AUTOMATIC INTERPLANETARY MISSIONS. OPTIMISATION OF A MISSION WITH LOW THRUST STRATEGY.

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Abstract

In view of the exploration of the solar system, a lot of automatic missions could be proposed. The propulsion system of the interplanetary vehicle is a key element of the system, impacting the launch mass, the useful mass, the mission duration and the vehicle architecture. The mission scenario and especially the trajectory optimisation are an other key element of the success of the mission, strongly linked to the choice of the propulsion system of the vehicle. Therefore an optimisation process, at least, between mission strategy and propulsion system is very helpful in pre-study phase, in order to compare different solutions for a given mission.
After a brief review of the main automatic interplanetary missions (past and present) and the propulsion architecture of the relevant space vehicles, this paper is focussed on the optimisation process of an interplanetary mission of our solar system. An example of a Titan orbiter mission is given.

Introduction

As seen during the LOTUS 2 workshop (see *Ref [1]*) in June 2002, the emerging low thrust propulsion systems lead to new trajectories computation techniques. In other hands, as described in *Ref [2]*, the propulsion system is a key element for an interplanetary mission, as it will have, for a given mission, a direct impact on the vehicle mass (which has a strong impact on the mission cost) and mission duration (which has an impact on the costs but also on the scientific goal of the mission and the ground team). Except for very favourable planets conjunction configurations, the more on board propulsive ΔV (velcocity increment) the vehicle will have, the more complex and fruitful, the mission could be. Therefore it is attractive to make studies and comparison on mission using the performance capability offered by the promising electrical propulsion technology.

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The present situation for interplanetary vehicle

The following table shows the different class of interplanetary mission in our solar system, from the easier one up to the most complex one

Legend : R : realised EC : current mission (dev / flight) P : project * : on the relevant planet satellite	**SUN**	**MERCURY**	**VENUS**	**MOON**	**MARS**	**ASTEROIDS**	**COMETS**	**JUPITER**	**SATURNE**	**URANUS**	**NEPTUNE**	**PLUTO**
FLY BY	R	R	R	R	R	R	R	R	R	R	R	P
REENTRY / LANDING		P	R	R	R	R	EC	R	EC*			
ORBITER	R	P	R	R	R	R	EC	R	EC			
SAMPLE RETURN				R	P	EC						
CREWED MISSION				R								

Possible use of solar cells

Fig 1 Status of the interplanetary mission

For the fly by missions, favorable cunjunction of planets of our solar system, combined with clever gravity assist manoeuvers, could allow a very few propellant on board. This was the case of Voyager, currently tripping out of our solar system. The re-entry / landing missions and orbiter missions have been performed or are currently on the way up to Jupiter with the Cassini mission and up to Mercury with the Bepi columbo mission. The relevant interplanetary vehicle is generally a monostage vehicle, with a quite high propellant ratio (that means the ratio between the propellant mass and the vehicle mass on the launcher) for the orbiter, due to the necessary energy for orbit insertion on the target planet.
For sample return missions, multistage vehicles have to be considered, as the total ΔV of the mission will be quite high for a monostage vehicle.

The *fig 2* shows the global vehicle architecture ratio for ESA and NASA orbiter (Mars, Jupiter and Saturn). Venus orbiter is also possible with a chemical propulsion system, with a quite high propellant ratio (TBD). Mercury orbiter cannot be achieved without the help of the electrical propulsion due to a high ΔV requirement.

