has been carried out by separately changing the parametric data in the neighborhood of the reference configuration (underlined value) as summarized in the following:

| Power system: | 5 kW |
| Mission duration: | 15 years |
| NSSK yearly rate: | 42, 44, 46, 48, 50 m/s-yr |
| Satellite dry mass: | 976, 1204, 1426, 1644, 1872 kg |

The results so obtained are shown in Figs. 6-9. The Tables 8 and 9 report the results corresponding to the extremes of the allowable range of variation for each parameter considered in the analysis.

As expected, the EPS power increases with the satellite mass at GEO and with the required mission total impulse due to the higher gain achievable on the propellant mass budget by increasing the arcjet specific impulse. Besides, the optimization always yields an EPS power value internal to its allowable range, and quite close to the 1.8 kw nominal EPS power for the satellite launch masses envisaged by the present ASI system development programme. The optimized values of the remaining decision variables \( (DOD, P_{\text{BOL}}, P_{\text{EOL}}) \) always assume one of their limit values. This result has also been confirmed by varying their ranges of variation. The maximum \( DOD \) value has been changed from 0.4 to 0.8, the range of \( P_{\text{BOL}} \) from 18 to 22 bar, and correspondingly \( P_{\text{EOL}} \) from 8 to 12 bar. The optimized EPS power, total firing time, ON-OFF firing cycles and average specific impulse are shown in Figs. 10-13 as functions of the \( DOD, P_{\text{BOL}}, P_{\text{EOL}} \). The calculated values in the extreme points of the allowable range are reported in Tables 10 and 11.

\( P_{\text{BOL}} \) and \( P_{\text{EOL}} \) affect the mass flow rate during the blowdown. The results indicate that \( m' \) must be as small as possible, in order to provide the highest specific impulse for the given EPS power input. It appears that, from the point of view of the spacecraft mass budget, the increase in the specific impulse given by using the arcjet system at low mass flow rate is more efficient than the increase

