ADVANTAGES OF A CONTINUOUS THRUST STRATEGY FROM A GEOSYNCHRONOUS TRANSFER ORBIT, USING HIGH SPECIFIC IMPULSE THRUSTERS

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Abstract

Since mid-1997, Electric Propulsion is used aboard western Geostationary satellites for the North/South Station Keeping. This first step into the field of the High specific impulse propulsion will be followed by a generalisation of its use in order to cover all the propulsion functions aboard satellites, including the orbit raising up to the geosynchronous orbit.

The paper presents first a general consideration about the High Specific Impulse Thrusters as the Stationary Plasma Thrusters that are characterised mainly by their low thrust level : less than 0.5 N for a typical input power of 5 kW.

A second part deals with thrust strategies from an elliptical starting orbit to a circular one. Advantages of continuous thrust strategies are listed. In the presented study a new parameter is introduced: the Specific Impulse of the Manoeuvre, which combine thrust level, duration of the manoeuvre and initial mass of the space vehicle. Its expression, in speed units, is also related to the ideal velocity. This parameter is used to simplify the comparison between continuous thrust strategies. Optimisations of this parameter are shown.

A third part deals with a selection of advantages of an "All electric propulsion satellite", mainly a very substantial mass gains up to 1 000 kg for a 2 000 kg satellite in geosynchronous orbit, a complete deletion of any chemical system aboard the satellite and the subsequent deletion of toxicity and hypergolicity risks, the duration of the transfer manoeuvre may be as short as 45 days or 90 days when electrical power allowed to the propulsion is respectively 20 or 10 kW. The drawbacks, like the number of passages throughout the Van Allen belts, are finally discussed.

Key words: Electric Propulsion, Continuous Orbit Transfers.

Introduction

What are the **High Specific Impulse Thrusters**? The answer to this simple question is generally:

- **Electric Propulsion**, because the basic principle of the electric propulsion is to convert the electrical power aboard the spacecraft (which do not generally uses any mass) into mechanical power to the expelled mass and to thrust the spacecraft by reaction. In such conditions, it is clear that high levels of performance can be achieved with electric propulsion. It is sufficient to increase the electrical power transformed for having a higher mechanical power for the same amount of expelled mass: this is the performance that we call Specific Impulse.

The second point to highlight after the first question is that such High Specific Impulse Thrusters are always thrust limited, that means:

- **Low Thrust**, because the power aboard the spacecraft is always limited. We shall recall that a chemical 400 N thruster produce a mechanical power of 1 200 000 W (1.2 MW !). If such thruster were an electric one (same thrust, with higher specific impulse) the electrical power needed to feed the thruster would be 6.4 MW!

The High Specific Impulse Thrusters are then clearly (for the near term, for the mean term and probably also for the long term) an electric and a low thrust propulsion.

Now we shall introduce the Electric Propulsion. This kind of propulsion can be divided now into three generic classes corresponding to the acceleration process employed to expel the propellant[1] and the available level of performance (specific impulse):

- Electrostatic thruster, generally called "Ion Bombardment Thruster" or "Radio-frequency Ion Thruster" with a Specific impulse in the range 2 500-4 000 s (25 000 - 40 000 N.s/kg),
- Stationary plasma thruster (SPT or PPS in France), generally called "Plasma Thruster" or "Hall thruster" with a Specific impulse in the range 1 000-2 600 s (10 000 - 26 000 N.s/kg),
- Electro-thermal thruster generally called "Arcjet thruster" or "Resistojet" with a Specific impulse in the

range 300-1 000 s (3 000 - 10 000 N.s/kg). This last kind of propulsion is not a pure electric propulsion because it may use also the chemical energy of the propellant (in the case of the use of the hydrazine mono-propellant).

The firsts two classes of thrusters employ now the Xenon gas as propellant that is a clean, non toxic, easily storable and easily usable for space product, without specific tension surface devices because of the storage in gaseous supercritical state. The arcjet considered in this paper is fed with the common hydrazine mono-propellant.

The purpose here is to propose a simplified analysis showing the advantages of the continuous thrust strategy for the most important commercial orbit transfer, the transfer between GTO (Geostationary Transfer Orbit) and GEO (Geostationary Earth Orbit).

General Relations, Thruster Comparisons Principles

Before having any discussion, it is needed to describe the fundamental relationships and characteristics of the electric propulsion.

The first useful equation is coming from the specific impulse definition (the amount of impulse given by a unit of expelled propellant mass):

$$Isp=F.dt/dm$$
$$Isp=F/q$$

where

- F in Newton is the axial thrust,
- q is the total "propellant flow" of the thruster including anode, cathode and neutraliser propellant consumption, in kg/s,
- Isp in N.s/kg. With that definition, Isp can also be considered as an average velocity V of the propellant particles along the thruster axis in m/s. *An other usual definition is* $Isp=F/(q.g_0)$ *with Isp in seconds,* $g_0=9.80665\ m/s^2$.

