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ELV: Pressure Fed LOX/LH2 Upper Stage

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ABSTRACT

Cnes is leading extensive studies for the future generation of ELV's; The trend is today to research easy to use, robust and reliable solutions for the propulsion of future expendable launch vehicles but also with a high level of performances associated to as low as possible recurring cost of the launch vehicle. Among the different configurations (i.e. "Linear", Linear with SRB's, Tri Stages, Parallel with cross feeding) conventional two stages architecture seems the one of the best to reach these goals. In the two stages configuration, there is a competition between a SRB (500tons of propellant), a LOX/LH2 (350tons of propellant) or even a LOX/Methane, but always the second stage is cryotechnic (LH2/LOX), different technological solutions are possible.

A way could be an extensive use of composite materials both for the structures, the tanks and the engines with use, as much as possible of automatic processes. In a near future a breakthrough or improvement of the fibers strength may increase the interest of such technologies.

The aim of this paper is to present the technical choices for a 40 tons of propellant second stage leading to a cost effective configuration.

INTRODUCTION

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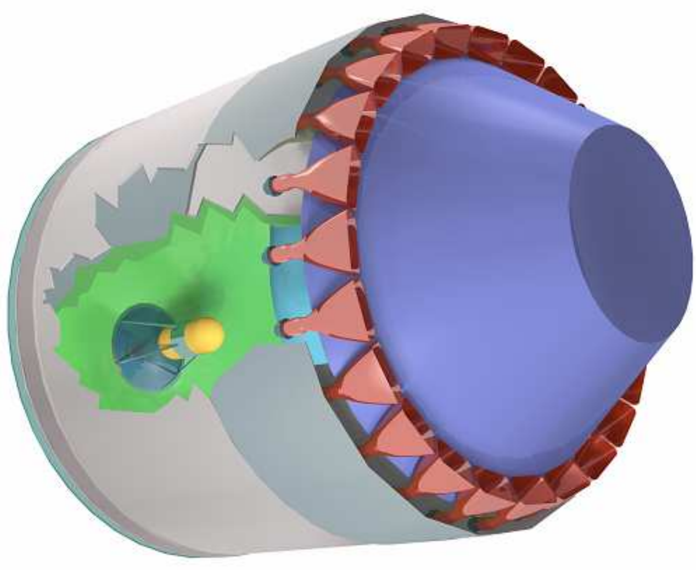
Liquid composite wound tanks can weigh three to four times less than a metallic one under the same operating conditions, the same results can be obtained for the chambers; So these technologies give a new interest to pressure fed solutions (as a first approach Vacuum Specific Impulse depends mainly on engine area ratio and less on operating pressure)

On an upper stage with a full diameter tankage system (with a common bulkhead),the use of composite tanks is enabling creation of room, or all around the aft dome of the tank or in length (moreover mass of



composite tanks is less sensitive than metallic ones to the shape).

Creation of thrust depends mainly on the possibility of implementing nozzles divergent surfaces.

On another side, thermal fluxes depend greatly on the operating pressure: low pressure engine no longer needs regenerative cooling system (radiation+ film or transpiration cooled ceramic engines could be used) Increase of throat area ratio is not a major problem with respect to the available room.

Moreover stage without turbomachinery is much less sensitive to the mission profile and become much more competitive with a multi-boost mission (GEO mission)

As a previous study [R1] showed the interest of these solutions, the aim of this presentation is to demonstrate the performance of a two stages launch vehicle with 40 tons of propellant second stage.

Interest of a pressure-fed upper stage/Stage Architecture configurations

Concerning the tank, wound composite tanks are able to withstand a much higher level of internal pressure for a mass lower than for metallic ones. Such a solution is without interest for a turbopump-fed system but greatly increase the performance of a pressure-fed system.
The specific impulse of an engine depends mainly on its area ratio and not on its combustion pressure (some % between 1 MPa and 7 MPa for an area ratio in the range 200-300 for the ODE Specific Impulse and Kinetic losses variations).
A pressure-fed upper stage may have a specific impulse of the same order of magnitude as a turbopump-fed stage if its engine may have a large area ratio.
Operating in a blow down mode for the last part of the flight allow a longer operating time (performance increase) and a lower maximum acceleration and so a much better fitting with a two stage configuration
Contrary to the Isv, the thermal fluxes depend greatly on combustion pressure ($P^{0.8}$), the choice of a combustion pressure close to 1 MPa enables the use of simple, lightweight cooled CMC engines. By comparison with an engine operating at 7 MPa, fluxes are reduced roughly by a ratio of 6.
When looking at the conventional solution, it appears that a conventional in-line (tank + engine) layout will be very long,.
Alternate solutions with a cluster of small engines (Plug) may use the allocated space better although they are handicapped by a lower engine combustion pressure. The engines may be implemented in the annular space, about 5m in diameter, between the tank aft dome and the interstage skirt taking into account the possibility of a special shape design for the aft dome. If the number of engines is large, their unit thrust will be low and they can be directly attached to the skirt, avoiding a heavy thrust cone. There is a large amount of available space, enabling the use of low-pressure engines.
Moreover, clustered low-pressure engines or engine with an appropriate idle mode (easier to realize on a pressure-fed engine) may add the following advantages:

- Stage separation and initial propellant settling system are no longer necessary : One or two seconds before the lower stage tail-off, four unit engines (plug) re ignited to maintain the propellant settled and to operate stage separation. Such a sequence is not very easy to select with an high thrust expander (Lower stage integrity) equipped with an Extendible Exit Cone(EEC integrity);
- Attitude control and thrust vectoring can be achieved by on-off modulation or thrust throttling (no need for movable engines)
- Development cost could be reduced by the need to develop only one small engine

Plug solution do not use turbomachinery, nor regenerative cooling circuit with the following major advantages:

- They do not need a heavy specific thrust frame as previously mentioned;
- Better potential reliability,
- Easier in-flight re-ignition,
- No need to satisfy any pump NPSP, so thermal management of propellants is much less critical and may lead to a lower amount of pressurant .

These two last points lead to a better fit to multi-boost mission (relative insensitivity to the mission profile)

For competitive performance levels, there are a few major drawbacks:

- For cryogenic use, composite tanks are not in the European SOTA
- The mass of Plug could be heavy except if CMC materials are used. These technologies are not yet in the European SOTA, but a strong R&D was done on Plug
- Feed of clustered engines could be heavy and somewhat more complex.

<u>Operating conditions/ basic hypothesis</u>

The basic hypotheses for this study are:

- Stage lay-out consistent with the allocated space or shorter if possible;
- Roughly 1 MPa combustion to be in the SOTA of radiatively cooled engines;
- End of mission blow-down operating mode to minimize pressurizing gas masses and therefore the end of mission mass
- The thrust at the beginning of the flight is 300KN , except for the end of expulsion (blow-down mode), ratio of about two on the thrust, and a slow decrease in the acceleration level.
- Engine mixture ratio 5.6, Stage O/F 5.85
- Diameter 5,4m
- Loaded Propellant:: 40 000 kg

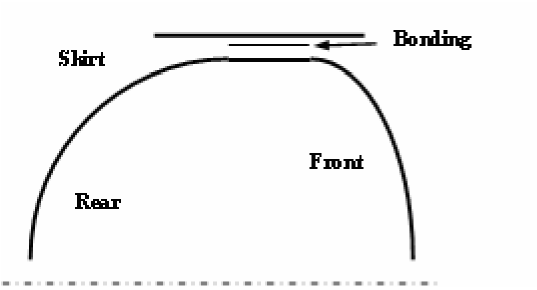
Technical solutions

<u>Tanks</u>

	LOX	LH2
Loaded mass (kg)	34161	5839
Tank Volume (m3)	31,54	86,57

The internal tank pressure is 1.85 MPa.(Burst 2.35) corresponding to minimal thicknesses. The pressure has to be a little higher in the LOx tank than in the LH2 tank due to the common bulkhead. This can lead to the requirement for special propellant and pressure management during the stage mission (ballistic phases).
The liquid hydrogen tank is an axi-symmetrical composite structure with 2 domes and a cylindrical part. It is made of one piece by filament winding. This cryogenic tank is equipped with an external cold insulation due to the relative high MEOP and with an internal liner(so microcracking in composite walls are acceptable) fabricated using two thin layers of aluminum foil bonded together (see [R2]):

As this liner system is not load-carrying (sized only to prevent from leakage), high strength carbon fibers must be used in order to optimize stage mass,. T1000G fibers have been selected.



As the rear dome shape has been designed mainly in function of stage arrangement constraints, this shape is not optimized from the filament winding point of view. So, the winding sequence defined in order to optimize the front dome and the cylindrical part is not fitted to the rear dome:The winding sequence defined for the front dome and the cylindrical part of the tank is enough for the rear dome axial flux but must be completed in the circumferential direction (flux < sizing flux). Adding some circumferential reinforcements made with fiber placement enables to withstand internal pressure loads:

The liquid hydrogen tank is equipped with two metallic polar bosses. The interface between composite walls, the liner system and the bosses is one of the major singularities. One of the best concepts is the packing gland-type boss with reinforced domes. Three of the main metal selection criteria are the chemical liquid hydrogen compatibility, mechanical characteristics at 20K and the thermal contraction at 20K.