<table>
<thead>
<tr>
<th>Table 7 Reference configuration system analysis</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>PARAMETERS</strong></td>
</tr>
<tr>
<td>Power (Sun and battery)</td>
</tr>
<tr>
<td>( \Delta V ) for NSSK</td>
</tr>
<tr>
<td>Mission duration</td>
</tr>
<tr>
<td>Baseline spacecraft mass</td>
</tr>
<tr>
<td><strong>INPUTS</strong></td>
</tr>
<tr>
<td>Input power to PCU</td>
</tr>
<tr>
<td>Feed pressure BOL</td>
</tr>
<tr>
<td>Feed pressure EOL</td>
</tr>
<tr>
<td>Depth of discharge</td>
</tr>
<tr>
<td><strong>OUTPUTS</strong></td>
</tr>
<tr>
<td>Baseline spacecraft mass</td>
</tr>
<tr>
<td>Power system mass</td>
</tr>
<tr>
<td>Electric propulsion system mass</td>
</tr>
<tr>
<td>Propellant feeding system dry mass</td>
</tr>
<tr>
<td>Dry mass</td>
</tr>
<tr>
<td>Propellant for NSSK</td>
</tr>
<tr>
<td>Propellant for non-NSSK operations</td>
</tr>
<tr>
<td>Mass of helium pressurant</td>
</tr>
<tr>
<td>Satellite mass in GEO</td>
</tr>
<tr>
<td>Propellant mass for apogee maneuvers</td>
</tr>
<tr>
<td>Satellite mass at GTO</td>
</tr>
<tr>
<td><strong>MISSION</strong></td>
</tr>
<tr>
<td>Total firing time per arcjet</td>
</tr>
<tr>
<td>Maximum battery discharge time</td>
</tr>
<tr>
<td>ON-OFF cycles</td>
</tr>
<tr>
<td>Maximum allowable battery cycles</td>
</tr>
<tr>
<td><strong>PROPELLANT FEEDING SYSTEM</strong></td>
</tr>
<tr>
<td>Pressurant gas volume at BOL</td>
</tr>
<tr>
<td>Mass flow rate at BOL</td>
</tr>
<tr>
<td>Mass flow rate at BOL</td>
</tr>
<tr>
<td><strong>POWER SYSTEM</strong></td>
</tr>
<tr>
<td>Battery Power</td>
</tr>
<tr>
<td>Number of battery blocks</td>
</tr>
<tr>
<td>Cell capacity</td>
</tr>
<tr>
<td>Number of cells in series</td>
</tr>
<tr>
<td>Minimum discharge voltage</td>
</tr>
<tr>
<td>Maximum discharge voltage</td>
</tr>
<tr>
<td><strong>Solar array</strong></td>
</tr>
<tr>
<td>Power system</td>
</tr>
<tr>
<td>Minimum bus voltage</td>
</tr>
<tr>
<td>Maximum bus voltage</td>
</tr>
<tr>
<td>Total power at equinox</td>
</tr>
<tr>
<td>Total power at solstice</td>
</tr>
<tr>
<td>Solar cell current at equinox</td>
</tr>
<tr>
<td>Solar cell current at solstice</td>
</tr>
<tr>
<td>Cell voltage at equinox (EOL)</td>
</tr>
<tr>
<td>Cell voltage at solstice (EOL)</td>
</tr>
<tr>
<td>Number of cells in series</td>
</tr>
<tr>
<td>Number of cells in parallel</td>
</tr>
<tr>
<td>Total current per wing at equinox</td>
</tr>
<tr>
<td>Total current per wing at solstice</td>
</tr>
<tr>
<td>Total voltage per wing at equinox</td>
</tr>
<tr>
<td>Total voltage per wing at solstice</td>
</tr>
<tr>
<td>Total output power from two wings</td>
</tr>
<tr>
<td>Power margin</td>
</tr>
<tr>
<td><strong>ELECTRIC PROPULSION SYSTEM</strong></td>
</tr>
<tr>
<td>PCU</td>
</tr>
<tr>
<td>Input current initial discharge</td>
</tr>
<tr>
<td>Input current final discharge</td>
</tr>
<tr>
<td>Input voltage initial discharge</td>
</tr>
<tr>
<td>Input voltage final discharge</td>
</tr>
<tr>
<td>Average PCU efficiency</td>
</tr>
<tr>
<td>PCU efficiency at beginning of battery discharge</td>
</tr>
<tr>
<td>PCU efficiency at the end of battery discharge</td>
</tr>
<tr>
<td>Output average power from PCU</td>
</tr>
<tr>
<td>Output PCU power at beginning of battery discharge</td>
</tr>
<tr>
<td>Output PCU power at the end of battery discharge</td>
</tr>
<tr>
<td><strong>Arcjet Thruster</strong></td>
</tr>
<tr>
<td>Power to actuate at beginning of battery discharge</td>
</tr>
<tr>
<td>Power to actuate at the end of battery discharge</td>
</tr>
<tr>
<td>Average power to actuate</td>
</tr>
<tr>
<td>Arcjet at beginning of battery discharge</td>
</tr>
<tr>
<td>Arcjet at the end of battery discharge</td>
</tr>
<tr>
<td>Arcjet at beginning of battery discharge EOL</td>
</tr>
<tr>
<td>Arcjet at the end of battery discharge EOL</td>
</tr>
<tr>
<td>Arcjet at beginning of battery discharge EOL</td>
</tr>
<tr>
<td>Arcjet at the end of battery discharge EOL</td>
</tr>
<tr>
<td>Arcjet at beginning of battery discharge EOL</td>
</tr>
<tr>
<td>Arcjet at the end of battery discharge EOL</td>
</tr>
<tr>
<td>Arcjet efficiency at beginning of battery discharge</td>
</tr>
<tr>
<td>Arcjet efficiency at the end of battery discharge</td>
</tr>
<tr>
<td>Arcjet efficiency at beginning of battery discharge EOL</td>
</tr>
<tr>
<td>Arcjet efficiency at the end of battery discharge EOL</td>
</tr>
<tr>
<td>Average arcjet efficiency</td>
</tr>
<tr>
<td>Thrust at beginning of battery discharge</td>
</tr>
<tr>
<td>Thrust at the end of battery discharge</td>
</tr>
<tr>
<td>Thrust at beginning of battery discharge EOL</td>
</tr>
<tr>
<td>Thrust at the end of battery discharge EOL</td>
</tr>
<tr>
<td>Average thrust</td>
</tr>
<tr>
<td>Specific impulse at beginning of battery discharge</td>
</tr>
<tr>
<td>Specific impulse at the end of battery discharge</td>
</tr>
<tr>
<td>Specific impulse at beginning of battery discharge EOL</td>
</tr>
<tr>
<td>Specific impulse at the end of battery discharge EOL</td>
</tr>
<tr>
<td>Average specific impulse</td>
</tr>
</tbody>
</table>
Table 8 Calculated system values for 42 and 50 m/s/yr ΔV for NSSK