A very similar equation is the thrust equation when the pressure of the propellant particles in the exit section of the thruster can be neglected (this is the case for electric thrusters):

$$F=q.V.$$

where

- q is the total "propellant flow", as previously seen, in kg/s,
- V is an average velocity of the propellant particles along the thruster axis in m/s. Actually V=Isp.

The last equation is coming from the principle of energy conservation. Electric thruster can be considered as device that converts electrical power into mechanical power. To do that, of course, a certain quantity of mass must be expelled from the thruster at a high velocity, and in a privileged direction (generally the thruster axis).

For a pure electric thruster, the propellant is chemically inert so that only electrical power (P_e) is converted into mechanical power (or the rate at which the kinetic energy leaves the vehicle with the propellant), taking into account a certain value of efficiency. For an electro-thermal thruster, the produced mechanical power is coming only in part from the applied electrical power, the other part is coming from the chemical energy of the propellant used (hydrazine).

We can write in every case:

$$\eta.\ P_e = 1/2.q.\ V^2$$

where

- q is the total "propellant flow", as previously seen, in kg/s
- V is an average velocity of the propellant particles along the thruster axis, as previously seen, in m/s
- P_e is the electrical power at the input of the thruster Power Processing Unit (PPU), that is, this power P_e is directly the electrical power that the Power Conditioning Unit (PCU) of the satellite must provide to the thruster system, P_e in Watt. *Generally the power is given at the input of the thruster, after the transformations in the PPU. Such power is very similar to* P_e, *because the PPU electrical efficiency is over 90%.*
- η is the efficiency of the whole thruster system for the process of the power conversion. Actually, it is better to understand the above equation as an axiom of the definition of the whole efficiency of the electric thruster system.

Those equations are combined to elaborate a very interesting relation between the specific power consumption (electrical power needed to produce a unit of thrust also called the Power-to-Thrust[1,3] ratio $\mathbf{P_e/F}$]) of the thruster type and the specific impulse in N.s/kg:

$$\mathbf{P_e/F=Isp/(2.\eta)} \qquad \mathbf{(1)}$$

The data given for any thrusters are sufficient to compute the whole efficiency η of the thruster type. Generally that efficiency depends on the size of the thruster and on the operating conditions (mainly the discharge or accelerating potential), but it doesn't vary too much for a thruster type. It is considered as a characteristic for a qualified thruster type when working in its qualified operating domain.

For arcjet, this efficiency is computed to be $\approx$ 30%, while for plasma[10] (SPT Fakel, SEP η=40-50%) and ion thrusters (RIT 10, DASA and XIPS, Hughes $\eta \approx 44$-50%; XX and T5 $\eta \approx 55$-60%) the efficiency is about 50 %.

As a first order approximation, equation (1) implies that the specific power consumption is equal to the specific impulse[16]:

$$P_e/F = Isp \quad ! \qquad (2)$$

with Isp in N.s/kg , P_e Watt, F Newton

The relations (1) or (2) indicate that:

- The specific impulse is directly proportional to the electrical power by unit of thrust.

To produce the same thrust, the higher the specific impulse of a type of electric thruster is, the higher the electrical power need is.

- In other words, with the same electrical power, the thrust produced is inversely proportional to the specific impulse.

The first assertion means that the electrical power delivered by the electrical generation system of the satellite can be changed depending on the thruster to be used. This could be achieved with:

increase of solar panels dimensions,

increase of solar electric power transfer assemblies' mechanisms

increase of the power conditioning unit and batteries.

However, such approach is purely theoretical because each system must be completely qualified and those qualifications are most of the time more expensive and have longer development schedule than the propulsion system one.

In fact, the most probable strategy (for specific applications as non-commercial direct broadcast satellites or scientific spacecraft's) would be to decrease the power capacity of the spacecraft when using the lower Isp thruster. This shows that for a iso-thrust comparison, despite their lower intrinsic performance, medium high specific impulse thrusters can be very competitive with very higher Isp thruster, because the mass saving at system level (solar panels, batteries,...) can compensate largely the increase of propellant used.

The second assertion is practically the most important one, because it is almost always in those terms that the comparisons between thruster types are done (for commercial direct TV satellites). The electrical power of a satellite is primarily devoted to the payload, and only secondly, to the electric propulsion, if and when the whole satellite system takes advantage of that.

Historically, we can correlate this point of view with the increase of the electrical power of the last generation of satellites. The mean value for large commercial direct TV Geostationary satellites was 3 000 W in the beginning of the 90's, 5 000 to 8 000 W now and up to 15 000 W or 20 000 W for the beginning of the new millennium[4]. This electrical power is first used for an increase of the payload that is the only source of economical profits in a satellite system, and the increase of it, increases the potential profits and the competitiveness. Because those foreseen levels of electrical power are achievable, we are at the beginning of the age of the electric propulsion for satellites due to its well ad-equation with the needs of the payload.

