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SOLAR THERMAL PROPULSION

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ABSTRACT

The desire to reduce launch and operational costs in future space transport systems has increased interest in new propulsion concepts for upper stages. Solar Thermal Propulsion (STP) in one of these concepts which, by reducing the propellant mass fraction during orbit transfer can offer more payload than present chemical stages.
The results presented here are derived from a study sponsored by ESTEC and dedicated to Solar Thermal Upper Stage Technologies and Propulsion Orbit Transfer Systems.
A technological analysis of the main propulsion sub-systems is presented first and on this basis are predicted the performance improvement for some typical categories of launchers when upper stage is replaced by a Solar Thermal Orbits Transfer System and with a 30 days transfer duration assumption.

1. INTRODUCTION

1.1 GENERAL APPROACH

Spacecraft orbital transfers are today achieved with system based on chemical propulsion. Typical satellite transfer, as LEO/GEO transfer mission, currently requires the participation of two propulsion systems, the launcher upper stage and the satellite apogee engine. Both systems are characterised by high thrust and low transfer trip time but require high propellant mass.
New propulsion systems are investigated in order to reduce the propellant mass fraction and increase the payload mass, or use smaller launch vehicles for a given payload mass.
Electrical propulsion which can provide very high specific impulse does not permit a Satellite transfer's duration lower than several months, then does not seem commercially very attractive for many customers. Alternatives based on intermediate performances (trade-off between propulsive performance, transfer trip time & mass) could be interesting compromises between chemical and electric systems, from launcher as well as from payload point of view. Solar thermal propulsion is one of these alternatives.
The results presented here are derived from an ESTEC study performed by a team composed of EADS-LV (prime), ASTRIUM, CONTRAVES SPACE, AIR LIQUIDE.
The study is directed at defining how a solar propulsion system could lead to reduce launch & operational costs. The objective is to replace the upper stage of the launcher and provide direct transfer to the final orbit. The target for transfer duration is one month. On-orbit propulsion will not be discussed. Propellant is hydrogen, since maximum performance for this kind of vehicle concept is in connection with maximum achievable specific impulse.
First phase was dedicated to technological analysis of the main propulsion sub-systems and to a prediction of STOTS performance when applied to some typical categories of launchers. In order to be able to define a "best concept", the problem was first to answer some preliminary questions:

- What is possible today with present state of art? Tomorrow with reasonable development effort? Later perhaps with research effort?
- What are the more promising technologies of concepts?
- What is the best strategy for satellite transfer from injection to final orbit?

Second phase was dedicated to an application of these results on an ARIANE 5 configuration.

ABBREVIATIONS

$C_r = \left(\frac{R}{r}\right)^2$	concentration ratio
D	diameter of collector
F	focal length
R	collector radius receiver aperture radius

1.2 STOTS DESCRIPTION

The solar thermal engine serves as a high-temperature heat exchanger, collecting concentrated solar radiation and transferring this energy to a propellant causing a significant specific impulse. There are **two basic types of solar thermal propulsion.** **The first approach** simultaneously collects energy, transfers it to a propellant gas and produces thrust. This "continuous flow" system must perform the transfer as a spiral orbit-raising manoeuvre (refer to figure 1), with consequential high "gravity losses".

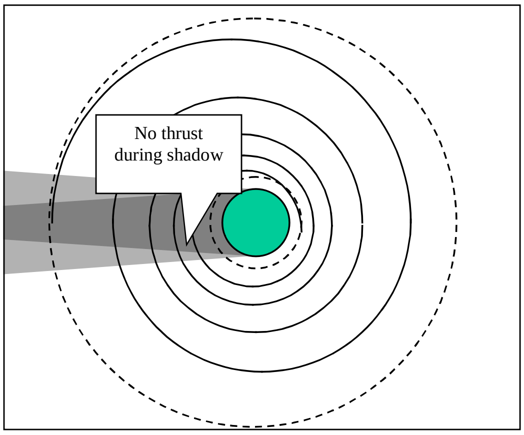

FIGURE 1: "CONTINUOUS THRUST"

