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Référence O.T. :

Édition : **Révision :**
N° interne : **FS0202794A**

Contrat :
Lot de travaux :
N° d'activité :
Classe :
Catégorie :

P R O J E T O U P R O G R A M M E

STOTS
ESTEC Contract N° 14023/99/NL/PA
EXECUTIVE SUMMARY

Diffusion Interne			
Page de garde	**Nb**	Document Complet	**Nb**
		FLTX	1
		FST	1
		FSTF A Melchior	
		FSTF C. Koppel	
		FSTC	
		FSTP	
		FLTMCDOC	1

Diffusion Externe		
Destinataires	Nb	Noms

Edition		Ed.1 Rev.1	Ed.2 Rev.1	Ed.3 Rev.1	Ed.4 Rev.1	Ed.5 Rev.1	Ed.6 Rev.1
Visa/date		16/09/2002					
Rédacteur	Nom	D.Valentian					
	Visa						
Rédacteur	Nom						
	Visa						
Vérifié par	Nom						
	Visa						
Resp. hiérarchique	Nom						
	Visa						
Qualité	Nom						
	Visa						
Programme	Nom	A. Melchior					
	Visa						
	Nom						
	Visa						

ENREGISTREMENT DE MODIFICATIONS				
N° ÉDITION	N° MODIFICATION	DATE	PAGES MODIFIÉES	RÉFÉRENCES ET RÉSUMÉ DES MODIFICATIONS
1		16/09/2002		Édition originale

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FICHE DOCUMENTAIRE

CONTRIBUTION A LA MEMOIRE D'ENTREPRISE

Référence PROJET :

DESCRIPTIF DU DOCUMENT

TITRE : Solar Thermal upper stages technologies (STOTS)

Executive summary. ESTEC Contract N0 14023/99/NL/PA

Confidentialité :

Programme :

Références : **N° interne : FS0202794A** **N° Édition : 1** **N° O.T. :**

Auteur(s): **D. Valentian**

Date : 16/09/2002

TYPE DE DOCUMENT (selon MPA02-09) **: S**

- **N** = normatif, guide, standards, savoir-faire ☐
- **S** = synthèse, état de l'art ☒
- **D** = définition, plan, prévision ☐
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RESUME (principaux résultats et non résumé de l'introduction) **:**

Solar Thermal Propulsion application to commercial orbit transfer (LEO - GEO, sub. GTO - GEO) and to planetary missions has been analysed.
The STOTS study led to the selection of the following technical solutions .
Rigid mirrors made of several panels, accumulation with a reduced accumulation time, the main advantage being to eliminate stringent pointing requirements during firings. Constant coast time firing strategy. Metallic LH2 tank with hybrid thermal insulation (foam and superinsulation).
The performance improvement bought by STOTS is nearly 1 ton in GEO for a 3300 kg GEO reference mass. The most promising application seems interplanetary missions.

MOTS CLES DESCRIPTEURS :

Mots clés libres System study, solar thermal propulsion, orbit transfer, trajectory optimisation, rigid mirror, fine pointing, liquid hydrogen, thermal accumulator

(*) si applicable

Emission de la fiche

Sigle :FLTX	**Nom :Valentian**	**Visa :**	**Date :16/09/2002**

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1. INTRODUCTION

Solar thermal propulsion is a fairly old concept first proposed by Ericke in the 60's.
The US made a large development effort in this field since 1985 with the objective of nearly doubling the payload in GEO for a given launcher.

Since this technique could give a competitive edge to US industry in the field of commercial GEO spacecraft, ESA / ESTEC decided to start two parallel contracts on solar thermal propulsion.

In this respect, it was necessary to analyse :

- The credibility of US claimed performances.
- The feasibility of Solar thermal propulsion in Europe.
- The critical points of STP (mirrors, receiver, LH2 storage tank, pointing requirements).
- The expected performance gains.
- The developments steps to reach qualification.

From initial hypotheses, the project focused on minimum risk technologies yet yielding very high performance gains.
Several transfer strategies were analysed in order to reduce transfer time while maintaining a significant payload gain.
The application of STOTS to planetary missions led to very promising results.
The feasibility of critical technologies was assessed and a development plan proposed including a flight experiment.

2. HISTORY

- Most studies and developments on solar thermal propulsion were made in the US involving mainly ROCKETDYNE and BOEING for overall design as several small companies for the critical technologies (among them, ULTRAMET for the heat exchanger and the thermal storage superinsulation). The "customers" were NASA and USAF.
- In Europe, the work on the subject was performed by DLR while SNECMA performed in-house studies to check the potential interest of US work.

Summary of US works : ISUS, STUS

The presentation of US works on the subject is a good introduction on solar thermal propulsion since many years of system studies and laboratory developments helped to select a general architecture and technical solutions forming the backbone of any new study.

ISUS means Integrated solar thermal upper stage. It is not a stage but a part of the host spacecraft.
In the ISUS concept, the solar thermal accumulator is used not only for the propulsion but also for electrical energy supply during the whole mission, thus replacing the electrical accumulators during the eclipses.
The energy supply was supposed to be provided by thermoionic diodes (figure 2.1).

The figure 2.2 shows a design case of ISUS for DELTA.
The spacecraft is shown in orbital configuration : vertical axis is North - South. The mirrors are sun pointing : therefore a first DOF is required to allow the mirrors to follow the sun in elevation.
A second DOF between tank and spacecraft is required to allow earth pointing.

It can be shown that the spacecraft undergoes very profound modifications : deletion of one radiator, deletion of batteries and solar panels. The limited electrical power level is too small for a telecommunication spacecraft.

Therefore ISUS is not applicable to a commercial spacecraft.

In addition, recent tests in USA showed that it was not possible to extract the required power from the diodes (too low "hot" diode temperature, itself due to excessive radiative losses).

Figure 2.2 is also representative of a classical solution in solar thermal propulsion.

- The solar flux is concentrated by a primary mirror (concentration ratio (4000 or more)

and a small secondary mirror (concentration ratio 4)
The chapter 6 will show that this layout is the result of an optical optimisation. However, the primary mirror concentration ratio induce a very severe sun pointing requirement. This is one of the main difficulties of STP.

- Hydrogen is heated in a receiver / accumulator (RAC), allowing to store thermal energy and fire at perigee (or apogee) thus allowing to reduce Delta V requirement close to Hohmann optimum.

In addition, accumulation offer the possibility to relax the pointing requirement during thrusting.

- Hydrogen is stored in a large tank with superinsulation and low thermal conductivity links to minimise heat inputs. Ideally heat input will provide hydrogen evaporation to supply the RAC. Since operation takes place in microgravity, phases must be separated. To this end several systems have been proposed :

- Passive thermodynamic vent LH2 is evaporated through porous vaporisers and cold vapours are used to cool the tank walls LH2 is supplied to vaporisers by vanes like on surface tension tanks. The device is efficient but heavy.
- Active thermodynamic vent ;: the principle is the same but hydrogen is circulated by a pump through an heat exchanger.
- Spray bar : this last entry can be coupled or not to active thermodynamic vent. The spray bar is used to re circulate cold liquid in the vapours ullage thus limiting temperature gradients and pressure excursion in the tank.

STUS : Solar Thermal Upper stage could be jettisoned after use as well as **SOTV** (Solar Orbital transfer Vehicle (Boeing Study).

STUS receiver has been tested in NASA Glenn tank 6 together with a multi faceted rigid mirror.

In parallel, USAF and NASA studied inflatable mirrors aimed at weight and - more important - volume gain. However the development of inflatable mirrors is very difficult an lags behind other techniques (see chapter 6 : mirrors).

SOTV is an upper stage located between the launcher and the spacecraft; this induces the necessity to use the LH2 tank as a load bearing structure and this induces a weight penalty and more conductive thermal losses. Since SOTV is mainly design to perform a flight experiment these drawbacks have a limited impact.

Pointing : the fine pointing is a difficult requirement in any case. In SOTV fine pointing and refocusing are insured by an hexapod (6 DOF) allowing to finely focus the primary mirror input onto the secondary one.

Continuous firing : this will require sun pointing while firing : this is a major issue and feasibility has to be demonstrated. US did not choose yet between accumulation and continuous firing.

Form the performance point of view continuous firing requires higher Delta V but accumulator mass is essentially removed. Orbited masses (if pointing is feasible !) are quite close.