The conclusion of this general introduction is that, **applied to commercial direct TV satellites, a valid comparison between electric thruster types should assume first that the same level of electrical power is available for each kind of thruster.** As consequence, the thrust produced by each type of electric thruster is reduced when the specific impulse is increased. Because the duration of an orbital transfer is proportional to the thrust, this is a very important penalty for the very high specific impulse thruster applied to telecom satellites. **While applied to non-commercial spacecraft, a valid comparison should assume first that the same thrust is produced by each kind of thruster.** As consequence, the duration is always kept constant and the electrical power produced by the satellite is reduced when the specific impulse of the thruster is not so high. Because the mass of the whole electrical system of the satellite is decreased (by e.g. a factor 2) there is a very important penalty for the very high specific impulse thruster applied to non-telecom satellites.

Thrust Strategies for the Transfer to the Geostationary Earth Orbit

The main drawbacks of High Specific Impulse Thrusters is their low thrust and the long duration of an orbit transfer.

Lots of strategies[2,6,9,11,12] have been investigated before finding a valid continuous thrust strategy between GTO and GEO which can minimise the Orbit transfer duration.

First studies were based on the thrust arcs strategy around the apogee (similar to the high thrust chemical propulsion "apogee burns"). The results indicates a slight increase of the Delta-V needs (or propellant mass) wrt. the corresponding high thrust strategy and the duration of the transfer could be considered as prohibitive. A computer output of this apogee-centred thrust arcs is shown **fig. 1**.

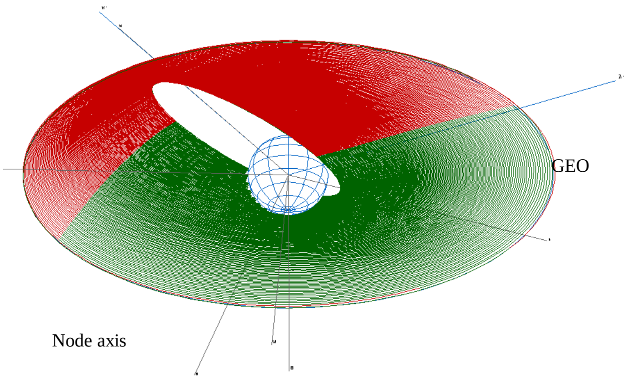

Fig. 1: Inclined initial orbit (Sub-GTO 28.5°) to GEO with Apogee-centred thrust arcs. Note that the effects of the large thrust arcs with a large increase of the perigee altitude, and a slight increase of apogee altitude. The total number of orbits is 295.

Second studies were based on the integration of the super GTO strategies into a thrust arcs strategy, after a suggestion of Mr. Marcel Pouliquen[5]. The results were that the duration of the transfer could be decreased with burns around apogee and burns around perigee. The increase of the Delta V needed for achieving the orbit transfer were more significant (**fig. 2**)

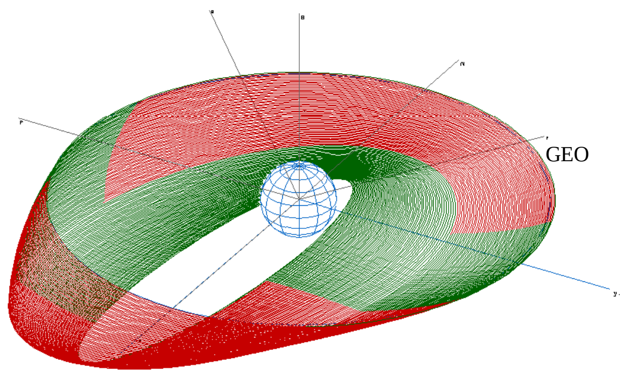

Fig. 2: Inclined initial orbit (Super-GTO 55 000 km and 28.5°) to GEO with Apogee and Perigee-centred thrust arcs. The total number of orbits is 178.

Finally the continuous thrust strategy (**fig. 3**) has been discovered in order to decrease at a maximum the duration of the transfer without taking care of the slight increase of propellant mass used during the orbit transfer manoeuvre. Heavy optimisation techniques conducted by S. Geffroy[17] have shown similar results as the proposed strategy. This strategy has been compared with the strategy given by Spitzer[9,12]. Lots of practical advantages have been founded in favour of our strategy[13,14,16].

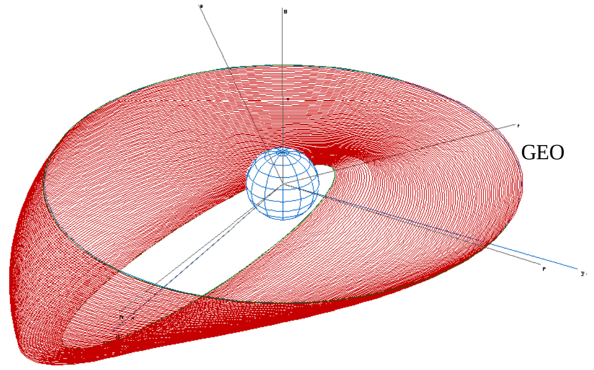

Fig. 3: Continuous Orbit Transfer from an Inclined initial orbit (Super-GTO 60 000 km and 28.5°) to GEO. The total number of orbits is only 91.