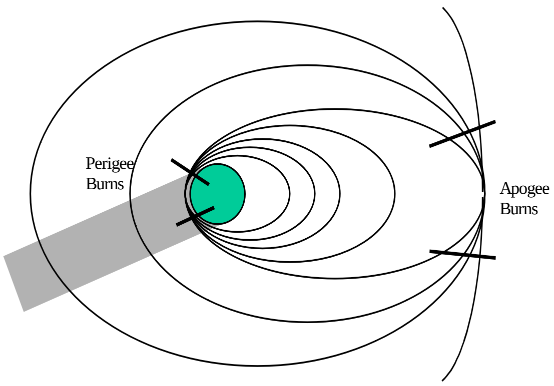

FIGURE 2: "INTERMITTENT THRUST" ORBIT

The second approach sequentially collects energy, stores it in a Thermal Energy Storage (TES) system and then transmits this energy to the propellant during short thrust periods. In this case of "intermittent flow" the STOTS operates at higher thrusts, allowing the vehicle to perform orbital transfer manoeuvres as a series of perigee and apogee boosts with low "gravity losses" (refer to figure 2). The "intermittent flow" system, however, must provide additional mass in the form of thermal capacity to store the collected energy before it is used for propulsion.
The solar thermal propulsion system consists of three interrelated sub-systems.

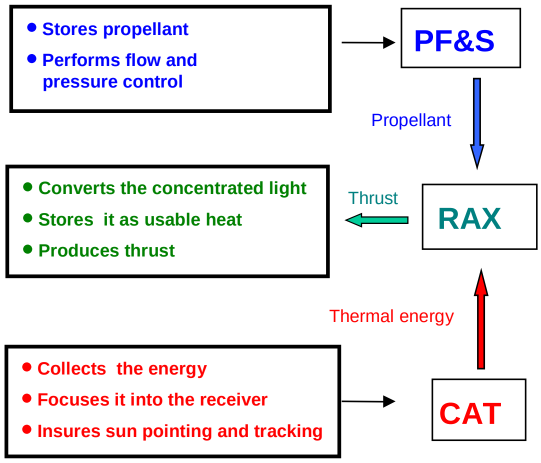

FIGURE 3: STOTS S/S and functions

- The Concentrator Array & Tracking Sub-system (**CATS**) in charge of collecting and focusing the sunlight for use by the system.
- The Receiver Absorber, or Receiver Accumulator eXchanger for intermittent flow **(RAX)** in charge of converting concentrated sunlight into usable heat to produce thrust.
- The Propellant Feed & Storage (**PF & S)** sub-system in charge of storing the propellant and feeding the engine.

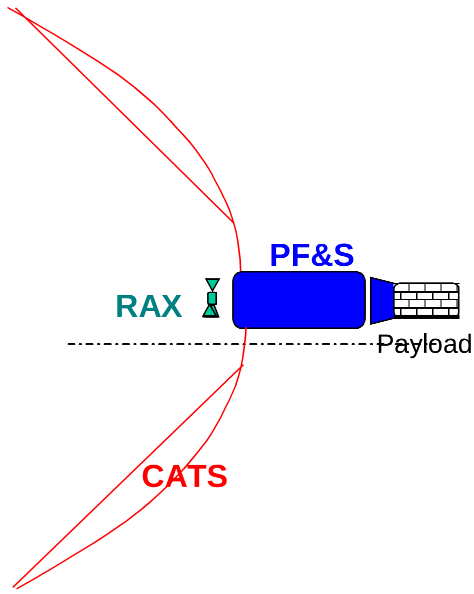

FIGURE 4: STOTS components

2. SUB-SYSTEM TECHNOLOGY

2.1 CATS

Worldwide, no technology has achieved flight status. If some of the concepts have it is only in the form of microwave antennae. In analysing seven generic classes of structural concepts, radial-rib reflector, tension-truss reflector, rim-frame reflector, composite reflector, petal-shell reflector, inflatable reflector, FRESNEL lens, area specific mass was found to range from 0.14 to 2 kg/m^2 and the packaging efficiency from 3 to 700 m^2/m^3.
As a synthesis of the analysis of performance (error budgets, collector efficiency, estimate of material-associated losses), figure 5, below, presents the levels of mass specific power for different collector configurations. It shows levels achievable with the different technologies compared with early space concentrator estimates.