Figure 2.1 ISUS Receiver with thermoionic diodes

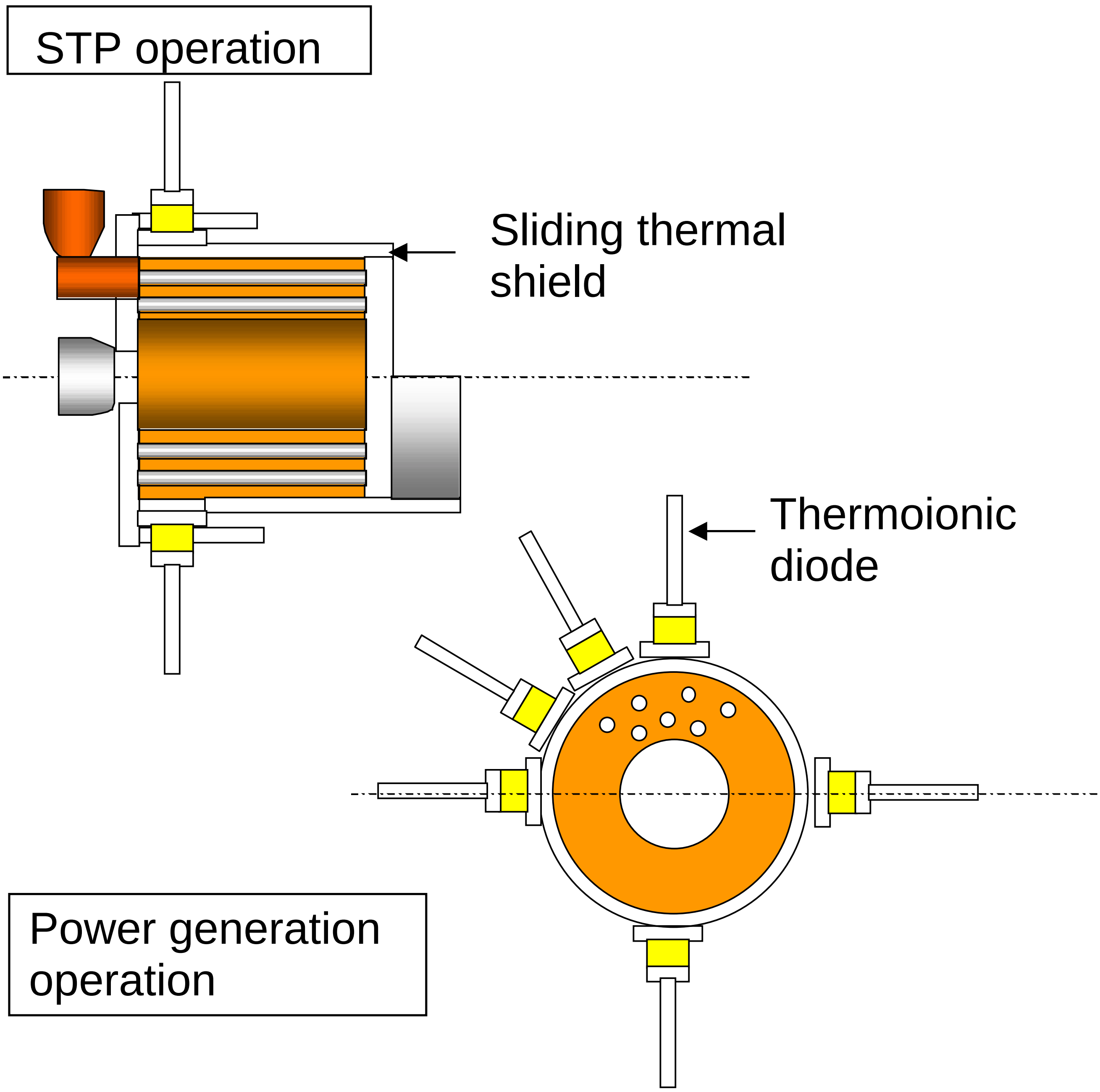
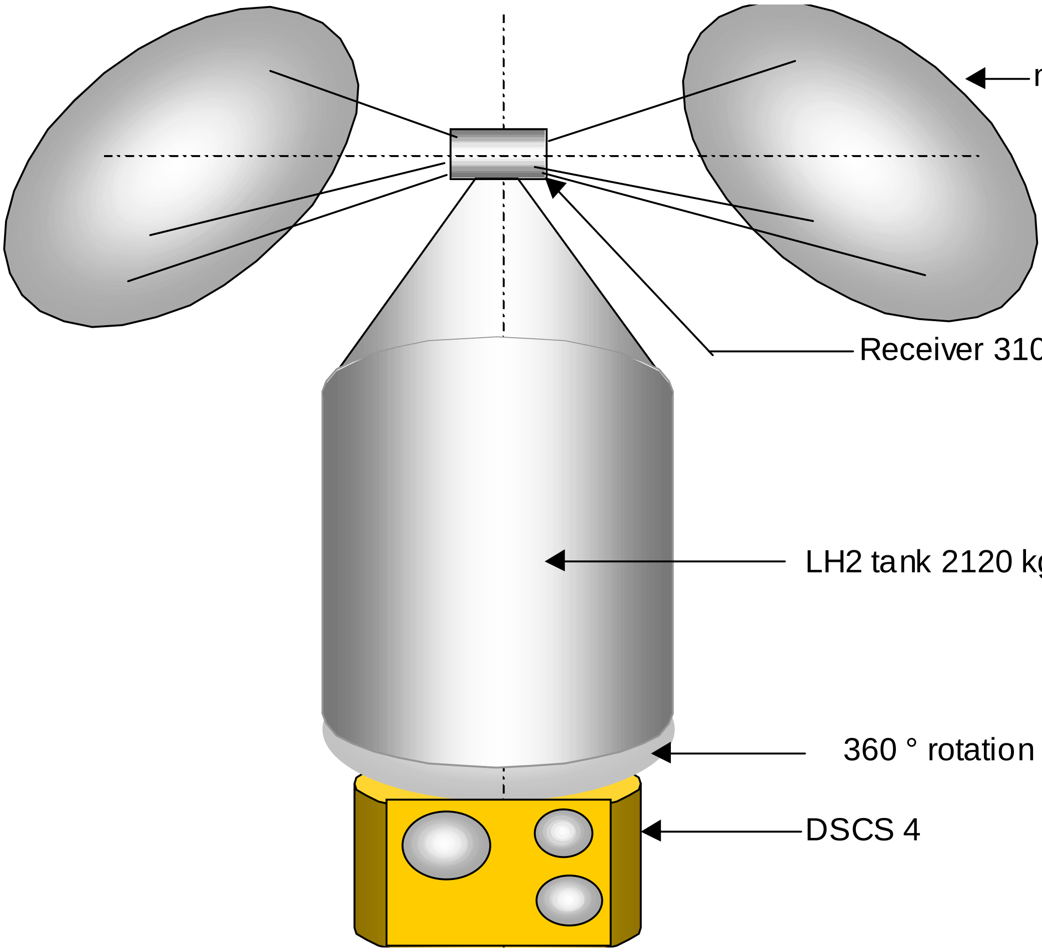


Figure 2.2 ISUS for DELTA



3. SYSTEM ASPECTS – SUBSYSTEMS -

Starting hypotheses

3.1. Global Geometry of primary mirror and launch configuration constraints

3.1.1. Rigid segments mirrors:

The space remaining between fairing and LH2 tank can be used in two manners:

- The mirrors are folded between the cylindrical space between tank and dynamic envelope.
- The mirrors are folded in the ogival shaped space above tank.

The cylindrical - elliptical tank is assimilated to a flat ends cylinder, the cylinder extremities being used to provide attachment points for the folded mirrors and their deployment mechanisms.
Practically, the attachment points will be part of a CFRP structure attached to the elliptical ends in a reduced number of points to minimise heat losses.
Figure 3.1.1.a shows a three-segment cylindrical mirrors assembly stowed under fairing. Six segments will be probably necessary for a flight configuration.

The petals mirror solution is very attractive from the optics point of view but is very detrimental from the trajectory point of view: the thrust vector must be close to sun direction (except if accumulation is used).

A third solution combining the advantages of the previous ones is shown on figure 3.1.2: the mirror is split in three segments, according to the ESA patent on large antennas, thus allowing to store the folded mirror (or two mirrors) under fairing. This will allow to use 6 m dia. Mirrors.

This proved to be the best solution.

3.1.2. Inflatable mirrors:

Inflatable mirrors can use rigid masts like rigid mirrors, and they can be wrapped around the LH2 tank. However, this may lead to water and or ice condensation on mirror. The other solution is to use a storage canister close to the receiver and to provide folded rigid arms or inflatable arms.

Inflatable mirrors offer the advantage of a lower mass

The main problems with inflatable are the lack of precision of the mirror surface (typical collection efficiency ~50 %) and the difficulty of each step of the manufacturing process.

For example the castable polyimide film process requires a single block mirror polished master. Today it is not possible to exceed 6 m in largest dimension. Inflatable mirrors are interesting only for much larger sizes.

Figure 3.1.1. : Cylindrical mirrors holders : three panels layout : stowed configuration.

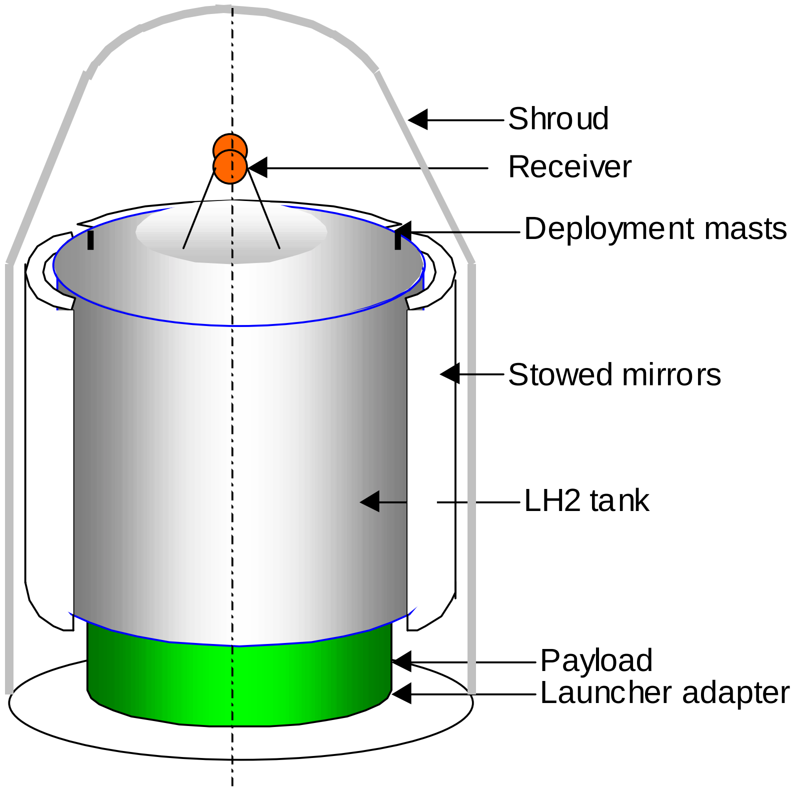
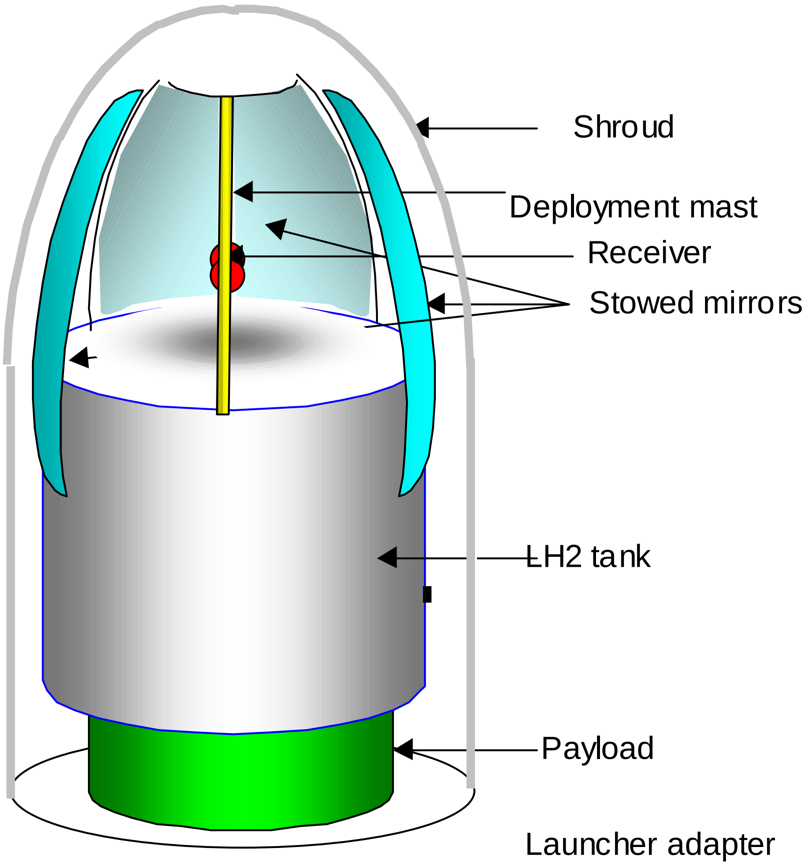