The hypothesis and a comparison between those different strategies is shown in **table 1**.

Table 1: **Hypothesis and comparison between strategies.** The shortest duration is given with a continuous strategy.

	Fig. 1 Apogee-centred thrust arcs	Fig. 2 Apogee and Perigee-centred thrust arcs	Fig. 3 Continuous Orbit Transfer
Initial alt. (km)	185x26 500	185x55 000	185x60 000
Initial incl. (°)	28.5	28.5	28.5
Initial Mass (kg)	3 000	3 000	3 000
Thrust (N)	0.9	0.9	0.9
Isp (s)	1 600	1 600	1 600
In plane thrust orientation	Local horizontal	Local horizontal	Local horizontal
Out-of-plane thrust orientation	± 41.5°, constant	± 33°, constant	± 26°, constant
DeltaV (m/s)	2 540	2 120	2 475
Theoretical impul. minimum deltaV	2 026	1 678	1 657
DeltaV efficiency (impulse/real)	80%	79%	67%
Launcher deltaV increase (m/s)	- 191	+ 233	+ 264
Final Mass (kg)	2 552	2 621	2 562
Duration (days)	148	113	**89**
Number orbits	295	178	**91**

Because the initial orbits of table (1) are not the same, the launcher increase of performance required is given algebraically wrt. the GTO transfer (185 x 36 000 km, 28.5°). The Out-of-plane thrust component is also given algebraically in table 1, because the change of sign at the radius perpendicular to the ascending node axis. The Delta V efficiency is simply the ratio between the minimum impulse delta V required for the transfer and the real delta V needed.

This simple historical shows that the duration of the orbit transfer between GTO ant GEO has been reduced by more than 60 % (at same thrust level). This is naturally the counterpart of the loss of the manoeuvre efficiency (67% instead of 80%).

One should note also that the total number of orbits needed for the transfer has also been reduced by more than a factor 3. This is a very important result, because the radiation of the Van Allen belts can be minimised when the number of orbits crossing the proton belts are minimised. This number can be considered as directly in relation with the total number of orbits.

Continuous Orbit Transfer.

The low thrust is continuously provided to the satellite. The thrust orientation strategy (in the local orbital plane) is quasi-optimised when considering the three following general phases:

- **First Phase**: Increase of perigee and apogee altitudes. The best results are obtained for the following thrust orientation. When the satellite is around the perigee, the thrust is "tangent to the trajectory" (thrust in-plane component collinear to the velocity). When the satellite is around the apogee the thrust is in the local horizontal plane (perpendicular to the orbit radius vector). The term "around" refer to the whole half ellipses centred to the apogee or perigee. The transition between perigee and apogee orientation strategy is located at the ellipse minor axis. With such thrust strategy, this phase is curiously characterised by an increase of apogee altitude higher than the perigee one. Another good thrust orientation is to have the thrust always is in the local horizontal.
- **Second Phase**: Increase of perigee altitudes and decrease of apogee altitudes until the circular orbit is reached. The best results are obtained with a similar strategy as in phase one. However, a major change in the orientation of the thrust around the perigee is to be anti-collinear to the velocity. Another good thrust orientation is to have an in-plane component oriented continuously wrt. an inertial direction.
- **Third Phase**: Decrease of perigee altitudes and decrease of apogee altitudes until the GEO is reached. This is a typical spiralling strategy between two circular orbits. The thrust is in the local horizontal plane.

The first phase can be skipped if the initial apogee altitude is very high or if the initial orbital period is higher than 24 hours.

The third phase can also be skipped if the transition between phase 1 and phase 2 is produced when the orbital period is equal to 24 hours.

Orientation changes are provided thanks to a thrust component out of the orbital plane. The absolute value of the out-of-plane component is held constant during the whole orbits, although the sign of that component is reversed twice per orbit when the satellite reach the orbital radius perpendicular to the ascending node axis[2,...18].

Figure 4 and **Figure 5** hereunder show the tri-axial output of the software. The thrust vector is shown in each point of the trajectory. One can easily see the various zones and transition points. One can distinguish so that the different thrust orientation strategies

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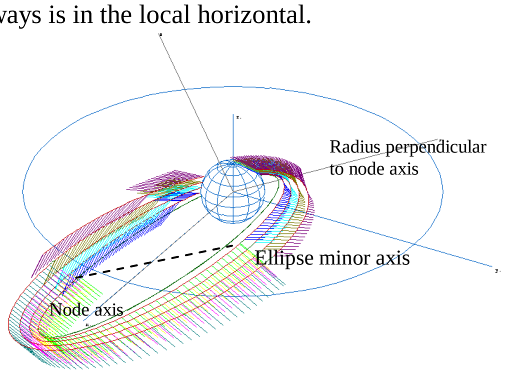