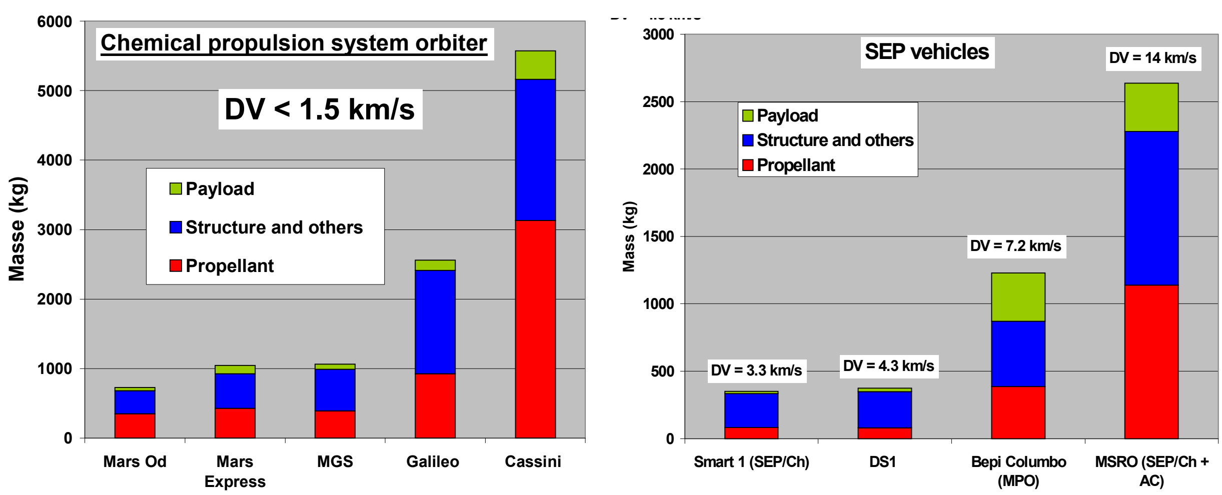

Fig 2 Vehicle architecture, chemical and solar electric propulsion

The following figure shows the energy needed with respect to the planetary position in our solar system.
As it will be discussed further, this energy could be shared between the launch vehicle and the interplanetary vehicule propulsion system.

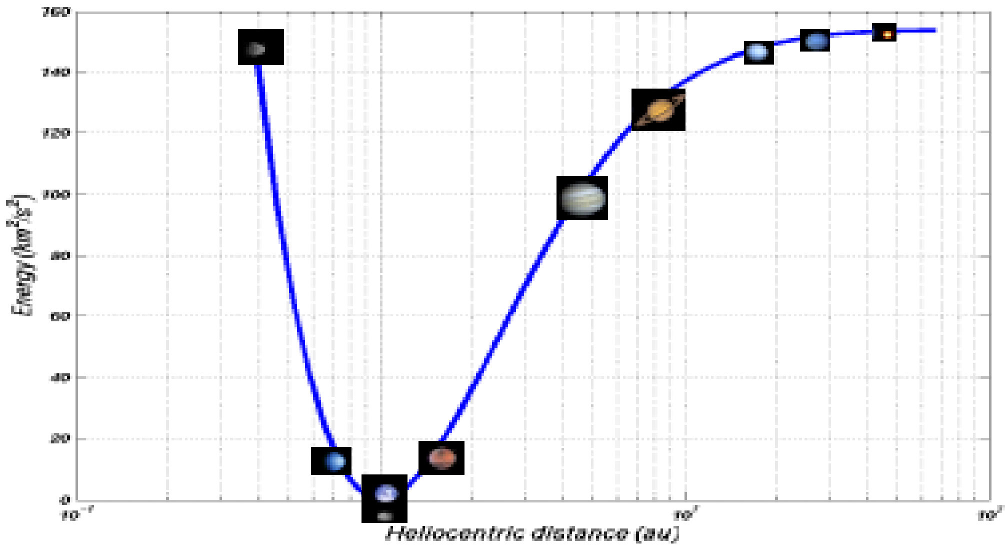

Fig 3 Necessary energy to reach the planets heliocentric orbit

For a monostage vehicle, the propellant mass fraction is directly linked to the ΔV requirement and the specific impulse of the propulsion system as shown on the *fig 4*. This clearly shows that electric propulsion technology gives the access to ΔV up to 40 km/s, with a still reasonable propellant fraction.

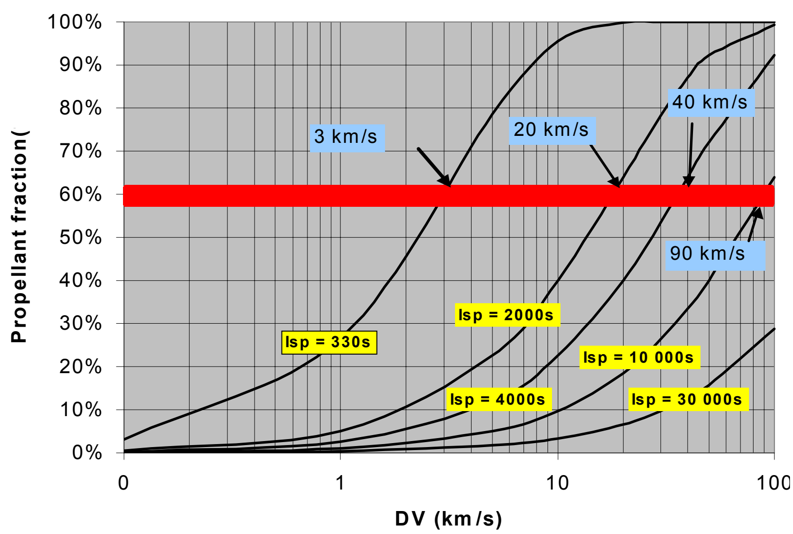

Fig 4 Propellant fraction as a function of the ΔV and Isp

<u>Mission analysis</u>

Two different kinds of interplanetary trajectories may be defined according to the propulsion system used:

- Impulsional trajectories,
- Low-thrust trajectories.

