LOX/LCH4 UPPER STAGE DEVELOPMENT STRATEGIES FOR FUTURE LAUNCHERS

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ABSTRACT: The reduction of Earth-to-orbit launch costs in conjunction with an increase in Launcher reliability and operational efficiency are the key requirements of future space transportation systems. This paper underlines the progress in LOX/CH4 upper stage engine development carried out by COMOTI and also being provided the prediction of the rocket engine performances at the conceptual and preliminary stages of design. This paper focuses on the trade-off studies for the engine architecture definition, considering both open and closed thermodynamic cycles. Various subsystems configurations have been taken into account, analyzing the optimum configuration in terms of performance. The main operating and geometrical parameters were discussed: combustion pressure, optimum mixture ratio, turbine pressure ratio, thrust chamber geometry, and the turbopump size is addressed.

KEYWORDS: upper stage, liquid rocket engine, turbopump, LOX/CH4 cryogenic propellant, thrust chamber

NOMENCLATURE

\( A \) – turbine characteristic area
\( A_e \) - nozzle exit diameter
\( A_t \) – nozzle throat diameter
\( A_c \) – combustion chamber area
\( T_e \) – combustion temperature
\( \gamma \) - specific heat ratio
\( C_p \) - specific heat capacity
\( p_c \) – combustion pressure
\( C_f \) – thrust coefficient
\( c^* \) - characteristic velocity
\( g_0 \) – gravitational constant
\( x_{pf}, x_{pm} \) - axial width
\( \dot{m} \) – gas flow rate
\( P_{pox} \) – liquid oxygen pump power
\( P_{pfuel} \) – liquid methane pump power
\( P_T \) – turbine power
\( \sigma_p^* \) - turbine pressure losses
\( R \) – gas constant
\( \rho \) – density
\( M \) – Mach number
\( h_{pf}, h_{pm} \) – turbine blades height

1. INTRODUCTION

As the present trend in rocket engine development recommends a high versatility and low launch service cost, while preserving high performance, expander cycle upper stage based on LOX/CH4 being a key competitiveness factor recognized by the market. The new ESA-VEGA Development Program aimed at improving the VEGA competitiveness throughout three major high-level objectives: decrease of the Vega C Yearly Launch System Service Cost for lower price per launch into the reference orbit with a target of 10% as cost reduction in production and operations, Increase of the VEGA C Launch System Margins for higher mission flexibility complementarily with Ariane 6 Launch System and to Increase of the VEGA C Launch System Services Versatility for ad-hoc launch services solutions based on available products, including small spacecraft mission services (SSMS-C), dual – launch services (VESPA-C) and orbit transfer services (VENUS). Also, the introduction of VEGA-C that cover large part of the LEO payloads has led to focus main motivations for VEGA-EVO of the following main drivers:

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- Competitive: recurring Cost 20% less than VEGA C target price;
- Green: elimination of all the toxic propellants;
- Flexible: multiple altitudes and orbital planes;
- Scalable: a family of launchers suitable for different payloads.

As regards the upper stage, there is a pronounced need to develop a new class engine based on “green propellant” and relatively low power due to the development of low-costs and environmental options. The main distinguishing properties of the upper-stage compared to the first stage engines are as follows: the engine must start in a vacuum, while engines of previous stages continue to operate; the specific impulse must increase by using high-nozzle area ratios.

Fully cryogenic LOX/LCH4 is one of the most promising liquid-rocket-engine technologies due to the following advantages: reduced dry mass, compact architecture, cooling capabilities, reusability issues and mission flexibility. [4]

Methane is a soft cryogenic that is not corrosive and has low toxicity, therefore being easier to store, requires less insulation and fewer concerns than hydrogen fuel systems. Also, liquid methane is about six times denser than hydrogen; thus methane tanks require a smaller storage volume than comparable hydrogen tanks. Furthermore, methane has exceptional heat capacity properties that provide good cooling properties for expander cycle; methane fuel has lower pressure drops in regenerative cooling channels compared to kerosene fuel. In addition, methane is natural gas, being 5 to 10 times cheaper to acquire and store than liquid hydrogen. [18]

The highest risk involved in the expander cycle engine development is to match the coolant circuit capability to the engine requirements. The working pressure inside the combustion chamber imposed a sufficient amount of heat to be extracted from the coolant circuit to power the turbomachinery while still providing a suitable cooling in order to avoid structural damages. This risk can be mitigated through numerical analysis, combustion chamber components testing and engine test bench experimental testing. Any variation in the extracted energy can be controlled through orifices to adjust the distribution of flow between the combustion chamber and the turbine.