Fig. 4 Orbit Transfer from inclined orbit, First Phase (2 transition zones: one at the ellipse minor axis for Perigee and Apogee altitude increase, one at the radius perpendicular to the orbital node for the inclination change)

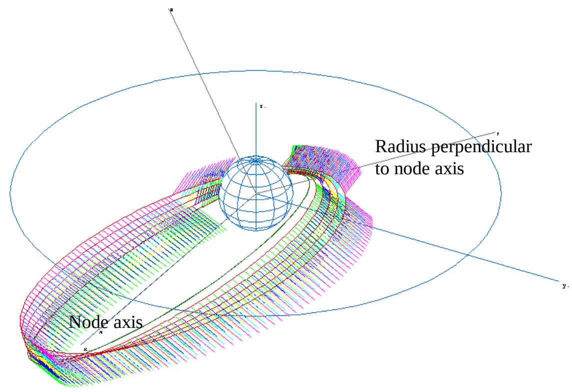


Fig. 5 Orbit Transfer Second Phase (in-plane component inertial) (1 transition zone, due to inclination change, at the radius perpendicular to the orbital node)

Specific Parameter Definition

The goal of the new definitions is to characterise the orbit transfer as simply as possible. The main characteristics of the orbit transfer are:

- The duration of the transfer **Δt**,
- The mass of propellant used,
- The number of perigee crossing the Van Allen belts,

for a satellite, having an initial mass at launch $\mathbf{M_i}$, powered by a given propulsion system with a specific impulse of the thrusters $Isp_{thruster}$ and a total given thrust **F**.

The first simple parameter has been called the **Specific Impulse of the Manoeuvre:**

$$\mathbf{Isp_{manoeuvre} = F.\Delta t / M_i} \qquad \mathbf{(3)}$$

with F Newton, Δt s, M_i kg, $Isp_{manoeuvre}$ in m/s.

This parameter allows to directly compute the duration of the orbit transfer when the thrust **F** and the initial mass of the satellite $\mathbf{M_i}$ are known.

Because the thrust is continuously applied to the satellite, the mass of propellant used during the orbit transfer ($\mathbf{M_e}$) is simply equal to the mass flow rate time the duration of the orbit transfer.

One shall mention that $\mathbf{Isp_{manoeuvre}}$ is almost equal to the Delta V of the orbit transfer. This is because the specific impulse of the thrusters $\mathbf{Isp_{thruster}}$ is high and then the initial mass $\mathbf{M_i}$ and the final one $\mathbf{M_f}$ are very close together.

$$Delta\ V = Isp_{thruster}.\mathrm{Ln}\left(M_i / M_f\right)$$

$$Delta\ V \approx Isp_{thruster}.\ \mathrm{M_e} / \mathrm{M}_f \quad \text{with} \quad Isp_{thruster} = \mathrm{F.\Delta t} / \mathrm{M}_e$$

$$Delta\ V \approx F.\Delta t / M_f$$

$$Delta\ V \approx F.\Delta t / M_i = Isp_{manoeuvre}$$

Because the $\mathbf{Isp_{manoeuvre}}$ is the order of the Delta V to be produced for the orbit transfer, the adopted definition is very interesting for electric propulsion continuous manoeuvres. Its minimisation lead to minimise in the same time, the duration of the transfer and the minimisation of the propellant mass needed for the transfer.

The second simple parameter has been introduced for having simplified comparison between orbit transfer strategies, from a point of view of the Van Allen Belts radiation dose. The high density (fluence > 20 000 protons per cm² per second) and high energy (>3 MeV) protons Van Allen belt are centred at an altitude of 5 000 km. The belts begin at an altitude of 2 000 km and end at an altitude of 7 000 km. So that each transfer strategy is characterised by its number of perigee crossing the Van Allen belts (i.e., having a perigee altitude higher than 7 000 km or 8 000 km with a margin of 1 000 km).

This second parameter has been called the **Specific Perigee Number.** It is a normalisation of the number of revolution needed by a satellite, propelled by a thrust **F,** with an initial mass $\mathbf{M_i}$, to escape the Proton Van Allen belts (i.e. for having a perigee higher than 8 000 km).

$$\mathbf{Perigee_{Specific} = Number_{perigee\ crossing}\ .\ F/\ M_i}$$

with F Newton, M_i Ton, $Perigee_{Specific}$ in #N/Ton.

The $\mathbf{Perigee_{Specific}}$ Number allows to directly calculate the $\mathbf{Number_{perigee\ crossing}}$ of the orbit transfer when the thrust **F** and the initial mass of the satellite $\mathbf{M_i}$ are known.

It is also, of course, the number of orbits described by a satellite of one Ton propelled by a thruster of one Newton needed to reach an orbit having a perigee altitude higher than 8 000 km. In order to have realistic values for the $\mathbf{Perigee_{Specific}}$ Number, it is preferable to compute it with the mass expressed in Ton.