Impulsional trajectories

In general, the probe is directly put by the launcher on an Earth escape hyperbola. Moreover, this last one gives a sufficient outgoing relative velocity in order to allow the probe to reach the target planet. Nevertheless, this depends on the initial mass and on the initial ΔV required. Then, considering an orbiter mission, the interplanetary vehicle has to provide the energy to do the propulsive insertion around the target planet. However, this strategy involves a sufficient launcher performance, this could be some times impossible for technical or costs considerations. In this case, the ΔV requirement could be drastically reduced by using:

- Gravity assist manoeuvres of intermediate planets during the heliocentric phase,
- Gravity assist manoeuvres of satellites for the escape or insertion phases,
- Aerobreaking / aerocapture techniques (when it is possible) for the insertion phase.

The design of optimal impulsional trajectories is quite easy, because a finite number of manoeuvres and gravity assists manoeuvres has to be computed. Furthermore, the associated mission duration is generally short (10 months for a Mars mission) compared to the duration of low-thrust missions. But, the outer planets can not be reached by using classical chemical engines, without the development of complex (many gravity assist manoeuvres) scenarios.

Low-thrust trajectories

These trajectories come directly from the use of electric propulsion systems. Their characteristics are:

- A high specific impulse which decreases the propellant consumption,
- A low thrust magnitude, typically less than 1 Newton, which increases the mission duration,
- Complex sequences of coast and thrusting periods.

Using low-thrust engines, the transfer law employed may be defined as follows:

- **The escape phase:** this strategy may be similar to the impulsional case. On the other hand, the high Isp allows to put the probe on an Earth orbit, then the probe reaches the escape hyperbola thanks to a spiral trajectory (as described in *fig 5*): this Low Earth escape strategy allows to use smaller launch vehicles for a given probe mass, or to increase the initial launch mass of the interplanetary probe.

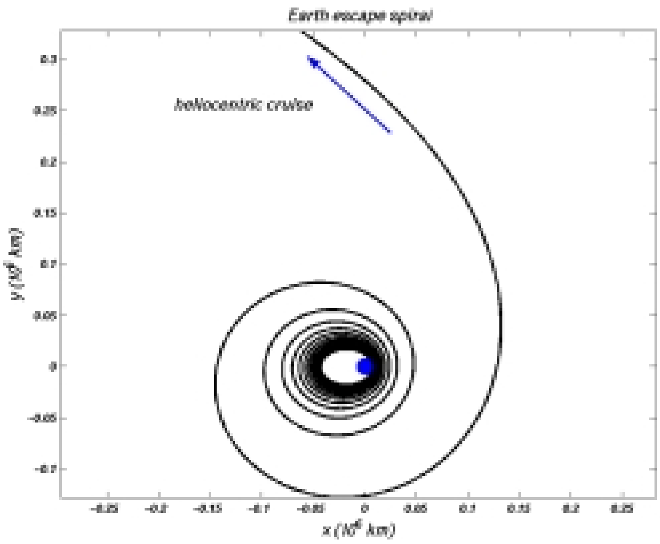

Fig 5 Example of an Earth escape spiral.

- **The heliocentric phase** defines a transfer between two planets (or bodies). Gravity assist manoeuvres of intermediate bodies may be introduced in order to decrease the mission ΔV.

- **The insertion phase** may be obtained by a spiral trajectory or by using aerobreaking / aerocapture techniques.

- **The performance index** which has to be minimised during the various phases may be the transfer duration or the propellant's consumption. If the mission duration is considered, then the engine has to be always switched on. On the other hand, for the consumption criterion, sequences of coast and thrusting periods have to be correctly located on the trajectory.

These optimal trajectories are extremely difficult to design (complex methods, numerical sensitivity…), see Ref [3] and Ref [4], and they require the implementation of complex guidance laws. Nevertheless, the low-thrust trajectories ensure generally more flexibility of the launch window and allows to reach the outer planets.