Figure 3.1.2. : Segmented mirror stowed inside fairing



4. MISSIONS : DEFINITION

Subsystems considerations

Receiver operating modes :

Equal emphasis was initially made on heat storage operation (US concept) and on permanent low thrust mode (expecting that Delta V penalty will be partly compensated by the dry mass reduction due to heat accumulation elimination).

Solar pressure induced torque

With an integrated payload between the receiver and the LH2 tank, the solar pressure induced torque is assumed to be small. In other cases, the proposed concept is to use a solar sail atop a collapsible mast in order to balance the solar radiation pressure on primary mirrors

Missions :

It consists in transferring to GEO from elliptical orbits with apogee of 8000 km (20000 and 36000 km). The elliptical orbit is provided by ARIANE 5 or by a chemical stage (RLV case). The interest of this starting elliptical orbit is to be safe from the atmospheric drag point of view and to mitigate the effects of irradiation in (protons) Van Allen belt. In addition, the eclipses penalty is less severe than with LEO initial orbit.
When a separate stage is used, the final orbit should be a "cemetery orbit" slightly above GEO DeltaV: 20 to 40 m/s provided by spacecraft).

The 100 000 km case was intended for planetary missions. In the case of half ARIANE V payloads, the GTO case was also considered (dual launch of a commercial spacecraft and of a planetary probe).

STOS sizing: Rigid mirror" initial study cases

The table 4.1.1 defines three sizing cases for STOTS with rigid mirrors, folded at launch around the cylindrical part of the tank.

The three cases correspond to LH2 masses of 1.3, 2.6, and 3.9 tons, leading respectively to theoretical total impulses of approximately 10, 20, and 30 MN.s.

The first line of table indicates the minimum shroud dynamic envelope diameter. The "large" case corresponds to ARIANE V.
Since the mirror support thickness is supposed to be 10 cm, the tank maximum outer diameter can be immediately deduced.
The provision for tank insulation and structure is 75 mm (85 mm for large tank). F(or an upper stage the thickness is 22 mm).

The tank shape is supposed to be cylindrical - elliptical, with 1:2 end ellipses.
The tank height includes the elliptical ends but not the insulation.
The tank structure is supposed to provide attachment points for the folded mirrors along two outer tank diameters separated by at least the tank height. This layout allows providing thermal stand-offs for mirror attachment points.

The tank volume is simply calculated form cylinder and ellipsoid volume formulas. The form factor helps to take into account possible variants (oval or conical profile instead of cylindrical).

Mirror height may include the payload height (small and large cases) in order to provide a larger mirror surface. The mirrors are supposed to cover 95 % of the tile surface. This is a little bit optimistic since external tiles edges will be probably not usable, leading to a too large numerical aperture. The correction factor is 0.95 x cosine(B/2). This helps to compute the effective collecting surface (net mirror surface). The intercepted power corresponds to 1400 W / m^2.
As in US publications, the primary and secondary mirrors transmission efficiency is held equal to 70 %.

The instantaneous power entering the receiver is deduced (net input power).

The tank filling-factor is equal to 95 %. The liquid hydrogen volume and weight are deduced.
With a specific impulse set at 7600 N.s / kg, the total impulse is computed.

From the net input power and the specific impulse the theoretical steady thrust can be computed, knowing the efficiency (60 %).
It should be noted that doesn't mean that the thrust is mandatory continuous. This is a method to compute the equivalent mission time.
It can be remarked that the power level and thrust levels have been adjusted to give missions times of approximately one month without eclipses of coasting periods.

Table 4.1.1 STOS SIZING: "Rigid mirror" case

ITEM	Unit	Small	Medium	Large
Minimum Fairing diameter	m	3,35	4,35	4,57
Tank outer diameter	m	3,15	4,15	4,37
Tank inner diameter	m	3,00	4,00	4,20
Tank height	m	3,25	3,85	5,00
Form factor	-	1,00	1,00	1,00
Cylindro-elliptical Tank volume	m^3	19,44	40,00	59,57
Mirror heigth	m	4,40	4,60	7,40
structure angle	deg	30°	30°	30°
Segments number	-	6	6	6
deployed mirror width	m	4,89	6,44	6,79
Mirror centerline - focus distance	**m**	**6,40**	**6,70**	**9,70**
"Vertical" aperture half angle	deg	20°	20°	22°
Numerical aperture	-	1,45	1,46	1,31
deployed mirror intercepted surface	m^2	21,19	29,19	49,26
Sun / secondary mirror axis angle (B)	deg	0,00	80,00	80,00
correction factor (k.cos B/2)	-	0,95	0,73	0,73
net mirror surface	m^2	20,13	21,24	35,85
intercepted power	W	28 187	29 741	50 188
Primary and secondary efficiency	-	0,70	0,70	0,70
Main Mirror number	-	**1**	**2**	**2**
Net input power	W	19 731	41 638	70 263
Tank filling factor		0,95	0,95	0,95
LH2 volume	m^3	18,47	38,00	56,60
LH2 mass	kg	1 293	2 660	3 962
Specific impulse	N.s/kg	7 600	7 600	7 600
Theoretical total impulse at 7600 m/s	MN.s	9,82	20,22	30,11
Thruster efficiency		0,60	0,60	0,60
Theoretical continuous thrust	N	3,12	6,57	11,09
Thrust duration	s	3 153 490	3 075 195	2 713 929
	day	36,50	35,59	31,41
Thrust integral	MN.s	9,82	20,22	30,11

5. TANK

<u>Acceptable heat input :</u>

The average evaporation rate of hydrogen shall be slightly lower than the required average flow. Taking into account a safety factor of 0.6, this defines a maximum acceptable heat flux in the tank.
For the three sizing cases, this gives the following values:

LH 2 mass (kg)	1293	2660	3962
Flowrate (30 days) (kg/s)	4,99E-04	1,03E-03	1,53E-03
Flowrate (45 days) (kg/s)	3,33E-04	6,84E-04	1,02E-03
Raw power (30 d) (W)	227	466	694
Raw power (45 d) (W)	151	311	463
Max. heat flux, 30 d (W)	136	280	417
Max. heat flux, 45 d (W)	91	186	278

The acceptable heat flux could be split half and half between conductive losses and radiative losses through superinsulation. In fact the total heat flux can be 5 times lower than these limits.

<u>Structure versus thermal aspects:</u>
In the baseline the tank is located on top of the spacecraft. Therefore the tank is holding only the receiver heat input from structure is on remote tank end. This simplifies the propellant management since the gas bubble is opposed to the H 2 acquisition system (see figure). The propellant remain stable under thrust (see figure 5.1.1).

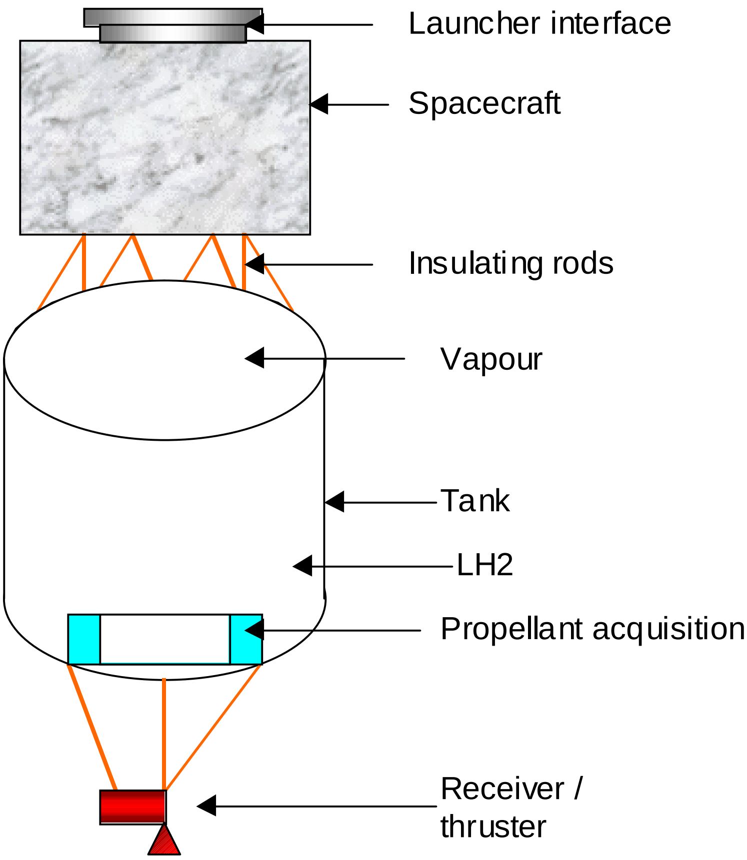


Figure 5.1.1 Baseline tank / spacecraft configuration (in space).

The previous table shows that the conductive losses are not a limiting factor for tank structural supporting rods as far as a low thermal conductivity material is used.