This Number depends firstly on the initial orbit transfer altitude (GTO, Sub-GTO or Super-GTO 60 000 km or Super-GTO 90 000 km) and depends secondary on the inclination change to be done during the transfer. It is of course faster to escape the Van Allen belts when no inclination change are produced during this first part of the transfer.

Main results of a parametric study

The parametric study takes into account:

- an initial elliptical orbit with a given inclination wrt. the equatorial plane,
- the final Geostationary orbit.

In order to have an analytic view of the orbit transfers, the results of the computations have been summarised thanks to a multiple regression analysis techniques, in the case of the low thrust electric propulsion (< 2 N).

For the first phase, for a given initial orbit, a given final perigee altitude and a given total inclination change, a first regression allows the estimation of the eccentricity and the apogee altitude at the end of the first phase. A second regression allows the estimation of the DeltaV efficiency (lowest impulse Delta V divided by the real Delta V needed).

The lowest impulse DeltaV is given by a classical Hohmann transfer, as shown fig. 4, with the inclination change totally produced during the apogee impulse.

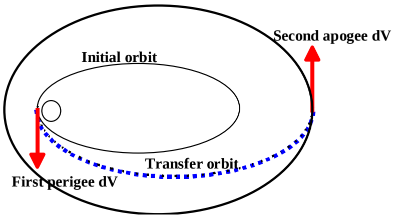


Fig. 4: First phase of a continuous transfer: Equivalence with the classical Hohmann impulse transfer.

For the second phase, for a given initial orbit and a given total inclination change, a unique regression allows the estimation of the DeltaV efficiency. For that case, the lowest DeltaV is given by the classical Bi-elliptic transfer, as shown fig. 5, with the inclination change totally produced at the first apogee impulse.

The main results of the regression analysis are shown in **table 2**. Various inclination changes have been considered from 0° to 30°, according to the latitudes of the main launch facilities. The results are first given in terms of **Isp** $_{\textbf{manoeuvre}}$. For information, the values of the real DeltaV needed are shown in brackets (in case of use of a thruster having a specific impulse different from 1 600 s).

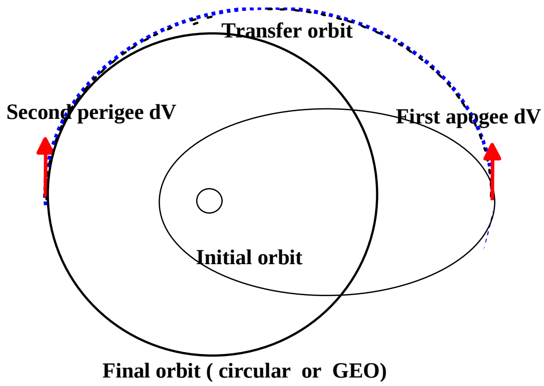


Fig. 5: Second phase of a continuous transfer: Equivalence with the classical Bi-elliptic impulse transfer.

On **figure 6** are shown the results dealing with the number of perigee crossing the Van Allen belts during the orbit transfer. Obviously, the inclination of the thruster wrt the local orbital plane (between 0 and 30°) is not of prime importance for this number. Thus, **table 3** doesn't take into account any inclination change.

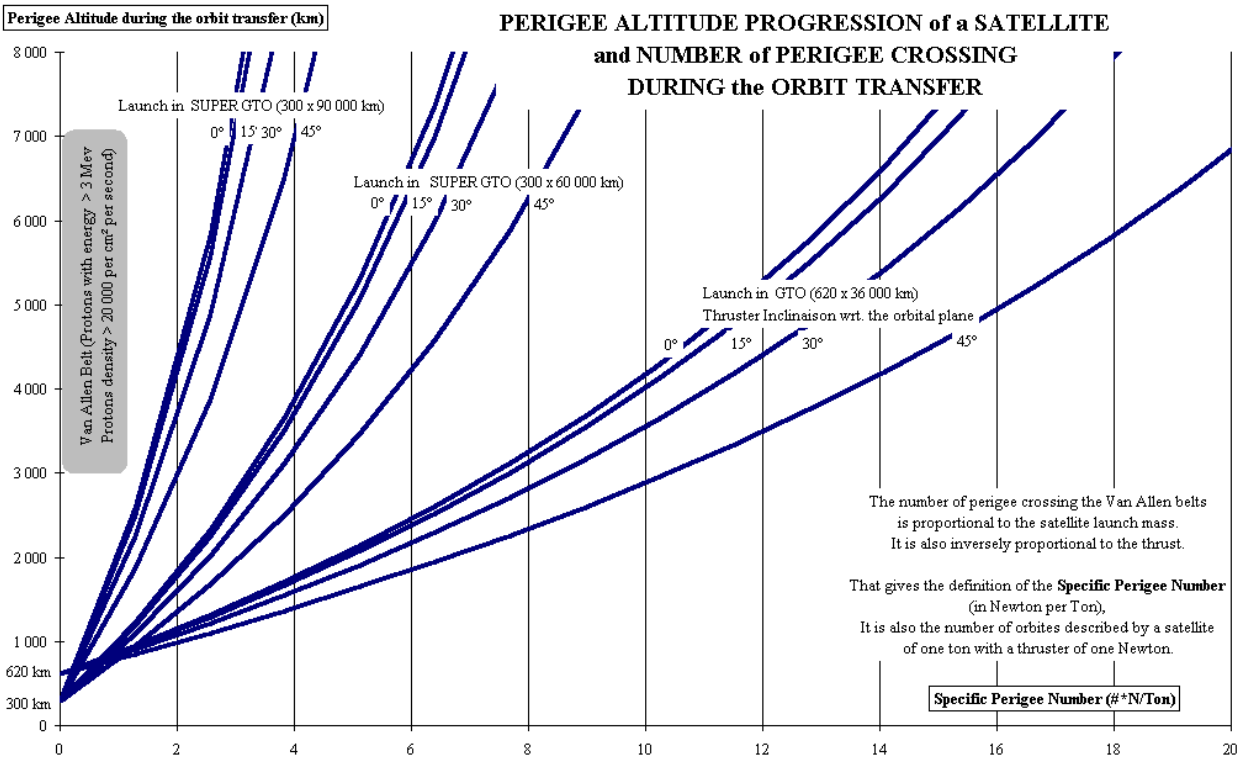
Table 2: Isp $_{\textbf{manoeuvre}}$ **in m/s for different initial orbits and different initial inclination.** A Super GTO with an apogee altitude of 60 000 km is about optimum whatever the inclination change is. Isp $_{manoeuvre}$ $=F.\Delta t / M_i$