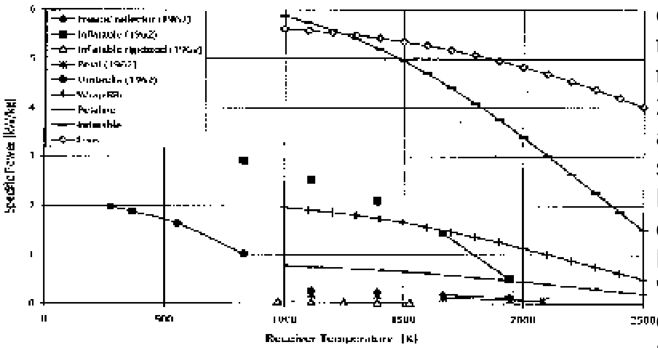

FIGURE 5: COLLECTOR SPECIFIC POWER COMPARED WITH EARLY SOURCE DEMONSTRATORS

Receiver temperature is directly linked to available specific impulse. For a receiver temperature Tr = 1 600 K (Isp ~ 650 s), the mass-specific power exhibits an excursion of one order of magnitude, if one compares objects with the same aperture. The inflatable technology offers the best reflectors with the flexible lens/space rigidified structures. The least efficient concept is the petaline one. The difference between inflatable and lens becomes significant beyond 2 000 K (Isp ~ 720 s).
If one compares the volume limited specific power, the range of values extends over two orders of magnitude, with the petal concept at the lower end and the inflatable reflectors as the most efficient ones.

Conclusion on technology availability

- For **near term** (SOA technologies), Sunflower-like petal reflector offers the major advantage of a low development risk. However, because of the petaline designs' aperture limitations and poor packaging efficiency, this type of items is limited to an experimental step.
- For **medium term** (asking for reasonable development effort), inflatable reflectors offer a good balance of properties, with a moderate development risk, and with room for further improvements. No size restrictions exist within the specified range for this class of objects. Although these elements still need to be fully qualified, past work has established the basic feasibility of their realisation, and also identified the range of achievable contour errors. With their unsurpassed packaging efficiency, inflatable seem well suited as "first generation" collectors.
- For **long term** (needing research effort) FRESNEL lens have the best potential for increasing the receiver temperature, but their feasibility has yet to be demonstrated. It uses only flat elements, and the lens manufacture may be amenable to industrialised replication, with associated cost advantages.

So, **for medium term project** the inflatable technology has been retained. The selected concept is composed of a two-collector array using offset primaries, with a double lens/torus configuration, each lens formed by two parabolic caps joined at their rim. The typical dimensions for a LEO/GEO transfer with a heavy launcher are an area of 2x200 m^2, a tripod height of 15 /18 m and a SMA around 20 m. The torus must be pressurised at a significantly higher level to stabilize the assembly. Rigidization is forgone because of the short lifetime of the mission.
For wall materials and fabrication process a "modest" and a "conservative" solution corresponding both to well-established products are possible. Materials' technology being sufficient it should be possible to concentrate the development effort on functional properties (accuracy, installation, lifetime).
A minimum mass solution has been searched for the system composed of the skins of the parabolic chamber, the stabilisation torus, three supporting beams and a multiple of the gas quantities for nominal filling of aforementioned chambers. The main design parameter is the parent paraboloïd's F/D ratio. The mass per unit area never exceeds 0.2 kg/m^2 and remains mostly below 0.15 kg/m^2. Packaging efficiency is orders of magnitude better than for other concepts.

Collector performance for a given concentration ratio depends on the receiver aperture's size which must be matched to the quality of the primary mirror. For each sun image size (σ_f) it exists an optimal receiver aperture r_r/R (i.e. an optimal concentration ratio) that maximises the collector efficiency (ratio of net rate of heat collection to the direct solar power incident upon the concentrator).

3. THE ENGINE

The engine design depends on the transfer strategy and on the mission.

3.1 CONCEPT DESCRIPTION

Design for continuous thrust

The receiver design could be either a "flat plate" heat exchanger illuminated from both sides or an insulated cavity (refer to figure 6). In either case the input area of the receiver defines a minimum re-radiation area, and as a consequence the first area of heat loss from the system. With the flat plate heat exchanger the re-radiation temperature varies from the propellant storage temperature at one end to the exhaust temperature at the other. Only the hottest parts of the heat exchanger will exhibit high re-radiation losses. A cavity, on the other hand, will have a high absorptivity, but behave as a high temperature black body for re-radiation. By increasing the concentration ratio the aperture area can be decreased without changing the heat exchange surface area. This allows re-radiation losses to be controlled.