None launcher upper stage based on the combination LOX/LCH4 has yet flown, but ground testing of such systems is being carried out, as an example, the upper stage LM10-MIRA foreseen for the evolution of VEGA-E launcher.

Aim of the current study is to evaluate the performance of an oxygen/methane expander system intended to the upper stage with a specific focus on thrust chamber and feeding system design.

The clean-burning, non-toxic, high vapor pressure propellants provide significant advantages for reliable ignition in space. The main components in a cryogenic upper stage engine are the thrust chamber assembly; the propellant feed control system, the turbopumps, the propellant tank pressurization system, the electrical system, the hydraulic control system and the flight instrumentation system.

A liquid rocket upper stage can be divided into two main parts: a feeding system and thrust chamber assembly. The feeding system suitable for this type of application is based on turbopumps that are designed to provide the required energy to the propellant for an optimum combustion process. Another technique to characterize the engine cycle is based on the turbine and thrust chamber arrangement, the cycle is classified as open or closed. Advantages of a well-established power cycles include lower development cost, reduced development risk, repeatable starts and a compact assembly.

For each configuration, the following design steps were performed:
- Definition of the engine operational parameters;
- Thrust chamber geometry definition;
- Pump concept design;
- Turbine concept design.

This procedure allows estimating the engine performance based on sub-systems concept design procedure. For higher performance, mechanical pumps must be used to feed the combustion chamber.

2. LIQUID ROCKET ENGINE CYCLE DESIGN

Considering a thrust range of 30kN, a LOx / CH4 cycle analysis was performed to determine the most performant global architecture. The investigation has been channeled on three directions: gas generator cycle, staged combustion cycle (both fuel-rich and oxidizer-rich) and expander cycle. This analysis allows selecting thrust chamber performance, turbopump assembly power, thrust chamber size and cooling system configuration.
In the Gas Generator Cycle (Fig. 2.1.a) a small fraction of the pressurized oxidizer and fuel are directed through a medium-temperature burner (Gas Generator) which therefore produces a fuel-rich gas to drive the turbine. These are designed with large pressure ratio and the remnant gases are usually reintroduced in the exhaust nozzle, after the critical section in order to assure extra thrust.

For engines, utilizing hydrogen or methane as fuel, the gas generator could be replaced. Therefore, the fuel could be routed from the exit manifold of the nozzle cooling circuit to the turbine inlet. This could be possible because methane behavior is supercritical at the pump exit and follow a simple expansion procedure as gains heat. The resulting expander cycle is compact and efficient because all amount of fuel is used in the thrust chamber. The principal limitation of the cycle is the relatively small amount of heat available from the regenerative cooling, which limits applicability to combustion pressures under 7MPa. [5]

For rockets with high chamber pressure, the staged combustion cycle is the optimum solution. A small amount of oxidizer is added to the fuel after the cooling circuit.

For any cycle, the pump must meet the pressure and flow rate requirements of the turbine while in turn it should drive the pump within the pressure ratio limitation. Through upper stages with Gas Generator Cycle the combustion chamber pressure generally are maintained below 10MPa in order to avoid high turbine flowrates.
A pump-fed rocket engine with turbine extracted from the cooling system is referred to an expander cycle, being intended for upper-stage, considering its limitation in terms of available power when compared to staged-combustion and gas generator cycles. The absence of gas generator allows for easier start-up sequences and avoids the controlling of the mixture ratio of the gas generator. From other points of view, the expander cycle is not suitable for high thrust level, due to the limitation imposed by the power that could be extracted from the cooling process of the thrust chamber.

In the Expander Cycle (Fig. 2.1.c) the chamber pressure is limited due to the amount of heating available for the turbine working fluid. At the same imposed combustion pressure, the expander cycle requires a higher pump discharge pressure. For the stage-combustion cycle, the chamber pressure could attain a chamber pressure of about 20MPa because the preburner provide high-energy working fluid for the turbine [16]; 20MPa is actually an upper limit for the turbine drive cycle because for a given turbine inlet temperature and turbopump efficiency, the required pump discharge pressure rises at higher quotations. Expander cycles are reliable and have multiple restart capability, while the LOX/LCH4 offers high specific impulse with attractive bulk density and handling characteristics [19].