Inclination change **Isp** $_{\textbf{manœuvre}}$ **and (real deltaV)** Initial orbit	0°	10°	20°	30°
Super GTO 200 x 50 000 km	2 045 (2 191)	2 120 (2 278)	2 290 (2 475)	2 530 (2 759)
Super GTO 200 x 60 000 km	**1 950** (2 082)	**2 010** (2 151)	**2 150** (2 312)	**2 350** (2 546)
Super GTO 200 x 70 000 km	1 950 (2 082)	2 000 (2 139)	2 120 (2 278)	2 290 (2 475)

Table 3: Perigee $_{\textbf{Specific}}$ **Number in N/Ton for three different initial orbits.** A Super GTO with an apogee altitude of 60 000 km is characterised by only 23 perigee crossing the Van Allen belts (for a 3 Tons satellite and 0.9 N thrust).

Initial orbit	**Perigee** $_{\textbf{Specific}}$ Number (N/Tonne)	**Number** $_{\textbf{perigee crossing}}$ **in the Van Allen belts** Satellite mass 3 Ton, Thrust 1 N
GTO 620 x 36 000 km	≈ 17	≈ 57
Super GTO 300 x 60 000 km	**≈ 7**	**≈ 23**
Super GTO 300 x 90 000 km	≈ 3,5	≈ 12

Fig. 6: Number of perigee crossing the Van Allen belts.
Fast escape of the proton Van Allen belts are achieved with Super GTO initial transfer orbits!



The conclusion of the parametric study shows the very interesting properties of the proposed continuous transfer strategy between Super-GTO 60 000 km and GEO. **The all electric orbit transfer duration can be as short as 77 days (from Kourou) or 90 days (from Cape Canaveral)**. The mass of propellant needed for the transfers are respectively 381 kg and 444 kg. Those data are valid when considering a satellite initial mass of 3 000 kg and a thrust 0.9 N, having a specific impulse of 1 600 s (i.e. 16 kW/N) and an electrical power of 14.4 kW. In that case, **the number of perigee crossing the protons Van Allen belts is only 23**. This is also a main advantage of the proposed Super GTO transfer with a continuous low thrust strategy.

Transfer from GTO and SUPER-GTO to GEO

It is clear that the high efficiency of electric propulsion is very interesting for the orbit transfer compared to the classical chemical propulsion. Depending on the latitude of the launch facilities, the advantage, between 1 200 and 1 600 kg, is in favour of electric propulsion for a 3 tons satellite. (i.e. a Chemical High thrust satellite transfer requires an initial mass of 4 200 to 4 600 kg for delivering in GEO, the same mass as after the electric transfer).

Even when the increase of performance required for the launch into Super GTO instead of GTO is taken into account, the net advantage is still comfortable between 780 and 1 150 kg. (The launcher decrease of performance is quoted at iso-launch price. In the case of Ariane 5, the penalty is 13 % in terms of mass, for a launch in Super GTO with a 60 000 km apogee instead of a launch in the classical GTO with a 36 000 km apogee).

Of course, when comparing an all electric satellite, having a launch mass of 3 000 kg, with an all chemical satellite for a 15 years mission, the chemical one should have a launch mass between 5 200 an 5 700 kg! The net advantage is 1 750 to 2 250 kg (because the accumulation of the electric orbit transfer benefit with the electric on-orbit station keeping benefit)!