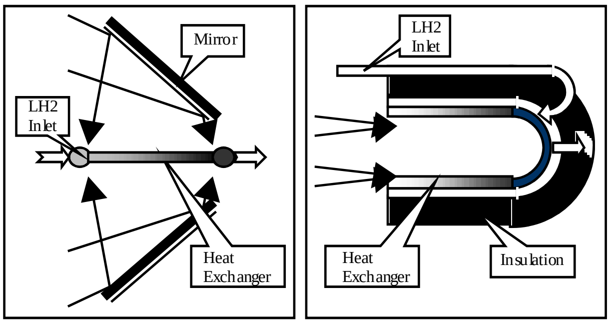

FIGURE 6: EXCHANGERS FLAT PLATE AND CAVITY HEAT

Design for "intermittent thrust"

For the intermittent thrust system, a major new items is the introduction of a thermal storage mass. The input solar power is used to heat up this mass, and later the propellant extracts this energy for the purposes of propulsion. The appropriate design for the Receiver/Absorber in this instance is a cavity surrounded by thermal storage material through which the propellant is passed during thrusting periods.

The available energy input time per orbit is the time that the vehicle is in sunlight. The length of the propulsion arc sets the available propulsion time. The minimum input time occurs on the first orbit. As the apogee height is raised, the vehicle spends longer and longer times in sunlight.

The performance of the "intermittent thrust" system will depend on selecting a material with a good thermal capacity at high temperatures. Materials with a melting point in the range of interest will store additional energy as latent heat of fusion. Selected candidate materials are: silicon, boron, graphite.

		MELTING POINT (K)	HEAT OF FUSION (kJ/kg)	SPECIFIC HEAT[1] (J/kg K)	THERMAL CAPCITY[2] (kJ/kg)	DENSITY (kg/m^3)
Boron	B	2 352	4 644	1 026	8 553	2 340
Graphite	C	3 700	N/A	710	4 021	7 850
Silicon	Si	1 683	1 800	710	3 647	2 330

TABLE 1: CANDIDATE THERMAL STORAGE MATERIALS

Graphite as a single-phase material has a comparable thermal capacity to two-phase materials, but only at high temperatures. The data assumes a graphite temperature of 2 500 K.

Materiel properties

At temperatures above 1 700 K the principal candidates for high temperature structures are rhenium (or more probably rhenium-molybdenum alloy) and Carbon-Silicon Carbide (C-SiC) composite. The absorptivity and emissivity of rhenium and rhenium-molybdenum are low in comparison to that of C-SiC, but Ultramet have developed Chemical Vapour Deposition (CVD) processes for generating "dendritic" surfaces for which the absorptivity and emissivity are both close to unity at 2 000 K. Rhenium is expected to be suitable for heat exchangers operating up to 2 800 K, and is readily formed into thin walled complex shapes by CVD. An experimental rhenium resistojet was built and tested in the UK in the 1970s using this technique with wall thickness' varying between 0.125 mm and 2 mm and tube diameters down to 2.1 mm.

C-SiC ceramics are being worked on within EUROPE both at SNECMA (SEP) and in GERMANY (Universtät Stuttgart). Silicon carbide is not affected by nitrogen, hydrogen or methane at least up to 1 900 K, although carbon can be attacked by hydrogen at very high temperatures. C-SiC may also be limited at high temperatures by the comparatively high vapour pressure (0.1 atm at 2 800°C).

As structural materials, the high temperature wall materials will be subject to both pressure and thermal stresses. The pressure stresses give a lower limit to the wall thickness. Thermal stresses are created by the temperature drop in the tube wall, set by the rate of heat transfer across the wall, and set an upper limit to the wall thickness.

The actual wall thickness will have to lie between these two limits.

3.2 S/S PERFORMANCE OPTIMISATION

A preliminary trade-off analysis was made at engine level and results are presented for continuous thrust mode.

Optimisation of some engine parameters

Increasing area nozzle ratio above about 125 induces very little change in P/L fraction. Number and diameter of tubes were calculated to minimize the gas side temperature differential taking account of the constraints induced by the general shape of the focal spot and by a maximum pressure drop within the heat exchanger.

Influence of concentration ratio

For a given concentration ratio there is a peak performance (payload fraction) at a certain temperature with the performance dropping rapidly once this temperature has been exceeded due to T^4 re-radiation losses. To achieve high temperature (and so high specific impulse and correlatively high P/L fraction) the cavity must operate at high C_r.