For the Gas Generator Cycle, turbine pressure ratios of about 20 are required in order to minimize the flow rate of the turbine working and therefore maximizing the specific impulse. For the expander and staged-combustion cycles, the optimum turbine pressure ratios are typically less than 1.5, because of the large turbine working fluid available. For that configuration, turbopump assembly weight is greater comparative to expander and staged-combustion cycle. These differences originate from pump discharge pressure differences. In order to gain high system efficiency, pumps in expander and staged-combustion cycle either operate at higher speed or have more stages than those in the gas generator

2.1 CASE STUDY

Recent studies for new generation upper stage cryogenic engines in Europe with the main goal to increase the reliability and performance, have engines with restart capability and low development costs identify expander cycles as a promising technology. The main elements of an expander cycle are a tank system, turbopump system, cooling system, injector system, main combustion chamber and exhaust nozzle [1].

For this type of technology, the heat exchange by regenerative cooling is a critical point in expander cycles as the hot gases formed need to drive the turbine. When is used a regenerative cooling technique, the coolant, which is the fuel because oxidizers at high temperature lead to corrosion issues, is passes in cooling channels along with the thrust chamber before it is redistributed into the combustion chamber [17]. The major limitation is the relatively low achievable combustion chamber pressure level, which ultimately limits the maximum thrust level. Therefore, the expander cycle is a proper candidate for an upper stage.

The LOX/LCH4 engine mixture ratio has been set to 3.36 in order to minimize the turning inlet temperature to 1100 K and to be far from the stoichiometric ratio. In order to avoid the two-phase flow in the cooling circuit, for the LCH4 case, the pressure must be higher than the critical pressure, of 45.9 bars. [16]

For this study, considering the potential of a higher density of liquid methane compared to liquid oxygen, a system with a single turbine acting both fuel and oxidizer pumps is considered. For an optimized cooling of the thrust chamber, the total mass flow rate flows in the cooling system.

3. PERFORMANCE EVALUATION AND THRUST CHAMBER PRELIMINARY DESIGN

Fuel, oxidizer and mixture ratio determines the start characteristics of the flow field in the combustion chamber and further the chamber geometry influences the nozzle performance. In this paper, it has been analyzed how a combustion pressure in 1-10MPa influences the mixture ratio choice for LOX/CH4 combination. Also, an expansion ratio \( (A_c/A_l) \) with a variation in the range of 10 – has been investigated.

The specific heat ratio \( (\gamma) \), the combustion temperature \( (T_c) \), the molar mass, the density and the specific heat capacity \( (C_p) \) for mixtures of fuel and oxidizer were obtained from CEA. From the specific heat ratio, the value of the Van Kerckhove function \( \Gamma \) [2] was calculated:

\[
\Gamma = \sqrt{\frac{2}{\gamma + 1}} \left( \frac{\gamma + 1}{2(\gamma - 1)} \right) \quad (3.1)
\]

The Mach number \( (M) \) at any point in the thrust chamber could be found from solving the area-Mach number relation:

\[
\left( \frac{A}{A_l} \right)^2 = \frac{1}{M^2} \left[ \frac{2}{\gamma + 1} \left( 1 + \frac{\gamma - 1}{2} M^2 \right) \right]^{\frac{\gamma + 1}{\gamma - 1}} \quad (3.2)
\]
The performance parameters such as thrust coefficient \( C_F \) and characteristic velocity \( c^* \) could be expressed with the following relationships [1]:

\[
C_F = \Gamma \sqrt{\frac{2\gamma}{\gamma - 1}} \left( 1 - \left( \frac{p_e}{p_c} \right)^{\frac{\gamma - 1}{\gamma}} \right) + \left( \frac{p_e}{p_c} \right) \frac{A_e}{A_t}
\]

\[
c^* = \frac{\sqrt{\gamma R T_c}}{\Gamma}
\]

Knowing the thrust coefficient and characteristic velocity allows calculating the thrust chamber specific impulse:

\[
(I_{sp})_{ideal} = \frac{F}{g_0 m} = \frac{C_F c^*}{g_0}
\]

Selecting the design point chamber pressure is a key decision in the development of a new rocket engine. Higher chamber pressure generally enables higher impulse. For an expander cycle, selection of the chamber pressure is important since the extracted power to operate the turbopump system is extracted from the combustion chamber cooling circuit.