Mission optimisation with low thrust strategy

As each mission has its specificity's, it will not be reasonable to give absolute rules for interplanetary mission optimisation. In addition, there is a mission launch window dependency such as the energetic need of the mission will vary accordingly to the launch window. This is particularly true for chemical propulsion vehicle and we can expect that SEP (Solar Electric Propulsion) or NEP (Nuclear Electric Propulsion) vehicle will allow relaxing the launch window constraint, which could be a serious advantage.

Ideally, the comparison for a given mission will be based on :

- The payload delivered $M_{p/l}$, or the payload ratio $M_{p/l}$ / M_0
- The mission duration
- The initial M_0 mass

SEP vehicle architecture

In order to realise comparative studies in the frame of a mission analysis, it is necessary to have a mass model of the vehicle, typically,

$$\mathbf{M_0 = M_p + M_{str} + M_{sep} + M_{p/l}}$$

- $\mathbf{M_0}$ is the launch mass of the vehicle
- $\mathbf{M_p}$ is the propellant mass,
- $\mathbf{M_{str}}$ is the mass of the structure and other non explicitly mentioned subsystem (avionics, thermal control, telecommunication etc …) of the vehicle
- $\mathbf{M_{sep}}$ is the mass of the electric propulsion system, and the solar array, including batteries.

- M_{ep} is the mass of the electric thrusters, the Power-Processing Unit (PPU) and the Thrust Orientation Mechanism (TOM). As presented on the *fig 8*, the specific mass of the thruster is decreasing with the power. In a first approximation and for system studies, we can suppose that we have:

where Pe is the availble eletric power on board.

- $\mathbf{M_{sa}}$ is the mass of the solar array, including batteries. This mass is obviously depending of the power level required for electric propulsion at the maximum distance from the sun. A typical specific mass for earth solar array is β_e = 10 kg / kWe, and the mass penalty to have the same kWe at n A.U (sun - earth distance) could at first be assesed as $_2$, depending however of the used technology.

NEP vehicle architecture

If we consider nuclear power generation, the mass model of the vehicle will be changed according the following description:

$$\mathbf{M_0 = M_p + M_{str} + M_{nep} + M_{p/l}}$$

- M_{nep} is the mass of the propulsion system including nuclear reactor, conversion cycle, radiators and shield. Mnep = Mep + αPe, where α is the specific mass of the nuclear system, as described in the *fig 7.*

In a propulsion point of view, as the vehicle is not clearly defined, good comparison criteria could be, for a given mission, the propellant and propulsion system mass, compared to the initial vehicle mass.

For low thrust / high Isp vehicle, the mission analysis will optimise the following:

- Launch orbit: a lot of possibilities from LEO (taken into account the Van Allen radiation belts) up to direct injection to the planet. Escape time from earth against performance gain is the main trade off. This supposes a close loop between launcher performances and vehicle performances analysis.
- Heliocentric phase: ballistic or thrust phase, depending on mission time constraints, vehicle velocity at the arrival to the planet
- Insertion or re-entry phase.

Domain of use of solar electric propulsion: it can be used between the sun and Mars (see R[5]), may be Jupiter with appropriate concentrators, if they do not induce too much mass penalty and complexity.

The following schematics show the necessary mission time with respect to propulsive ΔV, for Earth and Mars solar environment, with reasonable electrical power fraction, that means 2 kWe per metric ton of vehicle on earth solar flux, and 0.86 kWe per metric ton of vehicle, at Mars. For example, we see that the escape time from Earth GTO to liberation (ΔV is approximately 3.6 km/s, in order to have Vinf = 0 km/s at Earth) is around 1 year for a 2000 s Isp propulsion system and 1.5 year for a 3000 s Isp propulsion system.