Continuous thrust - Comparison of plate and cavity concepts

A cavity receiver will have a higher effective absorptivity than a flat plate receiver, but also a higher emissivity, and will tend to re-radiate as a uniform black body at the upper temperature. At the same concentration ratio, the cavity receiver reaches its peak performance at a lower temperature.
As the payload fraction increases in line with the gas temperature, for any given concentration ratio the flat plate receiver will have a better performance than the cavity receiver.
At the same time, to achieve high temperatures, higher concentration ratios are required. And for the plate concept heat transfer problems become more difficult due to the heat-exchanger decreasing size as the gas temperature is increased. The cavity makes it possible to maintain the effective heat transfer area when concentration ratio is increased and hence avoid some of these heat transfer problems.
In conclusion to provide a high level of specific impulse the difficulty will be to have a sufficient concentration ratio in the case of the "cavity concept", and to have an efficient heat transfer for concentration ratio above 1 500 in the case of "plate" concept.

Intermittent thrust - Thermal capacity requirements

The thermal capacity installed will be a compromise between energy storage capacity and the capability of the vehicle to use that energy during "burns". Investigations showed that for many materials, except boron, the mass penalty associated with the thermal storage material meant that the maximum payload fraction was achieved with a thermal capacity which was even less than that required to absorb the available energy over the first orbit.
At the same operating temperature the two-phase systems give a better performance.

4. PROPELLANT & FEED SYSTEM

4.1 THERMAL AND FLUID ASPECT

During the mission the main issues for the P & FS are linked with propellant acquisition during low gravity phases and the management of pressure level in the tank during coast periods.
During low gravity flight the feed system uses a capillary Liquid Acquisition Device (LAD) which offers the advantage of providing vapour free liquid without consuming extra propellant to impose a settling thrust.
Tank thermal protection prevents hydrogen boiling and avoid a rapid increase of internal pressure. The pressure control is insured by a Thermodynamic Vent System (TVS), utilised in active of passive mode, which provides also a continuous flow for the engine. TVS is composed of a Joule-Thompson (JT) device associated to an heat exchanger. The propellant leaving the tank is throttled to a lower pressure and temperature and then vaporised through an heat exchanger which sub-cools liquid around the LAD to prevent it from boiling and insure at the same time bulk propellant cooling.
Sizing Propellant Management Devices (PMD) to limit pressure excursions in the tank is particularly difficult for intermittent burning transfers as the heat loads entering or leaving the tank depend on the ratio of burn and coast duration on each orbit and are changing all along the mission. Insulation can then only be tailored for a mean ratio of $\Delta t_{burn}/\Delta t_{coast}$ and a cooling and an heating device are necessary to keep the pressure nearly constant. Electric heater provide energy on demand to limit pressure drop during thrusting phases; cooling power is adjusted to the need using:

- a passive TVS (PTVS) sized on the fist orbit provides the minimum amount of energy needed to sub-cool the LAD and to keep the pressure constant,
- an Active TVS (ATVS) sized on final orbit using a pump to increase the heat-exchange coefficient on command.

Flow-rate limitations

For intermittent burning mode the lower acceptable flow-rate is fixed by the performance of insulation. It can be determined from the following criterion:

$$\frac{M_{burn} \times \left(t_{burn}/t_{orbit}\right) \times L}{Tank\,.\,area} \geq \Phi_{min}$$

With:

Φ_{min} = 0.3 W/m^2 for FMLI with approximately 100 layers of MLI, at 2.5 kg/m^2,

Φ_{min} = 1.4 W/m^2 for FMLI with approximately 30 layers of MLI, at 1.2 kg/m^2.
Maximum possible level results of sloshing amplification at engine start and cut-off, pressure excursions and propellant management devices (TVS and LAD) capability.
No LAD nor TVS have ever been experimented for LH2 flow-rate larger than 5 g/s.

Thermal insulation
After scanning of foams, MLI, FMLI categories of insulation, FMLI has been chosen as the more adapted insulation system for the STOTS.
FMLI constituted by 10 mm H920A foams 30 layers to 100 layers of MLI allow to reach the thermal performance required in most of the case. Increasing the thickness of the foam to 20 mm (thickness used on ARIANE 4 third stage) could suppress the need of purge gas on ground and during atmospheric ascent of the launcher but it could increase the mass of insulation.
The low performance of FMLI on ground (typically 120 W/m^2) will be used to obtain enough heat loads to pressurise the tank from 1 bar to 3 bars on ground, few hours before the launch.