![Fig. 3.1 Specific impulse function of expansion area ratio for different combustion pressures](image)

For Fig. 3.1 representation of the LOX/LCH4 engine mixture ratio has been set to 3.36 and the exit pressure to 0.01 bar. Nozzle exit conditions such as exit pressure, temperature or density are unknown; therefore, the change in conditions in the nozzle was performed step by step starting from the throat towards the nozzle exit. Therefore, the nozzle calculations have been done by dividing the nozzle into segments.

Through the performance evaluation section, for a thrust range of 30 kN imposed and LOX/LCH4 as propellant, the combustion pressure was varied in the range of 3-10MPa, with a step of 10, the compression area \( A_c/A_t \) between 1.2 ÷ 6, with a step of 1 and expansion ratio \( A_e/A_t \) between 10 ÷ 100, with a step of 10.

By varying all these operating parameters, it is intended to obtain a thrust chamber mass as small as possible, taking into consideration a specific impulse \( I_{sp} \) within the 370 s range.

After numerical investigation, it could be shown (fig.3.1) that the specific impulse increases with the expansion area ratio \( A_e/A_t \). After ratios greater than \( A_e/A_t > 70 \) the specific impulse growth rate is not pronounced. Also was demonstrated that the area contraction ratio does not influence the specific impulse which slightly increases with combustion pressure.

For what concern the thrust chamber dry mass is related by the following relation:

\[
m_{engine} = (0.001F + 49.441)N^{0.03}(A_e/A_t)^{0.004} [16]
\]
Fig. 3.2 Thrust Chamber mass dependence on $P_c$, $A_e/A_t$, $A_c/A_t$

Fig. 3.2 presents the variations on the dry mass of the thrust chamber. It can be seen that the mass of the combustion chamber and nozzle decrease as chamber pressure increases. This is expected because as chamber pressure increases, the throat shrinks for a given desired thrust and expansion ratio. Regarding turbopump, assembly mass is proportional to the power required by the turbine. As the combustion chamber pressure increases the power needed from the turbine increase to provide the pressure rise and consequently the overall turbopump assembly mass increases.

A high specific impulse is desired in trying to achieve a high-performance upper stage. Since expansion ratio increases the engine diameter increases, this was the limiting factor in choosing the expansion ratio. Given the specific limitations on exit diameter for upper stage engines, an expansion ratio of 60 was chosen.
From the graphics exemplified besides, thrust chamber mass increase significant with expansion area ratio and decrease with $A_c/A_t$ increment; the decrease is not as significant for the values $A_c/A_t > 3$.

Also, the thrust chamber mass decreases as the pressure increases in the combustion chamber. The minimum take-off mass was taken as an objective function. Since launcher development costs tend to vary as a function of gross take-off mass; therefore, minimum gross take-off mass vehicle may be considered as a minimum development cost concept.

The optimization function related to LOx/LCH4 upper stage performances is given by:

$$C = 0.5 \cdot \frac{m_{ref}}{m} + 0.3 \cdot \frac{V_e}{V_{e ref}} + 0.2 \cdot \frac{I_{sp}}{I_{sp ref}}$$ (3.7)

where $m_{ref} = 30.9502$ kg, $V_{e ref} = 3532.721$ m/s, $I_{sp ref} = 360.2373$ s.

The current analysis has been performed for a fixed engine thrust of $30kN$, combustion chamber pressure variable in the range 60-100 bar, expansion ratio between 10-100 and contraction ratio in the range 1.2-6. Our starting point for this analysis is the determination of the mass of the engine, being at the same time correlated with thrust. We determined, therefore, an optimum mass for a given thrust function on different operational parameters.

With the objective of obtaining a smaller mass but at the same time a specific impulse as close to 370 s, it was chosen as the optimal combination: $P_c = 60$ bar, $A_c/A_t = 4$ and $A_c/A_t = 60$.