Tank Structure versus pressure :

To simplify the design of heat exchanger and flow control, it is better to fix a fairly high tank pressure (for instance 7 bars). The flow can be simply controlled by a variable conductance valve.
A fairly high pressure simplifies also the structural design: the tank is stiffer and its resonant frequencies higher.
On the other hand, the tank mass is roughly proportional to pressure. For aluminium cryogenic tanks, service pressure corresponds to the technologically feasible minimum thickness and lead to a pressure of 0.2 MPa. Lower pressure provides also a higher latent heat of evaporation. With aluminium tank a smaller buffer tank enables to feed the receiver at higher pressure.

Propellant acquisition:

The propellant acquisition and phase separation is performed by cold wall effect. This concept is called "thermodynamic vent" on US designs.
A simpler candidate is the spray bar (active temperature homogenisation of tank walls and ullage).

6. MIRROR

Mirror rotation

Supposing that thrust is within orbit plane, the mirrors shall be rotated around an axis perpendicular to orbit plane. The solution proposed in some US studies, i. e. to mount the rotation mechanism around the receiver, does not seem adequate:

- The mechanism is potentially exposed to high thermal fluxes,
- The masts are constrained to a fixation on the receiver.
- The mechanism shall include either a very large roller bearing or a ring held by small wheels of limited accuracy.

It is proposed instead to mount the rotation mechanism at mast extremity (see figure 6.1.1).

- This allows recovering most of existing (solar array drive) mechanisms.
- The mirror can be rotated near its centre of mass.
- When stowed, the mechanism has a limited protrusion above the mirror envelope.

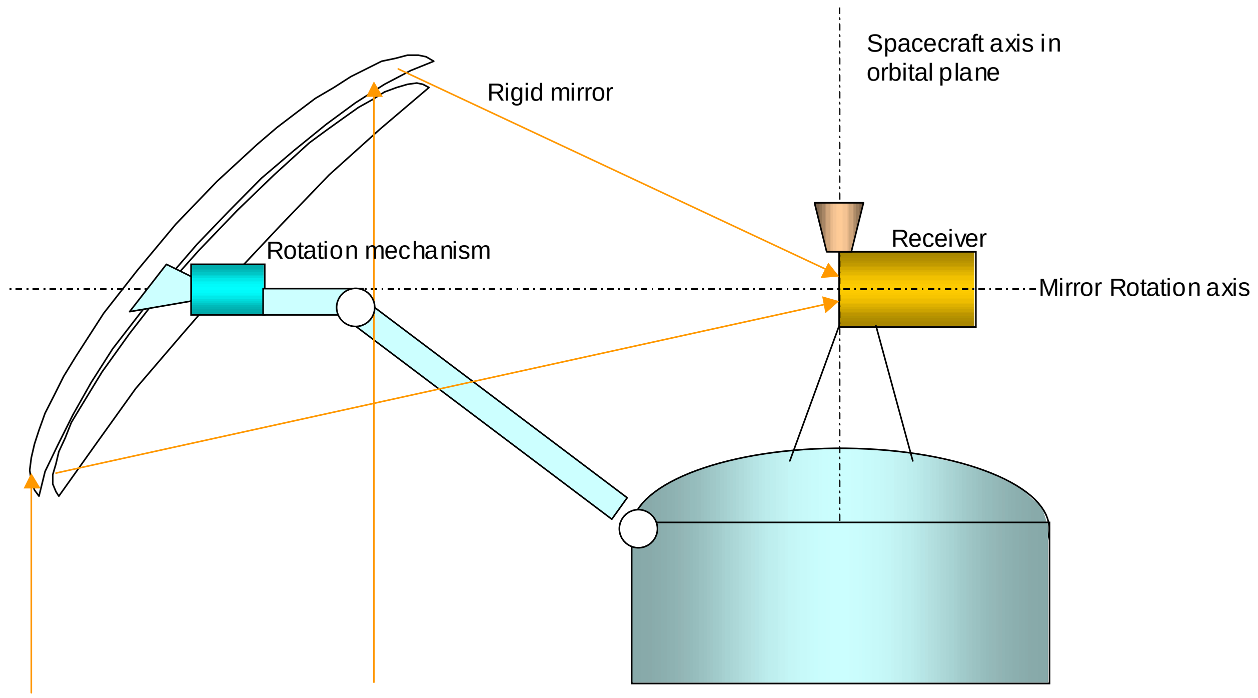


Figure 6.1.1 Mirror pointing and focusing on receiver

7. RECEIVER

Receiver assembly :

The nozzle is located on a flat face of receiver. This is due to a fundamental high temperature insulation rule: the radial superinsulation - made of coiled foils - shall not be drilled.
All openings (tube to nozzle, hydrogen inlet line, mechanical holder of carbon block) shall be located on flat faces. The axial insulation is hybrid: i. e. uses a mix of superinsulation foils and fibrous insulation. It is linked to the radial superinsulation in a stair case fashion on order to avoid radiative losses.
Preferred materials are:

- Accumulator: graphite block or SEPCARB ® assembly (carbon - carbon composite).
- Coating against hydrogen reaction: rhenium thin CVD LAYER.
- Manifold: built in the carbon block.
- Figure 7.1.1 shows a detail of the accumulator and the nozzle layout in the case of a 480 kW instantaneous power case. Figure 7.1.2 shows the front view of a two-cavity receiver (one hour storage time), and figure 7.1.3 shows the side view.

The following table gives the variation of Isp versus temperature:

Ref. temp.	Ref, power	H2 / total flow	Isp, e=250
K	kW	g / s	s
1700	443	17,4	715,4
1800	455	16,9	736,2
1900	468	16,5	756,4
2000	480	16,1	776,0
2100	492	15,7	795,2
2200	503	15,3	813,9