<table>
<thead>
<tr>
<th></th>
<th>42</th>
<th>50</th>
</tr>
</thead>
<tbody>
<tr>
<td>NSSK Mission ΔV [m/s/year]</td>
<td>42</td>
<td>50</td>
</tr>
<tr>
<td>Satellite Mass at GTO [kg]</td>
<td>2766</td>
<td>2864</td>
</tr>
<tr>
<td>Satellite Mass in GEO [kg]</td>
<td>1720</td>
<td>1781</td>
</tr>
<tr>
<td>Satellite Dry Mass [kg]</td>
<td>1416</td>
<td>1427</td>
</tr>
<tr>
<td>Propellant Mass for NSSK [kg]</td>
<td>218</td>
<td>267</td>
</tr>
<tr>
<td>Input Power to PCU [W]</td>
<td>1842</td>
<td>1906</td>
</tr>
<tr>
<td>Total Firing Time per Arcjet [hours]</td>
<td>767</td>
<td>941</td>
</tr>
<tr>
<td>On/Off Cycles</td>
<td>600</td>
<td>762</td>
</tr>
<tr>
<td>Mission Average Specific Impulse [s]</td>
<td>527</td>
<td>529</td>
</tr>
</tbody>
</table>

Table 9 Calculated system values for 976 and 1872 kg satellite dry mass

<table>
<thead>
<tr>
<th></th>
<th>976</th>
<th>1872</th>
</tr>
</thead>
<tbody>
<tr>
<td>Satellite Dry Mass [kg]</td>
<td>976</td>
<td>1872</td>
</tr>
<tr>
<td>Satellite Mass at GTO [kg]</td>
<td>1935</td>
<td>3702</td>
</tr>
<tr>
<td>Satellite Mass in GEO [kg]</td>
<td>1203</td>
<td>2302</td>
</tr>
<tr>
<td>Propellant Mass for NSSK [kg]</td>
<td>168</td>
<td>316</td>
</tr>
<tr>
<td>Input Power to PCU [W]</td>
<td>1752</td>
<td>1953</td>
</tr>
<tr>
<td>Total Firing Time per Arcjet [hours]</td>
<td>591</td>
<td>1115</td>
</tr>
<tr>
<td>On/Off Cycles</td>
<td>441</td>
<td>927</td>
</tr>
<tr>
<td>Mission Average Specific Impulse [s]</td>
<td>523</td>
<td>531</td>
</tr>
</tbody>
</table>

Fig. 6 Calculated satellite mass budget for various NSSK ΔV.

Fig. 7 Calculated system data for various NSSK/ΔV.

Fig. 8 Calculated satellite mass budget for various satellite dry mass.

Fig. 9 Calculated system data for various satellite dry mass.

provided by varying the power level. One can have, in principle, the same arcjet specific impulse and efficiency either with high power and high \( m' \) or low power and low \( m' \) at constant specific power, but, by minimizing the mass flow rate and power, one can minimize the EPS mass. The tank mass calculated by means of the tankage fractions also tend to decrease by decreasing the propellant mass. But, in general, the PFS mass can be considered fixed, due to the adoption of the EPS on a spacecraft already designed and tailored for chemical propulsion systems or, more simply, due to the availability only of qualified tankage systems. A comparison of the results obtained for different tank mass models (one function of the propellant mass and the other with constant mass) was made for the case of increasing blowdown pressure. The code predicts an improvement of just a few kilograms with the first model.
The **DOD** affects the allowed number of cycles of the battery, the cell capacity, and consequently their mass. If no restriction on the maximum number of cycles is imposed, the obvious trend is to maximize the **DOD** in order to choose a battery system with lower capacity and reduced battery mass. If restrictions are imposed on the maximum number of arcjet on-off cycles and the cell capacity is assigned, the **DOD** is adjusted accordingly.

For a given total impulse of the NSSK mission the total thruster firing time can be varied only by changing the average thrust level of the arcjet. If a restriction is imposed on the maximum firing time, the model predicts an increase both of the power level and the average blowdown pressure, in order to increase the thrust level.