As the cylindrical part of the tank is very short, only one continuous skirt is defined in order to simplify the architecture:
This skirt will be bonded on the tank without rubber, the strains of the composite envelope in the skirt attachment area must be limited by axial reinforcements made by hand lay-up or tow placement.
Except for the part bonded to the tank, the technology of the skirt is assumed to be the current technology used for interstage: (Carbon/Aluminum honeycomb sandwich)
The LOx tank, inside the LH_2 tank, is made of 2219 aluminum covered on the LH_2 side with insulation-filled honeycomb, itself covered with an A5 aluminum liner. This upper part is a common bulkhead, it is provided with a central opening for installation of the winding mandrel.

Structural materials of tanks

At 20 K	2219 T87 Aluminum	IM-T1000 Composite
σ_R (MPa)	575	2270
E (MPa)	80,000	180,000

Tank Thermal Insulation

The selected solution for the tanks is an external configuration (Klegecell) Insulation material is an expanded tight polyurethane foam (Klegecell 51 kg/m^3). The thickness will be 21 mm for LH2 part and 15 mm for LOx aft dome. The external part of the front LH2 bulkhead is insulated with additional MLI (Multi-Layer Insulation) to meet the payload cooling requirements.
The common bulkhead is made of a honeycomb structure able to withstand a small negative pressure difference between LOx and LH2 tanks
The ceramic insulated structure used as Aerospike ramp externally protects the aft LOx bulkhead from engine fluxes. LOx aft dome and pressurization tank are protected against engines plume fluxes by a blanket (Internal Multi-Screen Insulation) composed of several layers (metallic radiation shield –0.25 µm- covering a 20 µm ceramic sheet). Spacers are made of alumina ceramic felt. This multi-layer is covered with a 0.4 mm protecting bag made of Nextel 440 ceramic fiber.

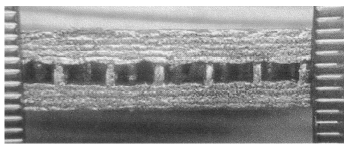
Engines

To obtain 300 kN, the Aerospike ramp is fed by 24 engine modules of 12.5 KN with an As/At =35
The equivalent area ratio of the truncated plug nozzle is in the range 100 for a 1.25 MPa chamber pressure.
The injector could be of the pintle type to enable thrust modulation, if this solution is preferred to on-off modulation for attitude control.
The engine chambers (basic solution) are made of ceramic/ceramic.

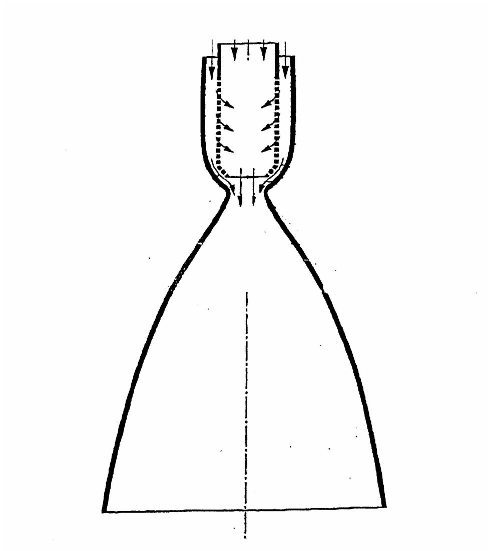
Min. Thick.= 2mm
Density =2800 kg/m^3
Θmax =1800K

Snecma NTO/MMH Ceramic test Engine

Two transpiration solution are under study in Germany (DLR and EADS ST) that can improve greatly the behaviour of the nozzle

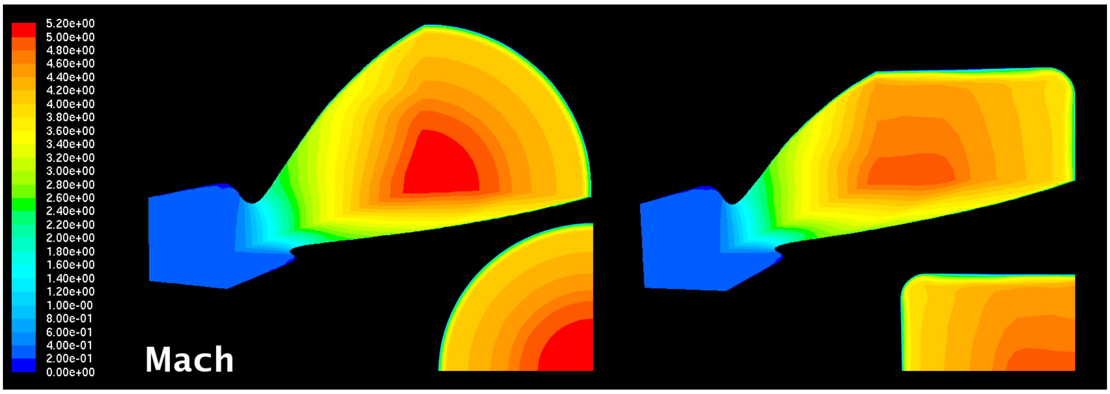


EADS C/C PTAH- SOCAR



Rectangular versus Circular Nozzle Exit

In order to obtain an estimate of the efficiency of the entire propulsion system, the aerodynamic losses of a nozzle with rectangular exit compared to a circular exit has been done with a 3D commercial Navier-Stokes code.
Geometry: The sub-sonic part of the two rocket chambers are completely identical. The rectangular nozzle shape is derived under the constraints that the overall nozzle expansion area ratio of Σ=35 is conserved