Figure 7.1.1 Accumulation thruster nozzle drawing

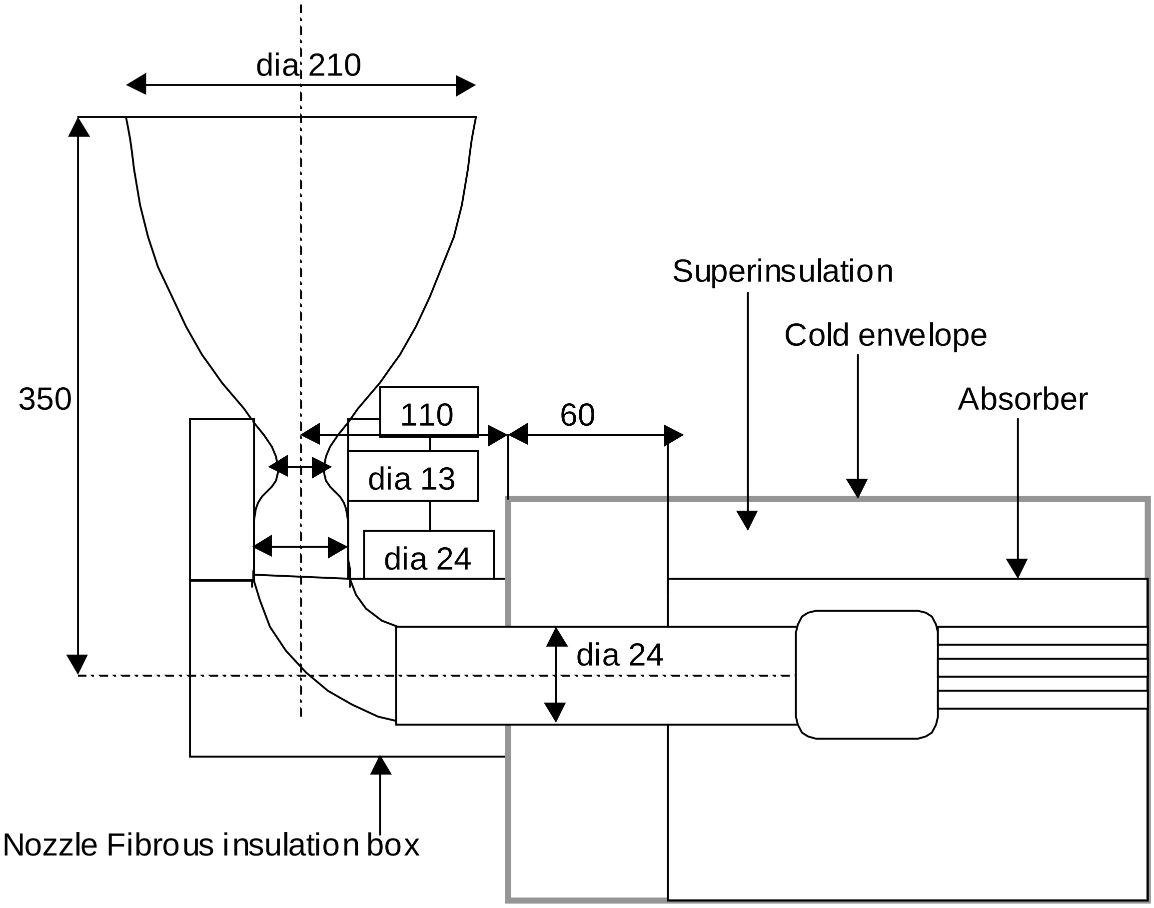


Figure 7.1.2 Front view of dual cavity receiver, 1 hour storage

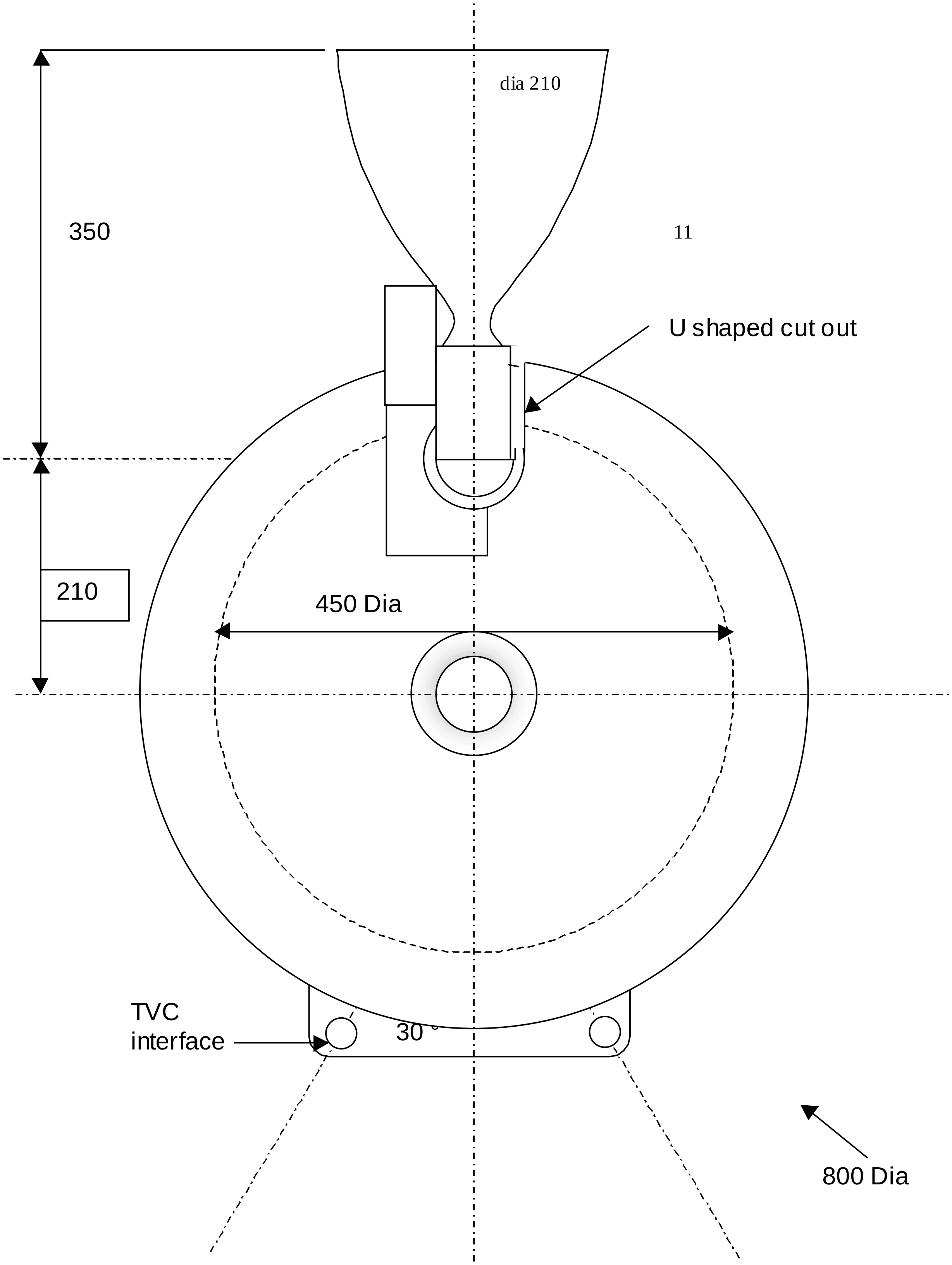
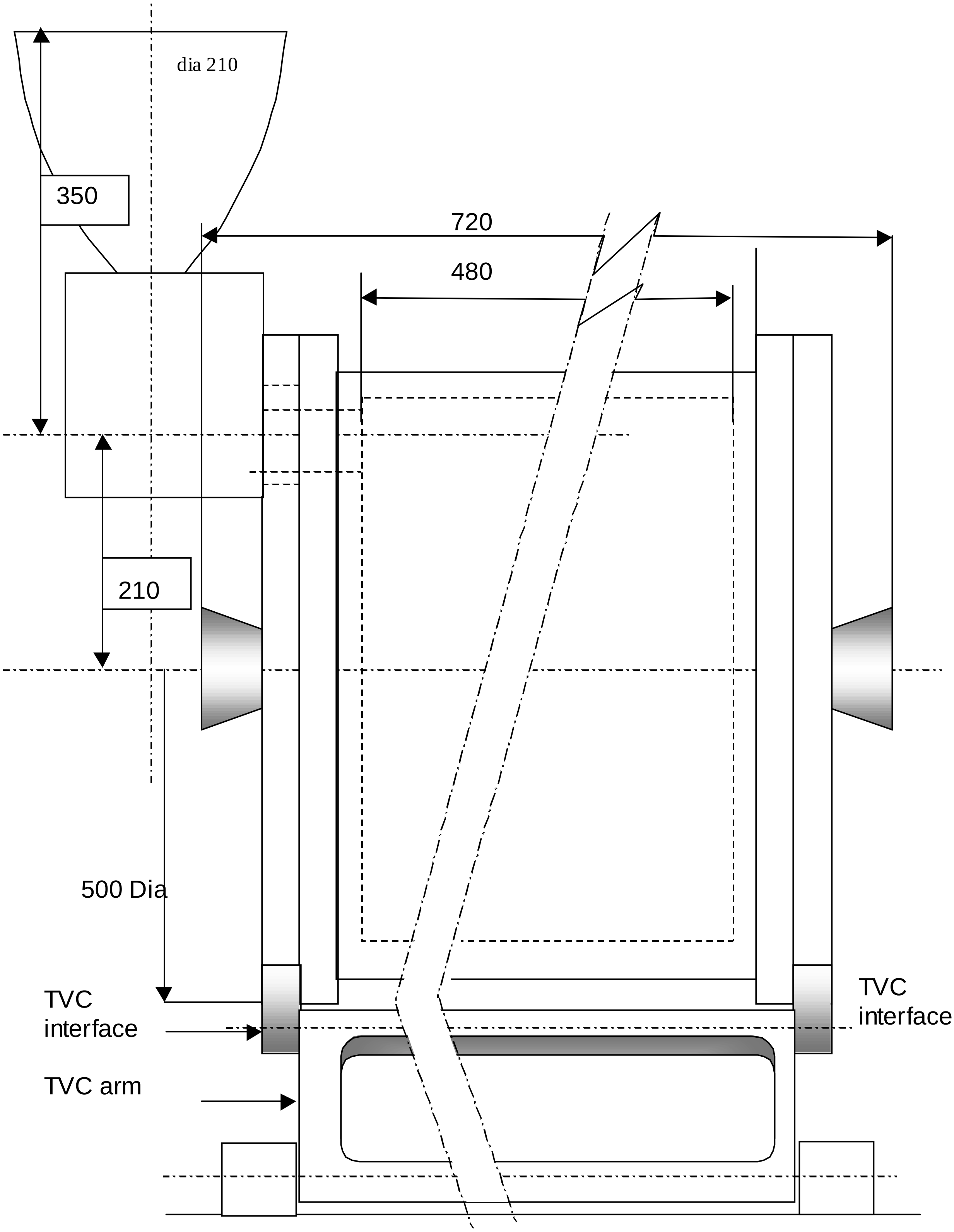


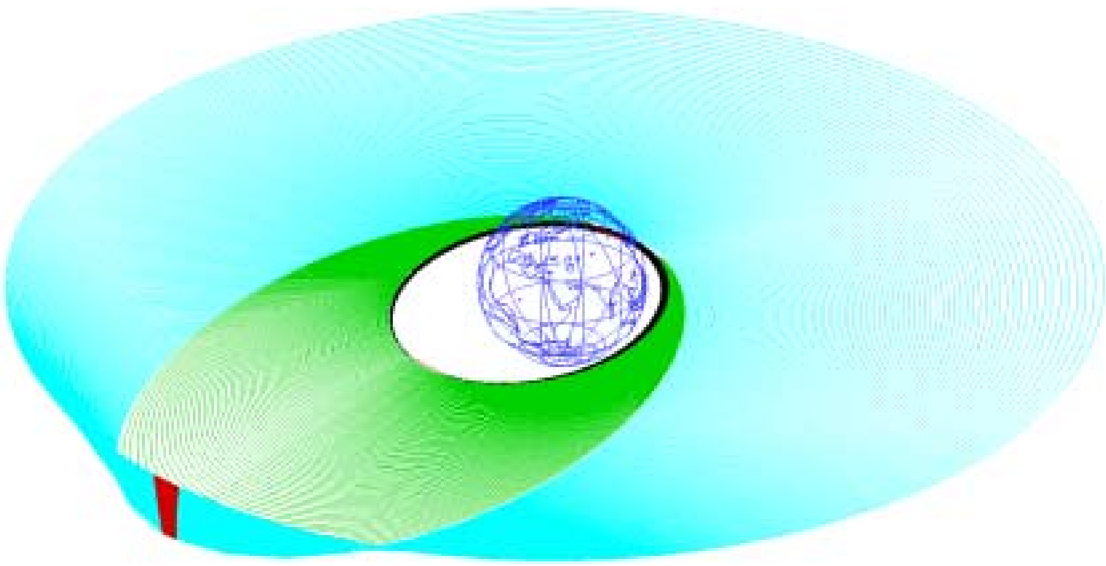
Figure 7.1.3 Side view of dual cavity receiver, 1 hour storage



8. TRAJECTORIES

DLR studied several trajectories options: the most obvious one is Hohmann transfer : it provides a minimised ΔV but the transfer time is very long (84 days) i. e. very similar to orbit transfer with electric propulsion. In addition, the time between firings is close to one day while reaching GTO and this is a sizing factor for the tank thermal design. Figure 8.1.1 shows the "Hohmann like" transfer trajectory.

Figure 8.1.1 Hohmann like transfer trajectory.



The use of opposite focal strategy (firings at perigee and apogee) provides a reduced transfer time (47 days) but the propellant mass increases (2846 kg instead of 2415 kg)) This is a logical consequence of thrust strategy.

A even shorter transfer could be obtained by a more refined strategy "constant coast arcs": the orbit is first circularized to 8000 km by perigee boosts (the added advantage is that the vehicle is above protons Van Allen belt) and transferred by evenly separated firings to GEO; this is a good approximation of a spiral transfer. The transfer time is reduced to 32 days. In addition, the storage time is reduced to 1 hour from 3 hours enabling to divide the thermal accumulator mass by more than 2. This dry mass gain (~400 kg) compensates in great part the 460 kg supplementary LH2 mass.
Figure 8.1.2 shows the "constant coast"transfer trajectory.
The table 8.1.1 enables to compare the three transfer strategies '2 cases for constant coast).
Note: the "orbited mass" is not the useful mass. It includes the STOTS dry mass.

Figure 8.1.2 "constant coast"transfer trajectory.