For the full cryogenic upper stage design mainly focuses on determining the basic dimensions of the engine (Table 3.1). For this type of application, a cylindrical combustion chamber with converging-diverging exhaust nozzle.
### Table 3.1 Thrust chamber preliminary design parameters

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Thrust (kg)</td>
<td>30</td>
</tr>
<tr>
<td>Combustion pressure (bar)</td>
<td>60</td>
</tr>
<tr>
<td>Mass (kg)</td>
<td>43.65</td>
</tr>
<tr>
<td>Specific Impulse (s)</td>
<td>360</td>
</tr>
<tr>
<td>Total flow rate (kg/s)</td>
<td>8.5368</td>
</tr>
<tr>
<td>Oxidizer flow rate (kg/s)</td>
<td>6.58137</td>
</tr>
<tr>
<td>Fuel flow rate (kg/s)</td>
<td>1.95542</td>
</tr>
<tr>
<td>Lcyl (mm)</td>
<td>197.19</td>
</tr>
<tr>
<td>R1 (mm)</td>
<td>43.16</td>
</tr>
<tr>
<td>R2 (mm)</td>
<td>85.5</td>
</tr>
<tr>
<td>Dt (mm)</td>
<td>57.55</td>
</tr>
<tr>
<td>Rn (mm)</td>
<td>10.99</td>
</tr>
<tr>
<td>Tn (deg)</td>
<td>35.75</td>
</tr>
<tr>
<td>Te (deg)</td>
<td>8</td>
</tr>
</tbody>
</table>

Maximizing the specific impulse in an upper stage has a greater positive effect on payload capacity. Chamber and nozzle heat transfer scales proportionally with surface area exposed to combustion products. The surface area of a nozzle is proportional to the chamber diameter. For a constant chamber length, the heat transfer increases with the square root of the throat area. In order to minimize this effect, the chamber length could be gradually increased, but this will reduce the allowed nozzle length and area ratio (for a fixed engine length), increase the chamber coolant pressure losses and increase engine weight. Therefore, considering that the total heat transfer does not scale proportionally with increased thrust, expander cycle chamber pressure decrease as the thrust range is increased.

### 4. LOX/LCH4 TURBOPUMP SYSTEM

In the expander cycle, the power required to drive the pumps is proportional to propellant flow rates and pressure rise. In order to have low pressures in the storage tanks but still have high pressures in the combustion chamber a pump system is needed to increase the propellant pressure. Therefore, the component modeling goal is to increase the oxidant/propellant pressure and determine the power needed to achieve this.

This effort details the design of LOX and CH4 pumps required by an expander cycle upper stage engine. Materials must be found than can tolerate LOX and LCH4 (inducer and impeller). The impeller and diffuser must be designed in such a way to maximize throttle ability. [5]

The pump component determines the required pump power. It could be deduced from the mass flow passing the pump, the required pressure rises over the pump, the density of the propellant and efficiency of the pump. The LOX/LCH4 pumps follow from [16]:

\[
P_{\text{pox}} = \frac{1}{\eta_{\text{pox}}} \frac{\Delta p_{\text{ox}}}{\rho_{\text{ox}}}
\]

\[
P_{\text{pfuel}} = \frac{1}{\eta_{\text{pfuel}}} \frac{\Delta p_{\text{fuel}}}{\rho_{\text{fuel}}}
\]

More specifically the power balance can be written as:

\[
\frac{1}{\eta_{\text{pox}}} \dot{m}_{\text{ox}} \frac{\Delta p_{\text{ox}}}{\rho_{\text{ox}}} + \frac{1}{\eta_{\text{pfuel}}} \dot{m}_{\text{fuel}} \frac{\Delta p_{\text{fuel}}}{\rho_{\text{fuel}}} = \eta_{T} \dot{m}_{T} C_{p} T_{\text{in}} \left(1 - \left(\frac{p_{\text{out}}}{p_{\text{in}}}\right)^{\gamma-1}\right)
\]

The pumps designs involve a large number of interdependent variables. This section outlines some of the key parameters and features that have been considered in the design of LOX/LCH4 pumps.

The required pumps flows are established by the rocket design, exhaust gas velocity, propellant densities and mixture ratio. Also, the pumps discharge pressures are determined from the chamber pressure and hydraulic losses in valves, lines, cooling jacket and injectors.