Technology availability
Liquid Acquisition Device (LAD) working with LH2 at flow-rate in the range of 1 g/s to 5 g/s is the most challenging equipment. The other critical equipment would be the TVS.
Using a continuous thrust at sufficiently large flow-rate could avoid using this equipment if it could be demonstrated that eclipse influence is negligible. In that case, all the necessary technologies required to build the PF & S sub-system could be considered as state of art.
In case of intermittent thrust a large technology effort shall have to be performed on the LAD and the coupled PTVS, including design studies, 1 g testing and low g testing.

4.2 MECHANICAL ASPECTS

The metallic tank is supported by struts because it is the only way to reach the objectives of heat flux for the whole tank. In selected configuration, the tank sustains the general loads and a jettisonable fairing insures thermal protection during the atmospheric ascent.
A mechanical sizing has been performed for a set of missions and for different configurations of tanks. Configurations studied have considered architectures based on a structural tank and an external jettisonable fairing. In **the** selected configuration, the tank sustains the general loads and a jettisonable fairing insures thermal protection during the atmospheric ascent.
In most of the cases an additional skirt is necessary in order to leave enough space under the lower bulk-head for the propulsion system. This together with the thermal protection fairing adds a considerable mass to the stage. Nevertheless these two massive items are abandoned very early in the mission.

4.3 MASS BUDGETS

At the beginning of the upper stage mission (after jettison of the fairing and the additional skirt), the ratio between the full mass of the PF & S sub-system and the mass of the LH2 stored is in the range of: 0.48 to 0.61 for a 8 m^3 tank; 0.28 to 0.37 for a 29 m^3 tank; 0.22 to 0.34 for a 80 m^3 tank.

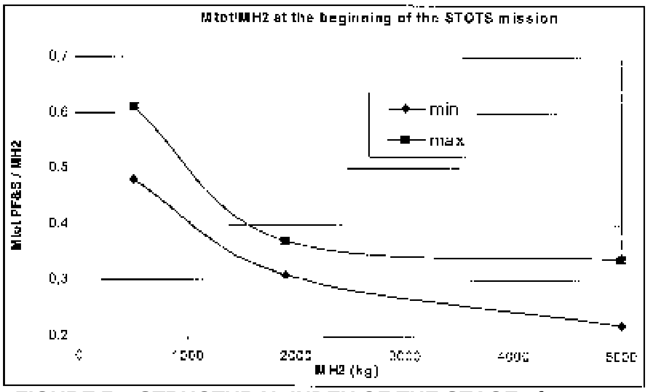

FIGURE 7: STRUCTURAL INDEX OF THE STAGE after the jettisoning of the fairing and of an additional skirt

The results of PMD sizing performed for different transfer strategies inducing different propellant mass flow-rates and results are shown on figure 8.

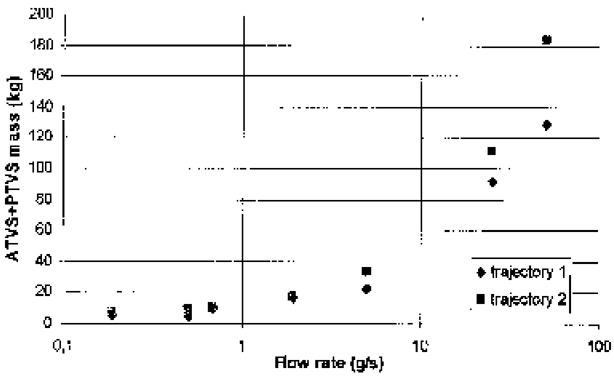

FIGURE 8: MASS OF THE PTVS AND ATVS AS A FUNCTION OF THE MASS ENGINE FLOW-RATE

5. PERFORMANCES

The study of STOTS interest for replacing current chemical transfer upper stage and payload apogee engine must be appreciated in function of propulsion system performances but also on constraints due to the necessary adaptation of the launcher to this new propulsion concept. Are particularly important the initial orbit stability requirement due to the low level of thrust provided by STP as well as the consequences of low density of hydrogen inducing large tank volume and sometimes adaptation problems for the vehicle or even for the launch pad.
Low thrust level requires STOTS stage injection on a stable orbit. Many solutions are possible to reach the required injection orbit, the use of solid propellants (if not limited by international agreements), thanks to their compactness and low cost seems an efficient way to provide the needed velocity increment. In the case of ARIANE 5 required propellant mass ranges from 500 kg to 1 500 kg. The problem is in fact different for each launcher. If for ATLAS V and DELTA 4 additional means are also necessary, in the case of SOYUZ launcher the first three stages can inject the solar stage directly on a stable orbit.