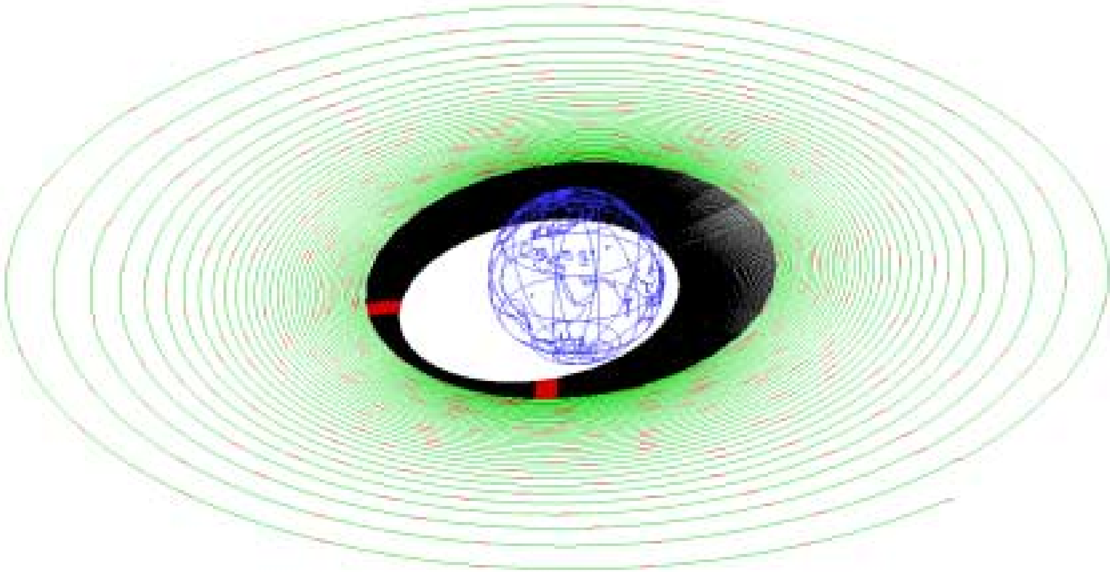


Table 8.1.1

	Hohmann like	**Opposite focal**	**Constant coast**	**Constant coast Reduced charg. time**
Final orbit	GEO	GEO	GEO	GEO
Inclination [°]	0	0	0	0
'in plane' TVC angle [°]	perpendicular to local horizontal	perpendicular to local horizontal	perpendicular to local horizontal	perpendicular to local horizontal
'out of plane' TVC angle [°]	11.3	12.5	17.5	19.0
Transfer Duration [d:h:min]	84:18:43	47:16:10	35:5:29	32:6:31
# of Van Allen belt crossings below 1000 km altitude	12	12	12	16
Total # of burns [-]	207	244	247	545
Applied dV [m/s]	2704	3313	3364	3430
Propellant consumption [kg]	2415	2846.7	2881.7	2880
Final / initial mass ratio [-]	0.69	0.63	0.63	0.63
Final mass [kg]	5435	5003.3	4968.3	4970.0

9. MISSIONS : description

9.1. Commercial missions

The initial performances and design of STOTS for GSO orbital transfer has been optimised and the overall design has been substantially modified.
The "cylindrical" mirror layout has been cancelled in favour of parabolic monolithic mirrors (in 3 segments). This enabled to have a better use of the fairing volume. In turn the maximum tank diameter and tank shape have been modified accordingly.
Of course STOTS puts heavy constraints on the Spacecraft:

- Mechanical interfaces (at launch)
- Structural role of the spacecraft
- Thermal control and electrical supply during orbital transfer.

The resulting overall layout is shown on figure 9.1.1. The figure 9.1.2 shows the deployed configuration in space.

Figure 9.1.1 Layout under ARIANE 5 fairing

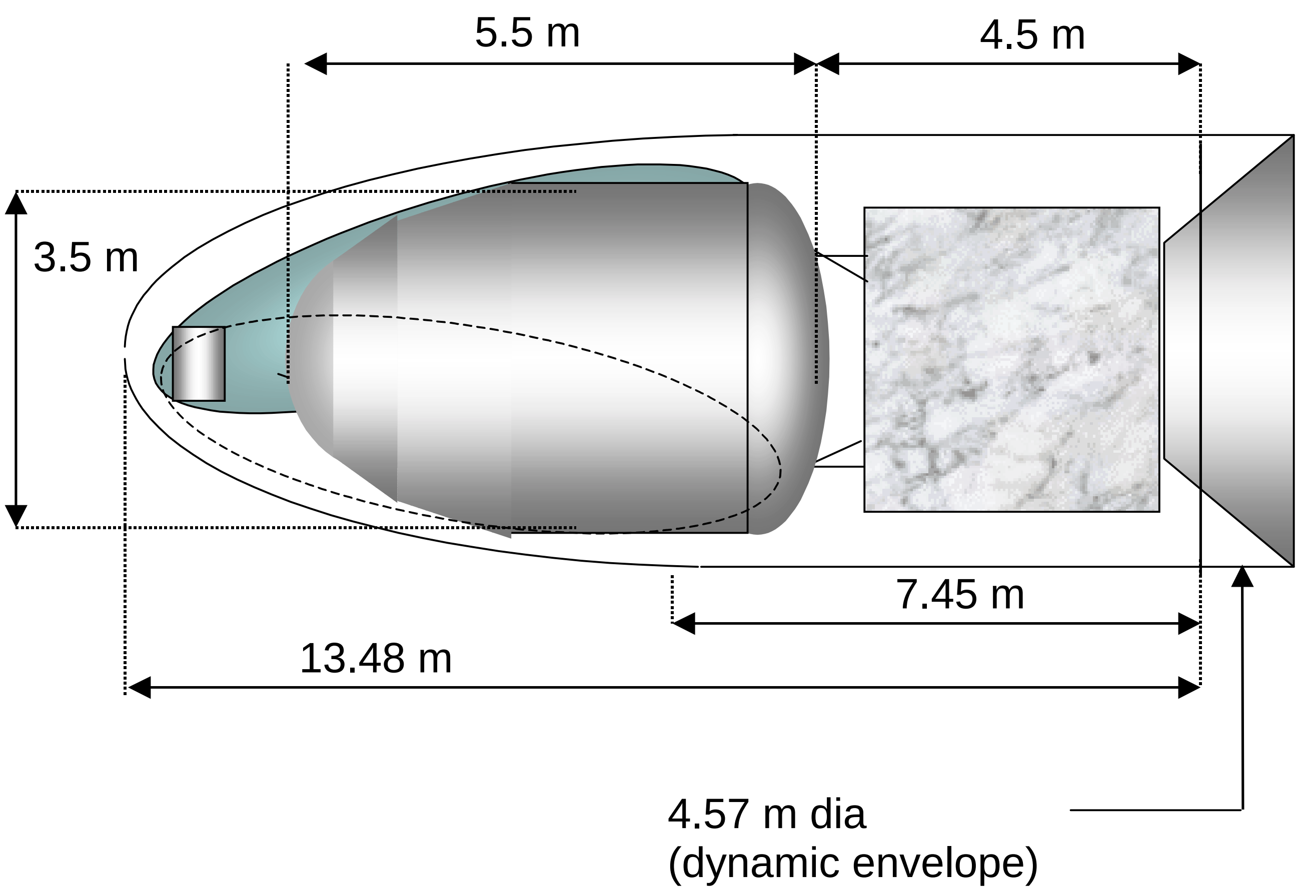
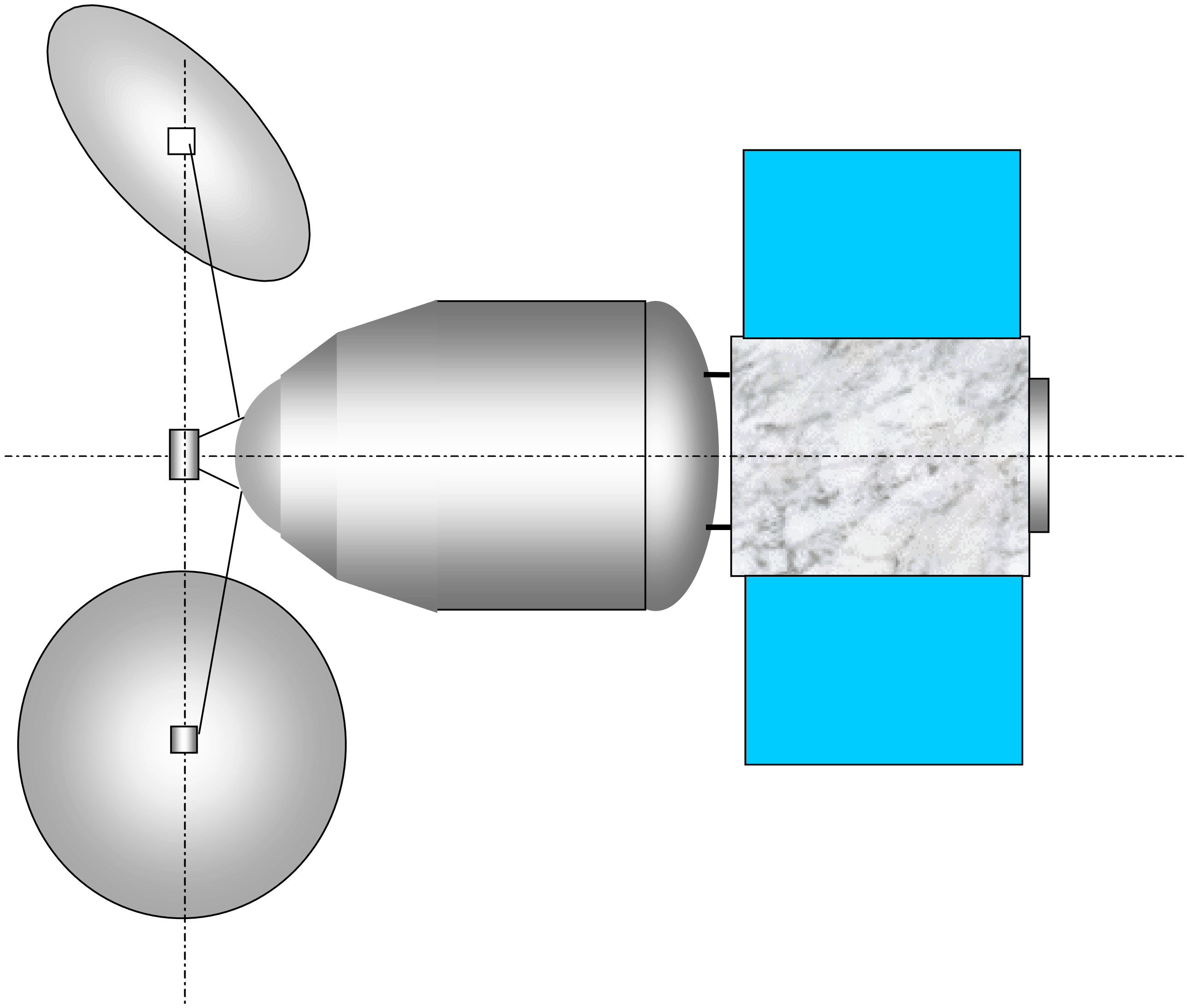


Figure 9.1.2

Receiver on Top of Tank (side view)



RLV applications :

RLV implies a LEO starting orbit. STOTS cannot start from this orbit for two major reasons:

- The atmospheric drag is very high in LEO (200 to 250 km)
- Eclipse complicates further the starting from LEO: available power is limited.