As a first step it was determined the specific rotational speed with the following relation [10]:

\[
n_{s} = 3.65 \sqrt{\frac{Q}{H^{3/4}}}
\]

The inlet and outlet rotor diameter could be determined by the aid of the following relations:
LOX/LCH4 upper stage development strategies for future launchers

\[ D_1 = (1.1 \sim 1.5)K_0 \frac{\sqrt{g}}{n} \]  

(4.5)

\[ D_2 = 19.2 \left( \frac{\text{N}_{\text{sopf}}}{100} \right)^{1/6} \sqrt{\frac{2gH}{n}} \]  

(4.6)

The exit rotor blade high is given by:

\[ b_2 = 0.78 \left( \frac{\text{N}_{\text{sopf}}}{100} \right)^{1/2} \sqrt{\frac{Q}{n}} \]  

(4.7)

Blade number:

\[ Z = 6.5 \frac{D_2 + D_1}{D_2 - D_1} \sin(\beta_1 + \beta_2) \]  

(4.8)

Also another important parameter is the net pressure suction head [13]:

\[ \text{NPSH} = \frac{p_{\text{inlet}} - p_{\text{vapour}}}{\rho g} \]  

(4.9)

Using ANSYS Vista CPD the two pumps were pre dimensioned. Among the required data are speed, gas flow rate, working fluid density, pumping head, inlet flow angle and velocities ratio.

### Table 4.1 Pumps inputs design parameters

<table>
<thead>
<tr>
<th>LOX Pump</th>
<th>LCH4 Pump</th>
</tr>
</thead>
<tbody>
<tr>
<td>Rotational speed (rpm)</td>
<td>30850</td>
</tr>
<tr>
<td>Volume flow rate (m³/h)</td>
<td>20.783</td>
</tr>
<tr>
<td>Density (kg/m³)</td>
<td>1140</td>
</tr>
<tr>
<td>Head rise (m)</td>
<td>689</td>
</tr>
<tr>
<td>Inlet flow angle (deg)</td>
<td>90</td>
</tr>
<tr>
<td>Merid velocity ratio</td>
<td>1.1</td>
</tr>
</tbody>
</table>

### Table 4.2 LOX/LCH4 rotor`s main parameters

<table>
<thead>
<tr>
<th>LOX Pump</th>
<th>LCH4 Pump</th>
</tr>
</thead>
<tbody>
<tr>
<td>Power (kW)</td>
<td>121.8</td>
</tr>
<tr>
<td>Head coeffi</td>
<td>0.430</td>
</tr>
<tr>
<td>NPSH (m)</td>
<td>36.73</td>
</tr>
<tr>
<td>Impeller inlet D1 (mm)</td>
<td>22.1</td>
</tr>
<tr>
<td>Impeller exit D2 (mm)</td>
<td>77.2</td>
</tr>
<tr>
<td>B2 (mm)</td>
<td>4.7</td>
</tr>
</tbody>
</table>

For full cryogenic 30 kN upper stage we have selected a closed expander engine cycle. For that architecture, the turbine that controls the pump is driven by hot gaseous fuel after it has passed as a liquid the nozzle where it is used for cooling. After the gaseous fuel has passed the turbine it is injected into the thrust chamber.

Rocket turbine pumps typically use impulse turbines with higher pressure ratios and higher speeds in order to achieve a compact geometry but affecting the TPO assembly efficiency. For an expander cycle it is preferable to use one stage turbine. [7]

The turbine must provide required shaft power for driving the LOX/LCH4 pumps at a predefined rotational speed and torque. In an impulse turbine, the enthalpy of the working fluid is converted into kinetic energy within the first set of stationary turbine nozzles. The power supplied by the turbine is given by: [8]

\[ P_T = \eta_T \dot{m}_T \Delta h = \eta_T \dot{m}_T C_p T_{\text{in}} \left( 1 - \left( \frac{p_{\text{out}}}{p_{\text{in}}} \right)^{\frac{\gamma - 1}{\gamma}} \right) \]  

(4.10)
The power delivered is proportional to the turbine efficiency $\eta_T$, the fuel flow rate through the turbine nozzles $m_T$ and the available enthalpy drop $\Delta h$. On the other hand, the enthalpy is a function of the propellant specific heat, the pressure ratio across the turbine and the ratio if the specific heats of the turbine gases. From the power point of view, turbine should cover the power required by the LOX/LCH4 pumps, by the auxiliaries and power losses in the bearings and seals. Innovative blade materials (monocrystals with unidirectional solidifications) or special alloys can assure high inlet temperature in the range 1400-1600K with the additional effects of a higher enthalpy that reduce the required turbine flow. From the cost reduction point of view the actual inlet temperature is limited in the range of 900-1100K [7]. Up to now, the turbine efficiency has a limited value around 0.723 [16], being dictated by centrifugal pump design considerations which limit the shaft speed for turbopumps assembly. For a good design practice, the turbine inlet temperature should be part of the optimization process. For an expander cycle the temperature upper limit is set to 1350 K, but in order to avoid overheating, the suggested design turbine inlet temperature is 1000 K.