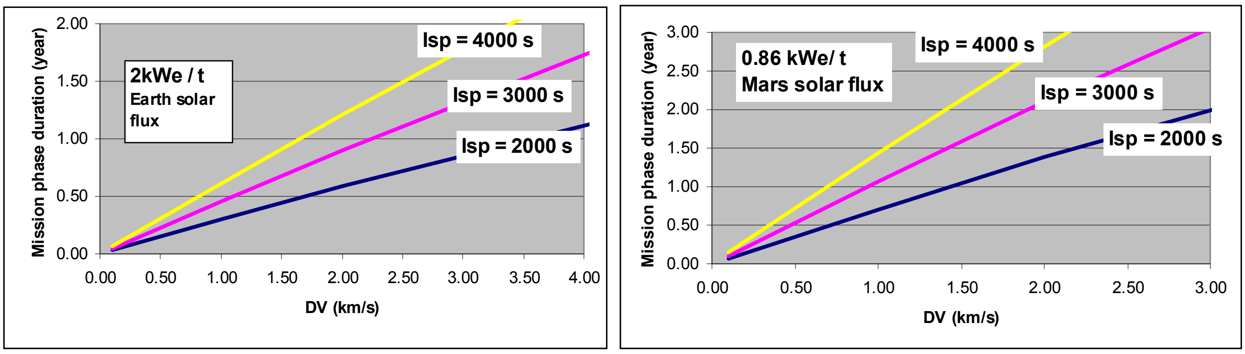

Fig 6 Mission time versus DV, for earth and Mars vicinity

Between the 2 planets, the variation of the available electric power will induce variation of thrust and thruster performances.

Example of an ambitious mission, the Titan orbiter mission

It is interesting to make comparison on a dedicated mission. The following assumption have been taken into account:

Launch strategy	A5-ESC-B direct injection. C3 = 15.74 km2/s2 T0 = 20/10/97	A5-ESC-B LEO injection
Vehicle	Chemical propulsion system M0= 7t Isp = 320 s	NEP vehicle, 150 kWe M0 = 19.6 t α = 23 kg / kWe Isp = 5000 s
Trajectory	Cassini type VVEJGA, Titan insertion on a 1000 X 1000 km orbit Total DV = 3959 m/s	*Earth escape, spiral out* *Earth - Saturn cruise* $V_{\infty,Terre}$ = 0 km/s $\rightarrow$ $V_{\infty,Saturne}$ = 0 km/s *Saturn insertion spirals down:* $r_{0,\ Saturn}$ = 47500000 km, e = 0 $\rightarrow$ $r_{1,\ Saturn}$ = 1221830 km, e = 0 - *Titan orbit insertion* $r_{0,\ Titan}$ = 37711.34 km $\rightarrow$ $r_{1,\ Titan}$ = 1000 km

Results

	Chemical propulsion system M0= 7t	Nuclear Electric Propulsion System M0= 19.6 t
Mission duration	6 years	7 years
Propellant mass ratio	71%	45%
Propulsion system mass, (including nuclear system)	600 kg	4t
Propulsion system mass ratio (%)	80%	65%

For the chemical option, the propellant ratio shows that the mission is probaby not possible with a chemical vehicle. The compararison with a Nuclear Electric Propulsion vehicle, with a quite high level of power on board, shows that the misson could be achievable with a Low Earth Orbit escape strategy (which allows a higher probe mass) without too much mission time penalties.

Conclusion

The use of electric propulsion could be very attractive for interplanetary mission, due to the high ΔV capability with reasonable mass vehicles. This supposes adapted strategy, with complex optimisation law and more generally optimisation work between launch vehicle, on board propulsion of the interplanetary vehicle and mission designer. The use of an appropriate vehicule architecture model, highly depending of the relevant propulsion (and others) technology will allow to make sensfull trade off in the future interplanetary missions.

Rererences

Ref [1] 2nd International Symposium on LOw ThrUSt trajectories LOTUS 2. Toulouse, juin 2002, Proceedings.

Ref [2] Muriel Nocca, "*Advanced Electric propulsion for outer planet exploration*". AAAF Versailles, May 2002.

Ref [3] R. Bertrand, '*Optimisation de trajectoires interplanétaires sous hypothèses de faible poussée*', Ph.D. thesis, Université Paul Sabatier, Toulouse, 2001.

Ref [4] R. Bertrand, S. Geffroy, J. Bernussou and R. Epenoy, '*Electric Transfer Optimization for Mars Sample Return Mission*', Acta Astronautica, Vol. 48, No. 5-12, pp. 651-660, 2001.

Ref [5] Sophie Geffroy, Rodolphe Cledassou, Nicolas Pillet, Jean-Renaud Meyer*
"Optimal use of electric propulsion for a phobos sample return mission"
AAS/ AIAA Space Flight Mechanic's meeting. Santa Berbara, February 2001.