The next particularly interesting study is about the comparison of GTO to GEO transfer using SPT with respect to Ion bombardment thrusters. Such orbital transfers are considered as interesting when only all-electric propulsion is used. The main arguments are related to the overall removal of the chemical propulsion system aboard the satellite, the deletion of subsequent toxicity, hypegolicity or pollution risks of liquid chemical propellant[8], the reduced cycle of manufacturing and integration of the satellite (without use of any simulated liquid propellant during vibration tests for example). The reduction of the initial mass leads also to other benefits as simpler ground support with a smaller volume of satellite. This induces also a better payload accommodation in the spacecraft volume, particularly for heat pipes.

The unfavourable aspects of electric propulsion are related to the duration of the whole manoeuvre, because of the low thrust of electric thrusters. This negative aspect can be partially compensated with the reduced cycle of manufacturing and integration of the all electric satellite, and with the spacecraft outgassing time. Other common drawbacks of electric propulsion are the damages to the satellite caused by the sputtering[7]. If the thrusters are fixed to the anti-earth face of the satellite, such damages are really non-significant.

To minimise the residual duration of the transfer, a continuous thrusting strategy, as exposed heretofore, is considered from the launcher injection into Super GTO to the final Geostationary earth orbit.

A first comparison between SPT and ion bombardment thruster has been shown in previous studies[16], in the case of an all electric transfer. Obviously, due to equation (2), the very high specific impulse of ion thruster (3 000 s) imply that at same input power as for the SPT (1 600 s) the thrust of ion propulsion is twice the SPT one. Thus, 90 days with SPT means ≈168 days with ion propulsion... This comparison shows the real advantage of SPT propulsion over the ion propulsion, whatever the reduction of propellant mass are.

In order to overcome this situation for ion propulsion, the most interesting strategy is a combination of chemical propulsion and electric propulsion (combined propulsion). Such strategy has been patented by Hughes[9,12]. It seems well adapted with XIPS (ion bombardment type thrusters system). It allows a significant reduction of the number of passages throughout the Van Allen belts (especially the protons' belts), but the advantages of the high specific impulse are naturally reduced.

When comparing, at iso-platform mass, such combined chemical/electric propulsion systems to an all electric system with SPT, one can found that the total manoeuvre duration's are very comparable and the mass at take-off is always largely lower with an all electric SPT satellite. This is the consequence of the higher performance of electric propulsion compared to the chemical one.

To have an economical valid comparison one shall take into account, once more, the higher energetic launch needed for a Super-GTO compared to a simple classical launch in GTO. This can be achieved when comparing the ratio of launcher capability rather than the mass. Such ratio is equivalent to the percentage of the cost of the launch. Even this economical comparison leads to a similar situation on those two kinds of systems. One can conclude that: for a same launcher cost, one can launch a combined Chemical/Electric ion bombardment satellite or an all electric SPT satellite that leads to transfer within about the same duration about the same mass in GEO.

The main drawback of the combined Chemical/Electric ion bombardment satellite is the presence of the chemical propulsion system. In addition the initial mass of the satellite is larger. Moreover, the duration of the manufacturing and integration process cannot be reduced as when the chemical propulsion system is removed (this duration seems to be larger, in fact, when it is needed to integrate two kinds of propulsion system).

The conclusion is that the overall advantages seem to be in favour of a medium high specific impulse system like SPT, instead of a very high specific impulse. This is due especially because it allows the complete removal of any chemical propulsion system aboard a commercial Geostationary satellite. The drawback of the all electric propulsion with SPT deals with the Van Allen radiation degradation. In the first part of this paper, it has been shown that when the initial orbit is carefully chosen, the number of perigee crossing the Van Allen belts can be as low as only 23. This figure is less than three time higher than the current design criteria of the commercial Geostationary satellites. There are no critical issues to overcome this drawback. One shall add that for other orbit transfer strategies, as the one exposed in fig 1, the number of Van Allen belts perigee crossing are about 160. This shows the large advantage of the proposed orbit strategy.

Conclusions

This paper shows the importance of the choice of the launcher transfer orbit injection. In order to minimise the number of perigee crossing the Van Allen belts, the best choice would be a very high Super GTO apogee. On the other hand, the higher the apogee is, the higher the increase of performance required to the launcher is.

The choice proposed here is based on the minimum duration of the whole transfer with electric propulsion.

This choice is a medium Super GTO apogee altitude (60 000 km). Only 23 perigee crossing the Protons Van Allen belts are needed. The corresponding continuous thrust strategy is analysed in details. This choice must be also based on the use of a medium high specific impulse electric thruster (SPT).

Compared to the other kind of propulsion, SPT propulsion shows the higher advantages. High mass savings with respect to the chemical propulsion (more than 780 kg for a 3 ton electric satellite). SPT allows a complete deletion of any chemical subsystem when compared to the ion propulsion.

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