MODELLING
Two levels of modelling have been successively implemented.
Early simulations use **simplified modelling** to analyse the sensitivity of results to main parameter variations and to have first comparative results for the two possible transfer strategies or the two possible receiver concepts (plate or cavity).
For continuous burning mode transfer, the EDELBAUM approximation has been improved in function of in-house available data. For intermittent mode transfer boosts applied at perigee or apogee are assumed quasi-impulsional and no corrections are considered.

Optimisation tools were then used to refine the performances of STOTS.

- **In the case of continuous burning** mode optimal thrust orientation is defined all along the fixed duration transfer.

- **For the intermittent thrust mode**, with a fixed transfer duration, the maximal performance is reached by the simultaneous optimization of the mass of the Thermal Energy Storage (TES), the surface of the concentrators, the propellant flow-rate, the thrust duration located at the apogee or at the perigee. The optimal solutions are situated in a domain of exchange between the average Isp, the mass of the TES and the surface of concentrators. For the same payload, the design parameters are enclosed between a high solution with a maximal mass of the TES (this one penalises the payload but allows to reach a higher Isp and a lower ergol mass), and a low solution (minimal mass for the TES with a lower Isp level and a more moderate surface of concentrators). The choice between both solutions has to be balanced between an easier accommodation (with the benefit of a lower ergol mass for example) and the technological difficulties (larger surface for the concentrators).

For mission analysis with optimization tools, the considered technologies correspond to a technical level under development: inflatable concentrators, graphite as material for the receiver. Two types of receiver (cavity and plate) have been systematically studied. Numerical results presented below in the text correspond to "cavity" which is supposed to adapt more easily to high temperature receiver.

6. MAIN RESULTS AND CONCLUSION

The results of technology analysis and performance modelling lead to following answers to the questions of § 1.1.

Sub-systems key parameters and critical issues

- **CATS**

Large surface collectors (several hundredths of m^2) are required if preferred missions are LEO to GEO transfers involving heavy launchers (for telecom market for instance). This is possible with inflatable solutions due to their packaging efficiency, and very low design mass. Required pointing accuracy will induce development program.

- **RAX**

Mission analysis confirms that, for direct delivery in GEO a high specific impulse is necessary to get a significantly better P/L than with a cryogenic stage. Table below resuming the performances of ARIANE 5 on a LEO/GEO transfer (assuming continuous thrust mode) shows that it looks as if an initial charge has to be paid to take into account the replacement of the natural upper stage.

Isv	700	750	**800**	850	**(ARIANE5 ESC-B)**
P/L	5900	6600	**7300**	7900	5400
$(P/L)/(P/L)_{ref}$ -1	9%	22%	**35%**	46%	0%

So, key parameters and critical issues for receiver/absorber/exchanger are related to the search for a high level of specific impulse and there is room for further improvement with regard to our conservative assumptions (Is around 700 s/750 s). In intermittent mode, materials for TES are more adapted to a sixty days trip time. A research phase to confirm boron applicability or to find new materials with high thermal capacity is necessary.

- **PF & S**

Cryogenic LAD and TVS have never been flown in low gravity. Studied equipment has been limited to low propellant mass flow-rates (< 5 g/s). Improvement of H2 tank structural index (composite structure) would induce a gain in P/L around 500 kg for an application to ARIANE 5 LEO/GEO transfer.

Transfer best strategy

For MEO or GEO missions, and for a thirty day transfer, **continuous thrust mode appears as a simpler and more efficient solution than intermittent mode**. It provides higher P/L mass for thirty days transfers. The **intermittent transfer** mode has to match the heat capacity of the accumulator with the changing orbital conditions. In consequence the results are very sensitive to thermal energy storage properties and to the specified transfer time. **Intermittent thrust transfer also induces much higher requirements on main propulsion system parameters**: CATS area, hydrogen mass flow-rate, addition of Thermal Energy Storage system.

Preferred missions and launchers

General trends show that STOTS will be the more so interesting as the initial stage mass and the velocity increment attached to the mission are important. Small ΔV are not favourable to high specific impulse propulsion systems. Small launcher stages are particularly disadvantaged by the high structural index due to hydrogen low density.
Some performances in GEO for a 30 days continuous mode transfer are shown on table below assuming, a C_r of 4 000 and a specific impulse of 750 s (payload in kg).