It is therefore necessary to use a chemical stage to raise the perigee.
However, a circular orbit introduces a slight Delta V penalty. It is better to use an elliptic orbit as in the case of ARIANE 5 : as the reference case, it allows to decrease the Van Allen belt stay, otherwise, the exposure to protons belt will be very detrimental to commercial spacecraft.

Among the various solutions to perform the chemical transfer, the preferred one is pressure fed LOX - LH2.
This presents the following advantages:

- The LH2 tank is common with STOTS.
- The Isp is quite high: 440 s
- The dry mass is very low (500 N engines, low LOX tank volume).
- The LOX tank size is limited.

The propulsion system layout will be very similar to the one proposed for planetary missions.

9.2. Planetary missions

STOTS offer the unique advantage to be usable as a propulsion system in a first phase and as a sunlight focusing device far from sun providing electrical power and heat, thus avoiding to use RTG (radio isotope generators), not available in Europe.
In other words, STOTS offer the unique opportunity to enable a scientific mission toward Jupiter / Europe or Saturn / Titan within the present European technical possibilities.

The power and heat source will use a solar panel backed by a radiator intercepting a variable percentage of concentrated flux depending on the sun - spacecraft distance. The required flux concentration factor is 27 at Jupiter distance and ~ 80 at Saturn distance.

The general concept consists in using STOTS for an outer planet mission in two different manners.

- In a first time, STOTS provides a significant part of the mission ΔV.
- In a second time only the primary mirror is used to focus the sunlight on a electrical converter in order to provide electrical power.

The waste heat of the electrical converter can be used to provide thermal conditioning of the payload (either by radiative exchange, heat pipes or fluid loop) thus allowing a mission toward Jupiter or Saturn without RTG.

At least one chemical propulsion module shall be included in the spacecraft:

- to perform injection in interplanetary orbit at perigee
- to perform orbit insertion around one satellite of Jupiter or Saturn and other orbital manoeuvres.

Two different chemical module types can be used:

- low-thrust pressure-fed cryogenic near earth.
- Earth storable or space storable for celestial body orbit insertion.

Useful mass performance :

The starting orbit is GTO, mass = 5500 kg (dual launch)
The table 9.2.1 provides the masses of propellants and equipment for the various phases of the mission and for several mission profiles:
2 AU implies subsequent Earth and Venus flybys to reach Jupiter EJO is feasible almost every year and the trip time is reduced but this is at the expense of payload.

Table 9.2.1 Mission mass budget

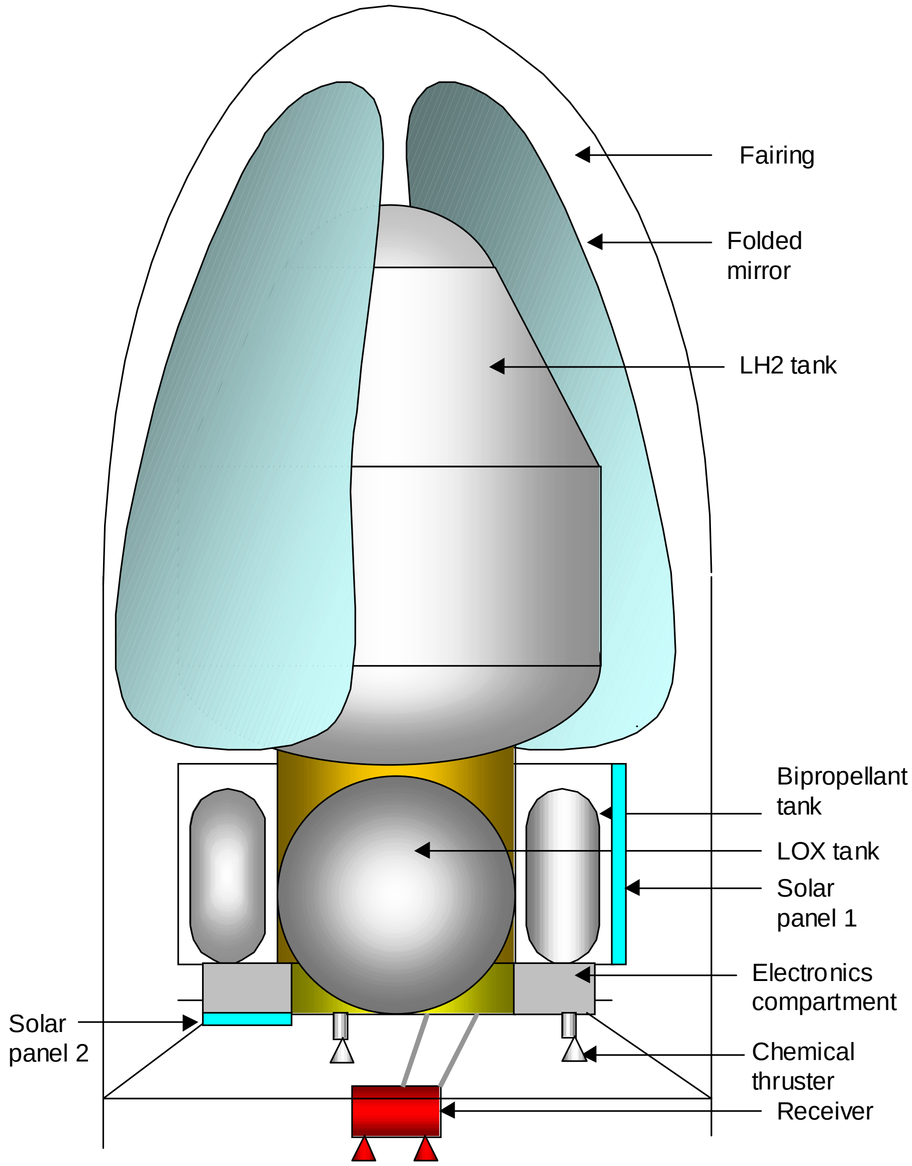
	Mars	2 AU	2 AU / EJO	2 AU / ESO
LH2 mass HEO	382,08	382,08		
LH2 mass impulsive burn, m r 6	92,78	194,76		
LOX mass impulsive burn, m r 6	556,68	1168,56		
LH2 mass EJO			1426,02	
LH2 mass ESO				1848,12
Total LH2 mass	474,86	576,84	2002,86	2424,96
(all masses in kg)				
Tanks mass	122,81	173,80	459,00	543,42
Mirror mass	80	80	80	80
Receiver mass	400	400	400	400
Total dry mass	602,81	653,80	939,00	1023,42
"Useful" mass	3865,65	3100,81	1389,58	883,06

Symbol keys : EJO : Earth / Jupiter Orbit (ESO : Saturn)

The figure 9.2.1 shows the interplanetary probe configuration under fairing.

Figure 9.2.1

CONFIGURATION UNDER FAIRING



10. SYNTHESIS

Performances

<u>Analysis of the results</u>

For both optimisation cases, the optimum starting orbit apogee is located between 5000 and 8000 km .

<u>Dry mass reduction:</u>

Two equipments offer the best potential for mass reduction: the LH2 tank and the accumulator.
LH2 tank mass gains are limited. The best benefit can be obtained by acting on accumulator.

Two ways are considered:

- Decreasing the thermal capacity (hence the mass) using several firings at perigee instead of one (steady firing with sun acquisition is still considered too risky).
- Design of a two stages accumulator with first stage (cooler) using a phase change material.

Dry mass was initially estimated at 950 kg for STOTS (40 % of propellant mass <u>2400 kg</u>).
The contributions of DLR, CSL and CRTBT provide the following values framing the target value.

	Standard	Advanced
• Mirrors	200	100
• Receiver	700	300
• Tank	360 (15 %)	270 (11%)
• Total	1260	670 kg

The values selected for the performance estimate (2800 kg) are intermediate:
Mirrors: 140 kg (2 kg/kW + arms)
Receiver: 400 kg (1 hour storage, CVD rhenium layer, integral manifold)
Tank: 460 kg

The following tables give the orbited mass, useful mass and mass gain for Hohmann transfer and for time optimised transfer.
The computation is of course valid for a new ELV of the 7500 kg GTO class.
Clearly the time optimised mode provides a lower gain but this was expected.

Apogee	Half major axis	Perigee speed	Delta V GTO	Delta V lib	Orbital period	ARIANE 5 performance	Initial Mass
5 000	8 978	8,768	1,48	2,247	2,35	24 871	8553
8 000	10 478	9,123	1,124	1,891	2,96	23 078	**7926**

Hohmann LH2 mass	Hohmann dry mass	Hohmann S/C mass	Hohmann Gain
2849	1140	4564	**1234**
2378	951	**4596**	**1266**

Constant coast mode	LH2 tank	Receiver	Mirrors	STOTS total	GEO spacecraft	Gain
LH2 mass 2800 kg	460	400	140	3760	**4126**	**796 kg**

Units: mass = kg, altitude = km speed = km/s
Reference GEO mass: 3330 kg useful mass = GEO mass less bipropellant system dry mass
(GTO mass = 5840 kg)

To increase the performance STOTS + electric propulsion provides even better results.