Due to the discharge profile of the V/I curves provided by the battery system and to the blowdown profile of the propellant mass flow rate, the arcjet operational parameters vary greatly throughout the mission. In particular, the

### Table 10 Calculated system values for 0.4 and 0.8 battery DOD

<table>
<thead>
<tr>
<th>Battery Depth of Discharge</th>
<th>0.4</th>
<th>0.8</th>
</tr>
</thead>
<tbody>
<tr>
<td>Satellite Dry Mass [kg]</td>
<td>1538</td>
<td>1364</td>
</tr>
<tr>
<td>Satellite Mass at GTO [kg]</td>
<td>3044</td>
<td>2700</td>
</tr>
<tr>
<td>Satellite Mass in GEO [kg]</td>
<td>1893</td>
<td>1679</td>
</tr>
<tr>
<td>Propellant Mass for NSSK [kg]</td>
<td>261</td>
<td>232</td>
</tr>
<tr>
<td>Input Power to PCU [W]</td>
<td>1899</td>
<td>1863</td>
</tr>
<tr>
<td>Total Firing Time per Arcjet [hours]</td>
<td>920</td>
<td>818</td>
</tr>
<tr>
<td>On/Off Cycles</td>
<td>743</td>
<td>648</td>
</tr>
<tr>
<td>Mission Average Specific Impulse [s]</td>
<td>529</td>
<td>526</td>
</tr>
</tbody>
</table>

### Table 11 Calculated system values for 18-8 and 22-12 blowdown pressure range

<table>
<thead>
<tr>
<th>Feed System Inlet Blowdown Range [bar]</th>
<th>18-8</th>
<th>22-12</th>
</tr>
</thead>
<tbody>
<tr>
<td>Satellite Dry Mass [kg]</td>
<td>1427</td>
<td>1447</td>
</tr>
<tr>
<td>Satellite Mass at GTO [kg]</td>
<td>2824</td>
<td>2872</td>
</tr>
<tr>
<td>Satellite Mass in GEO [kg]</td>
<td>1756</td>
<td>1786</td>
</tr>
<tr>
<td>Propellant Mass for NSSK [kg]</td>
<td>242</td>
<td>248</td>
</tr>
<tr>
<td>Input Power to PCU [W]</td>
<td>1876</td>
<td>2124</td>
</tr>
<tr>
<td>Total Firing Time per Arcjet [hours]</td>
<td>855</td>
<td>760</td>
</tr>
<tr>
<td>On/Off Cycles</td>
<td>682</td>
<td>687</td>
</tr>
<tr>
<td>Mission Average Specific Impulse [s]</td>
<td>528</td>
<td>525</td>
</tr>
</tbody>
</table>
Concluding Remarks

The results of the study described in the paper demonstrate the capability of the model and the associated computer code to trade-off the design characteristics of the various elements for maximizing the performance of the system as a whole. The discharge profile of the V/I curves provided by the battery system, the blowdown profile of the propellant flow rate, and the other arcjet operational parameters can also be calculated throughout the operational life of the spacecraft. The code is very flexible and can quickly be adapted to a wide range of missions, spacecraft configurations and propulsion systems by simply reformulating the relevant mission/spacecraft assumptions. Within this study, the results provided for a wide range of satellite dry masses yield an optimized EPS power in the neighborhood of the 1.8 kW nominal power envisaged by the present ASI system development programme. The calculated arcjet firing time and ON/OFF cycles are also consistent with ASI's reference values. As expected, in the absence of restrictions on the firing time and the ON/OFF cycles, the optimal trend is to reduce the propellant mass flow rate to its minimum value in order to obtain a higher specific impulse and save propellant weight. Similarly, the DOD tends to upper allowable value, thereby reducing the weight of the batteries and the spacecraft dry mass.

Acknowledgments

The present activity has been sponsored by the Italian Space Agency under contract No. 165-AF-1991 with Mrs. M.F. Rossi as contract monitor. Centrospazio has been involved as subcontractor of BPD Difesa e Spazio. The contributions of the students R.I. Calvo of the Universidad Politécnica de Madrid and S. Carreca of the Università di Pisa during the preparation of their laurea theses on this subject are also acknowledged. Finally, the authors would like to express their gratitude to Miss P. Nugent and Mr. H.M. Oest for their help in the preparation of the manuscript.

References


"Verniolle, J., Jamin, T., and Paugam, D., "SAFT Second Generation Ni-H, Cell in the 40Ah-100Ah Range for Geostationary Application," Proceedings of the