LAUNCHER TYPE	VERY HEAVY	HEAVY INTERMEDIATE	MEDIUM
Example	ARIANE 5	Delta 4 M	SOYUZ 2(*)
Payload	6 600	2 650	2 080

(*) From Kourou.

- Medium launchers which are able to deliver upper stage directly on a stable low earth orbit without the help of an additional stage and which perform orbit transfer in a month with moderate requirements for propulsion system are interesting for single launch.

- Heavy and very heavy launchers need and additional propulsion system to reach a stable orbit and this is one of the reasons for the "initial charge" to be paid for STOTS installation on the launcher. Orbit transfer calculations also show that better results are obtained with launchers capable of high mass in low orbit. In other words launchers which rely more on upper stage to build their performance are handicapped. This is illustrated by ARIANE, which provides a good performance on low orbit, and would be interesting for dual launch of future heavy satellites.

For an Earth-Mars transfer performed with ARIANE 5 in 2003 or 2005 a comparison STOTS/ESC-B gives (payload in kg):

DEPARTURE DATE	ESC-B	STOTS (60 DAYS)	STOTS (90 DAYS)
2003	5 978	7 134	7 420
2005	4 400	6 328	6 603

Study first phase conclusion

The examination of the different possible technologies for each sub-system has shown no stopping issue for the replacement of the launcher upper stage by a solar thermal propulsion system in a medium term perspective. A STOTS pre-project could be envisaged for all the classical missions on the basis of the results obtained during STOTS study.
For specific impulse exceeding 700 s, it would easily improve ARIANE 5 performance on a LEO/GEO transfer with double payload. Estimates performed with conservative assumptions lead to more than 20% increase on payload for a specific impulse of 750 s and a similar level of reduction on recurrent cost when compared to chemical propulsion.
An effort to solve the problems raised by high level of temperature or research level solutions (e.g. FRESNEL lens, boron thermal energy storage material, composite tank) could in the future still improve present previsions.

7. APPLICATIONS ON HEAVY LAUNCHERS

A specific application on LEO-GEO mission with Ariane 5 and Soyouz 2 using a complementary technological study and a new transfer strategy has been performed.
The final configuration is based on two inflatable parabolic reflectors with two secondary Compound Parabolic Collectors. Specific power was 3 kW/kg with a specific impulse increased to 800 s. Area ratio is around 4000.

RAX is a T-shaped cavity RAX (see figure 9) and material is C-SiC.

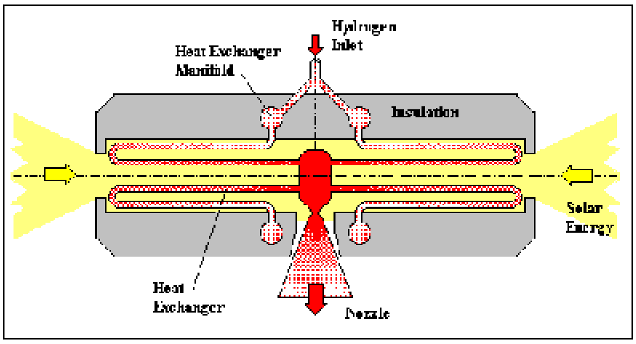

FIGURE 9: Concept for Dual Ended Cavity RAC

Thrust mode is quasi-continuous and thrust level is around 60-80 N. Induced transfer duration is one month.
Hydrogen is stored in a metallic tank and an active thermal system manages the pressure evolution. Capillary system insures a liquid feeding of the thruster.

Payload is increased and predicted value is around 7.25 tons. (Gain of 35% over Ariane 5 ESCB on GEO)

8. CONCLUSION

STOTS concept, associated to a 30 days transfer duration, offers an interesting potential in payload mass point of view. Retained concept is based on an inflatable collector, a cavity heat exchanger and a metallic tank with liquid and pressure control systems. Reasonable studies and development are necessary to confirm technological feasibility and performances. Technological key points are given below:

- Tracking methodology for a large inflatable mirror has to be developed,
- C-SiC behaviour from RAC has to be tested in representative conditions,
- Liquid acquisition device capability has to be demonstrated for the calculated flow rate value of hydrogen.

Medium term for technological aspect was a main driver for the present study but evolution of technology will increase performance of STOTS concept especially with the use of Fresnel lens.

Two others ways are possible to improve performance but induce constrains for customer. One is to increase transfer duration. The second is the use of electrical energy of satellite: a mixing of thermal and electrical solar energy allows stage to realise required Delta-V. Anyway customer is always right!