Annexes

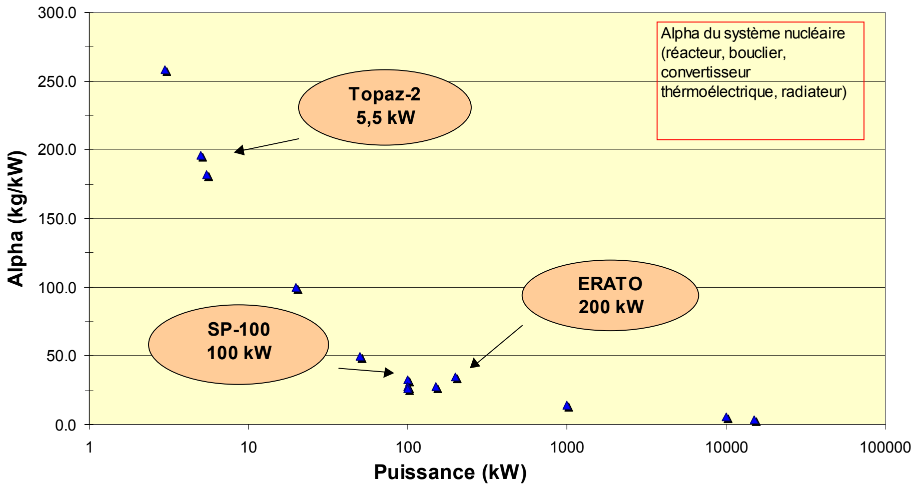

Fig 7 Specifc mass of nuclear system

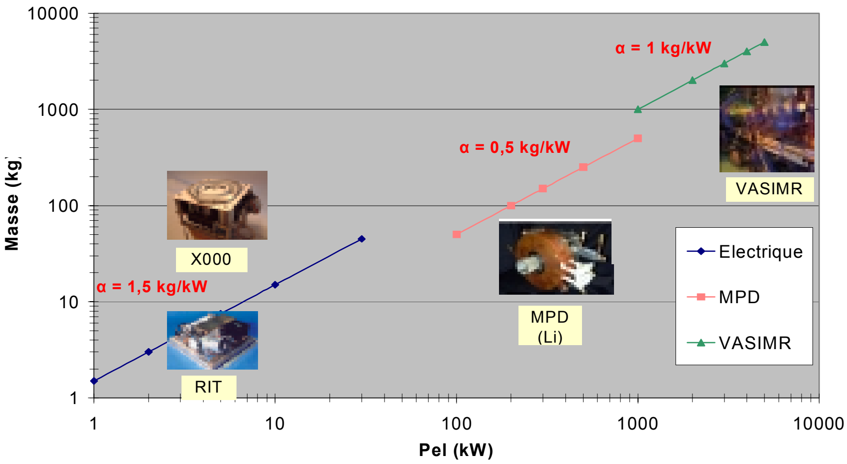

Fig 8 Specifc mass of electric thrusters

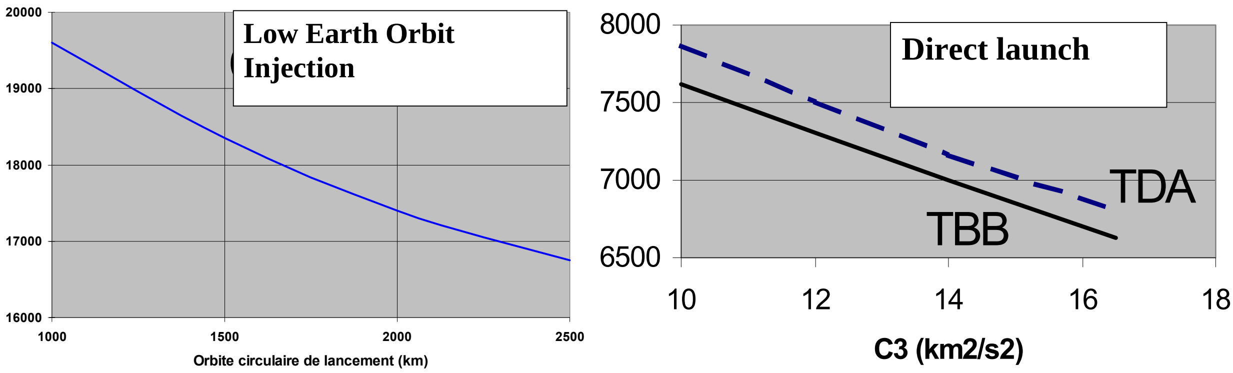

Fig 9 Ariane 5 - ESC B performances

Titan Orbiter Mission