For turbine design it will be imposed as input parameters the following: inlet pressure and temperature, gas flow rate, rotational speed and at least outlet pressure and temperature. On the other hand has been defined the turbine power, efficiency and geometrical constraints. The working channel dimensioning is achieved by calculating the areas in the turbine sections. The turbine input energy state is defined by the calculation of the specific enthalpy according to temperature and pressure as well as the working fluid. From continuity equation [9]:

$$\dot{m} = A \cdot \sin \alpha \cdot \sigma_p^* \cdot \frac{P^*}{\sqrt{R \cdot T^*}} \cdot \frac{k \cdot M^2}{\left(1 + \frac{k - 1}{2} \cdot M^2\right)^{\frac{k+1}{k-1}}}$$  \hspace{1cm} (4.11)

results the characteristic area:

$$A = \frac{\dot{m}}{\sin \alpha \cdot \sigma_p^* \cdot \frac{P^*}{\sqrt{R \cdot T^*}} \cdot \frac{k \cdot M^2}{\left(1 + \frac{k - 1}{2} \cdot M^2\right)^{\frac{k+1}{k-1}}}}$$ \hspace{1cm} (4.12)
To determine the work channel dimensions, the radius at the base of the blade is required and the radius at the top is deduced from the previously calculated area. It is also possible to impose the radius at the top and determine in the same way the radius at the base. At this stage of design the axial is determined by the following empirical relations:

\[ \frac{h_{pf}}{x_{pf}} \in (1.3 - 1.5), \quad (4.13) \]

and

\[ \frac{h_{pm}}{x_{pm}} \in (2 - 5) \quad (4.14) \]

It was checked that the divergence angle of the channel does not exceed 15 degrees. The turbine inlet is imposed by tank pressure, being proposed to have the value 2.5 bar. The required power to support pumps consumed power should be around 280 kW. Being an upper stage, with functioning at a high altitude, the exit pressure was imposed as having 0.001 bar value. The working fluid temperature is, in this case, a variable that can be adjusted by modifying the mixture of the two fluids. The temperature will be limited to 1100 K for reasons related to the thermal and mechanical resistance of turbine materials. In table 4.3 are presented the turbine pre-design results:

### Table 4.3 Turbine pre-design results

<table>
<thead>
<tr>
<th>Inlet temperature (K)</th>
<th>Exit Mach Number</th>
<th>0.15</th>
</tr>
</thead>
<tbody>
<tr>
<td>Inlet pressure (bar)</td>
<td>Inlet Base Radius (m)</td>
<td>0.09</td>
</tr>
<tr>
<td>Rotational Speed (rpm)</td>
<td>Inlet Tip Radius (m)</td>
<td>0.107</td>
</tr>
<tr>
<td>Power (kW)</td>
<td>Exit Base Radius (m)</td>
<td>0.07</td>
</tr>
<tr>
<td>Specific Work (kJ/kg)</td>
<td>Exit Tip Radius (m)</td>
<td>0.127</td>
</tr>
<tr>
<td>Pressure Ratio</td>
<td>Mass flow rate (kg/s)</td>
<td>0.5</td>
</tr>
</tbody>
</table>

Fig. 4.2 Section through the turbine work channel

5. CONCLUSIONS

This paper provides an overview of the advantages of using methane fueled expander cycle upper stage engine, an overview of the analysis performed to parametrically study the performance characteristics, with particular focus on thrust chamber design and turbo-pumps feeding system general requirements.

A turbine pre-dimensioning calculation was performed, with the working channel defined based on the computation of the areas of the inlet and outlet sections. Under the conditions of a single turbine stage, its loading is high, which will result in low turbopump efficiency.

Single shaft, turbopump, methane fueled expander cycle engines is the optimal solution for hydrocarbon engines under investigation for future application.
In this paper, it was conducted a design procedure of a liquid oxygen liquid and of the liquid methane. Using the literature, a calculation and design methodology of the pump rotors was determined and using the Vista CPD software, the volute rotor assembly for each pump was pre-dimensioned. For the LOx pump, a 77.6mm rotor outlet diameter with a power output of 122kW was obtained and for the LCH4 pump the diameter of the rotor outlet was 120.1mm with a power output of 162kW.

System performance and configuration studies indicate that a methane-fueled expander cycle is a viable candidate for upper stage propulsion.

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