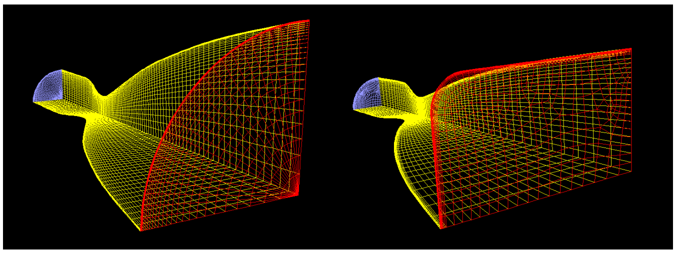


Comparison of the Mach contours.

Mesh:
The mesh for the axi-symmetric nozzle with circular exit has been defined in a similar manner in order to avoid variations due to the utilization of two different numerical schemes (2D and 3D). The mesh size is as follows:

- Circular: 72000 elements
- Rectangular: 75870 elements

The following Image show an isometric view of the two meshes.



Isometric view of the two meshes

The circular nozzle mesh is structured with hexa-elements and wedge elements at the axis. A structured mesh for the rectangular nozzle turns out to be rather distorted, therefore a paved scheme has been applied on the inlet and exit paces, which is projected through the volume.
Finally the mesh is refined at the wall to account for the boundary layer. 8 rows have been added with the cell height increasing by a factor of 1.3 starting from a value for the first cell height of 0.1 mm at the wall.

The computational model is characteristics are as follows:

- 3D Navier-Stokes
- Spalart-Allmaras 1-equation turbulence model
- No-chemical reactions
- Constant hot-gas composition: Ideal gas
- Specific heat capacities c_p are functions of the temperature (polynoms)

The calculation does therefore only allow a relative comparison between the two geometries but not an absolute prediction of the nozzle coefficient.

Results

The nozzle with rectangular exit results in an I_{sp} loss of 5.6 seconds or 1.3% compared to the nozzle with circular exit. An identical result has also been obtained in a calculation with constant c_p. Considering that no particular analytical method has been employed to generate the rectangular nozzle geometry an increase of I_{sp} can be expected if such tools would be used to optimize the nozzle shape. Concluding it can be said, that the loss in performance is surprisingly low which however corresponds to results obtained through others communications. So, the estimate of the specific impulse of the unitary nozzle is 425 s with an efficiency of 0,941
The ramp increase the area ratio from 35 to 100, the specific impulse increase due to the ramp and to the plug base is estimated to 20s.
The average specific impulse was estimated at 445 s

Pressurization system

To limit the penalty on the mass budget and the maximum acceleration, a blow-down operating mode is considered at the end of the mission. The tank pressure is assumed to decrease from 1.85 to 0.7 MPa.

Different solutions were studied and the following basic solution was selected:
Both tanks are initially pressurized with helium gas stored in high-pressure vessels at ambient temperature, located on the first stage and then pressurized by heated evaporated hydrogen. A heat exchanger will be implemented inside the ramp (plug). The helium tank is located inside the hydrogen tank to reduce its size. After a trade-off study a Titanium alloy tank was selected.

Mass Breakdown

Loaded Propellants	40 000
Helium	72,5
Engines & Ramp	540
Tank, Insulation & Equipts	2142
Press Tank & misc.	73,5
Additionnal Insulation (MLI,..)	72
CGRS	14
Miscellaneous (Ducts,…)	294
Electrical System	143
Pyro	20
Gaseous Residuals	480
End of mission mass	3 855
inter-stage skirt+ rear frame	592
Total Lift-off mass	44 468
Struct. Index (%)	9,8

Conclusion

The resulting performances –structural index close to 10% for a vacuum specific impulse of 445 seconds, an association with a LOX/LH2 first stage with a loaded propellant mass of 350 tons with a structural index of 14% will allow a payload mass - into a Geostationary transfer orbit- close to 8 tons. So, the performance study confirm the interest to follow on in the study of such a solution and it looks mandatory to implement R&D in the three following fields:

- composites cryotechnic Tank
- composites engines and ducts
- variable thrust engines

Different variants of this pressure-fed solution are under study with the aim to improve the performance or the cost effectiveness .

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