STOTS + electric propulsion

The power is set by the spacecraft anticipated performance as indicated in "electric propulsion characteristics" i. e. 20 kW.
The required Delta V is deduced from power and transfer-time and varies between 1100 and 1500 m/s.

- The power for 4 thrusters is 20kW (5 kW each) i.e. 21 kW at the input of the four PPU and 24 kW total spacecraft BOL power (20 - 22 kW EOL power depending on solar cell type).
- Total thrust will reach 1.42 N (14 kW / N) thus giving a total impulse of 7.361 MN.s in **60 days** and a xenon consumption of **500 kg**.

Total transfer time is 90 days, 30 for STOTS, 60 for EP. **Thus 500 kg of Xenon replace ~ 1 ton lH2**.

Hybrid mode

LH2 mass	LH2 tank	Receiver	Mirrors	STOTS total	MEO spacecraft	total GEO
1900	296	257	90	2542	**6026**	**5585**

STOTS dry mass = 642 kg

Useful GEO mass (after STOTS JETTISON) = **4943 kg.**

Commercial acceptability:
The most critical point is the impact of STOTS on telecommunication spacecraft interfaces, this requires substantial modifications on spacecraft structure.

Scientific missions:

STOTS (combined with chemical propulsion for interplanetary injection) offer an innovative solution for missions to outer planets without any need of RTG.

11. FUTURE WORKS / FLIGHT EXPERIMENT

11.1. Critical technologies and technology readiness

The following table summarises the findings on technology readiness.

Table 11.1.1 Identification of critical technologies and of their readiness level

Technology	Level	Origin	Required effort
Primary mirror			
Large CFRP off axis parabolas	8	Reflectors for telecom	Size extension to 6 * 3.5 m
Reflecting layer	8	mirrors	Size extension
Conformal layer	5 (US)	COTS	Size extension
Actuator	8	Antenna pointing mechanism	None (TBC)
CFRP arm	6 / 8	Space structures	Size extension
Secondary mirror			
Rhenium parts	COTS	Industry	Mirror development
LH2 Tank			
Metal tank /	8	H 10, ESCA	Mass reduction
Composite Vessel	2	laboratory cryostat	Demonstrator required
superinsulation	8	Standard S/C, ISO	Size extension
Immersed pump	8	Columbus water pump	Adapt to cryo fluid
Brushless cryogenic motor	8	ISO superconducting stepper motor	Technology applied to pump immersed motor
Spray bar	2		Demonstrator required
Technology	**Level**	**Origin**	**Required effort**
Receiver			
C- C accumulator	COTS	Furnace tooling	Development
Rhenium coating	COTS	X rays	Size increase
Superinsulation 1800 K	8	Space furnaces	Size increase
Superinsulation 2300 K	4 / 5	Space furnace DM	Size increase
2300 K vessel	2	Furnace cartridges	Welding / size
Orientation mechanism	6 / 8	Antennas pointing, chemical thrusters TVC	Dedicated design
H2 regulating valve	3 / 4	Proportional valve for gas generator	Adaptation of section. Hermetic closure

Autonomous control			
Autonomous navigation	4 / 5	ARTEMIS orbit raising	
2 axes 6' arc pointing	8	Scientific / GSO satellites	Adaptation to low frequency modes
TVS control	2		Fine tune by tests
Receiver cycling	2		Mirror pointing on / off H2 flow control

This table shows that most required "hardware" technologies are at hand in Europe. They need to be combined to form STOTS subsystems.

Spray bar and accumulator sealing are the least developed items.

Composite cryogenic tank option will also require a significant development effort but is paying off for many applications.

11.2. Synergies with other applications

Primary mirrors, superinsulated LH2 tank and cryogenic liquid phase control / separation can find other interesting applications:

Primary mirrors:

Besides the outer planets mission, large lightweight mirror have two types of application:

- Thermodynamic conversion.
- Large mirrors for millimetre wave astronomy payloads.

Several mirrors, 6 m long each will enable to form a very precise and fairly large radio telescope (0.01 mm error on 12 m).

Superinsulated LH2 tank

This technology will be mandatory for:

- cryogenic upper stages for interplanetary missions,
- low-cost cryogenic upper stage (LCCP),
- Mars manned missions (either chemical or nuclear thermal).

11.3. Critical technologies development

Critical technologies are of interest for several applications.

In order to reduce their development cost, they could be sized to fit a STOTS space experiment (subscale).

It is proposed to develop three demonstrators:

- Receiver.
- LH2 tank
- Primary mirror (and secondary).

11.4. Preferred development path :

Ground development

It is suggested to develop separately the major subsystems: tank, receiver and primary mirror. The receiver test facility can be updated for a conversion into coupling test (primary mirror / receiver).

Performing the developments at experiment size (4.4 kW per mirror) will considerably ease ground development. It will allow using a sun simulator for receiver development for example.

Flight experiment

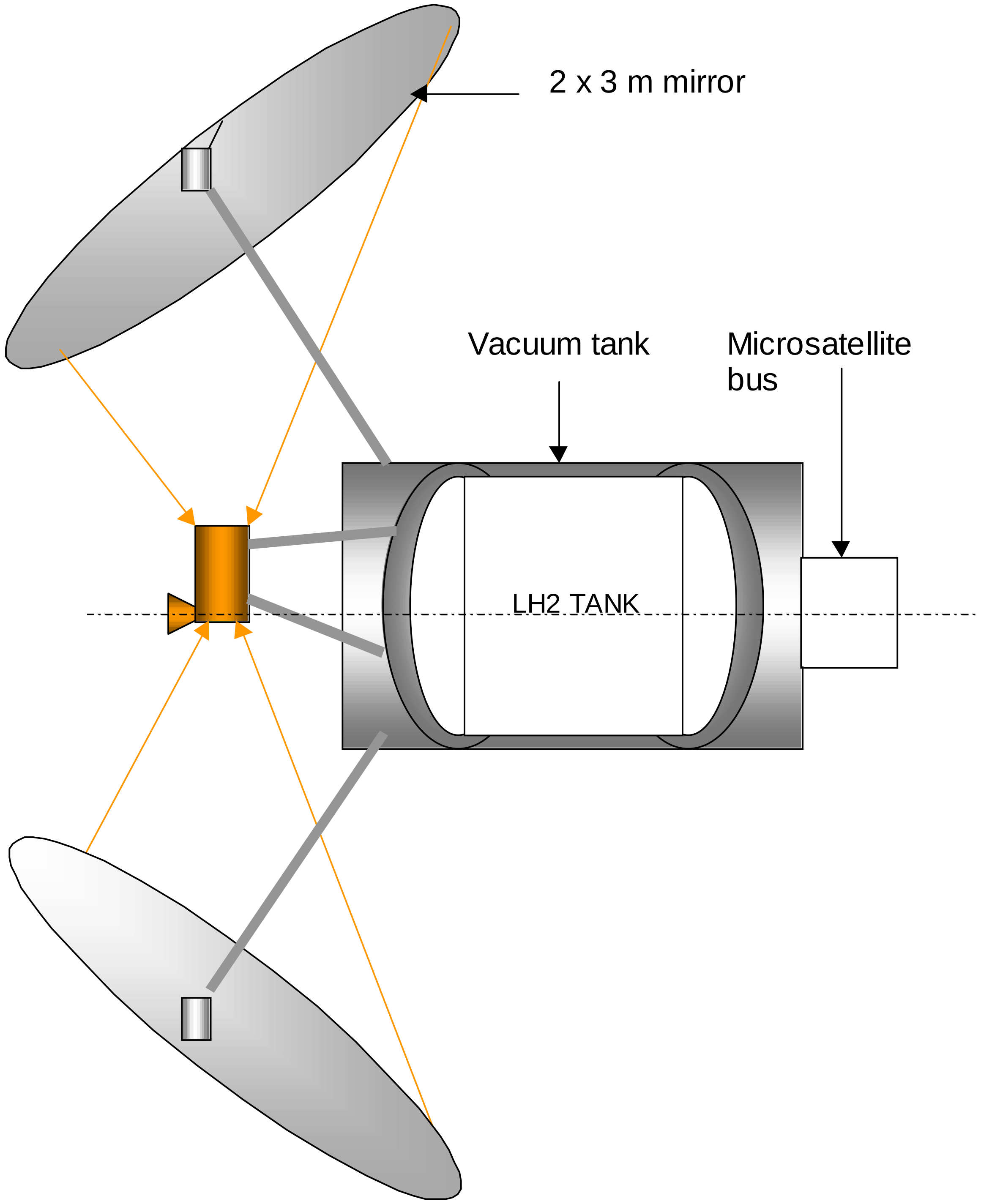
The simplified experiment (reduced size, LH2 stored in vacuum insulated tank) is preferred: it is still manageable, poses no major safety or launch pad waiting time problem, does not need launch tower modification. The figure 11.4.1 shows a typical experiment in flight configuration with 2 rigid mirrors deployed.

It allows the verification of almost every critical point except size and (possibly) life.

One possibility is a launch on ARIANE 5, possibly on top of lower commercial satellite in SPETRA, (the experiment may be too large to fit on ASAP interface). The operation will consist in performing a near GTO-GSO transfer enabling to test various firing strategies (including continuous firing and its effect on sun pointing).

The other possibility is a launch on VEGA.

Figure 11.4.1 Flight Experiment (Deployed)



12. CONCLUSION

The STOTS study led to the selection of the following technical options:

- rigid mirrors made of several panels
- accumulation with a reduced accumulation time, the main advantage being to eliminate stringent pointing requirements during firings.
- Constant coast time firing strategy
- Metallic LH2 tank with hybrid thermal insulation (foam and superinsulation).

All these techniques are either within hands in Europe or reachable within a few years.
The most critical problem is the high precision sun pointing.
This is why a flight experiment seems necessary, a reduced scale experiment has been proposed.

It will be very useful to decrease the dry mass of STOTS. To this end, it is advisable to continue R&D on advanced receiver / heat storage technology and tank mass reduction.
Mass gain for commercial mission is around one ton for GEO mission. Despite this great advantage, the interface requirements on the hosted spacecraft might severely reduce the customer interest.

More important performance gains are also expected by a combination of STOTS and electric propulsion.

STOTS offers very interesting perspectives for scientific missions especially to outer planets and Mars. This enables to take advantage of a half ARIANE 5 payload in GTO to perform a heavy scientific mission.