Development of Technologies for a CMC based Combustion Chamber

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Abstract

Within the MoU 'Propulsion 2010' which was signed in 2006, Astrium and DLR agreed to work jointly on Ceramic Matric Composite (CMC) materials and related technologies with the aim to have finally within the next three years all components and subsystems available for an entirely CMC – based thrust chamber assembly. Hence, propellant injection, combustion chamber liner and appropriate cooling technologies as well as CMC nozzle and nozzle extension will be developed and tested within the coming years. The paper reports the current status and focuses on propellant injection with porous face plates, heat and mass transfer within and along a porous combustion chamber liner and design and heat transfer issues of a cooled thrust nozzle.

1. Introduction

Various technology programmes in Europe are concerned with preparing for future propulsion technologies to reduce the costs and increase the life time of liquid rocket engine components. The application of modern and innovative materials and fabrication processes has become a major factor within the process of designing new components and developing new technologies in order to fulfil the future requirements and for realizing reusable and robust engine components is the application use. One of the key technologies which concern various engine manufacturers worldwide is the development of fibre-reinforced ceramics – CMC's. The advantages for the developers are obvious – the low specific weight, the high specific strength over a large temperature range, and their good damage tolerance compared to monolithic ceramics make this material class extremely interesting. Both partners of the 'Propulsion 2010' project have been active within the last years on various levels to develop materials and component design methods, system analyses, propellant injection and heat transfer issues and mechanical load handling aspects [1-10].

Within the Propulsion 2010 project a work split has been agreed upon where Astrium concentrates its efforts on the development of a regenerative cooled nozzle applying their SICTEX® (C/SiC) material and a radiation cooled nozzle extension which could be fabricated either from the standard material SICARBON® (C/SiC made by Liquid Polymer Infiltration) or the newly developed CARBOTEX® (C/C) [1, 6-8, 11, 12]. The work packages of DLR Institute of Space Propulsion include the effusion cooled C/C liner and throat technology which has been developed over the last ten years in cooperation with the DLR Institute of Structures and Design, a propellant injection system which relies on a porous face plate and simple oxidizer holes, qualification test of all components and subsystems as well as the final ceramic thrust chamber assembly test at the high pressure facility P8 [2-6, 9, 10, 13-15]. Within the project, the most critical issues, adjustment of material properties to the local cooling needs, proof of reproducibility and robustness of design methodology and production processes and leakages and, most important, development and validation of scaling laws will be dealt with.

2. Thrust Chamber Technologies

2.1 System analyses

From the very beginning intensive system analysis studies have been performed at DLR to identify potential applications of CMC's in thrust chambers and their specific requirements as well as to qualify as soon as possible critical issues of typical operational loads of liquid propellant rocket engines. While the pressure losses in the cooling channels of typical regenerative cooled liquid rocket engines may exceed 50% of the combustion chamber pressure, the driving pressure for an effusion cooled liner may be as low as the pressure drop across the propellant injections system. This advantage may be traded in the system among other options to either lower the requirements for the turbopump or increase the combustion chamber pressure. For comparisons reasons, the Vulcain 2, a gas generator cycle was analysed without taking into account within the analysis the hydrogen mass flow rate applied for film cooling. The wall temperature of an effusion cooled chamber liner as a function of blowing ratio is shown in Fig. 1. The domain where such a thrust chamber can be operated without any penalty on the specific impulse of the overall system which may result from the effusion cooling is indicated and requires materials which tolerate temperatures above 2200 K at blowing ratios which shouldn't exceed 0.6%. In a real application the I_{sp} penalty would be even lower and therefore higher coolant mass flow rates be possible since the skin friction along the liner walls will become almost negligible and thus the pressure losses inside the chamber would be considerably smaller in a chamber with effusion cooling.

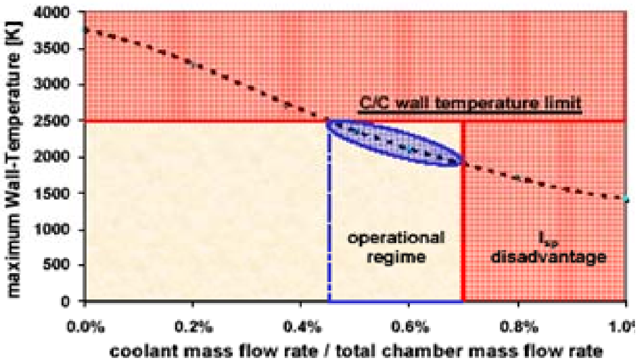

Figure 1: Operational domain of an effusion cooled TC liner

2.2 Heat and mass transfer in porous walls

In order to fully realize the potentials of CMC materials the final development is an integral thrust chamber assembly with a CMC liner with an outer load bearing carbon-fibre reinforced polymer (CFRP) and a CMC injection head. The main challenge of effusion cooling is the continuous adaptation of the local coolant supply to the varying heat fluxes. At each surface element of the liner static and dynamic pressure of both coolant and hot gas have to be balanced to prevent entrainment of hot gases into the porous C/C structure. The axial distribution of key parameters such as temperature, static and dynamic pressure and heat flux along a thrust chamber liner which have a significant influence on the performance of the effusion cooling design are demonstrated in Fig.2. The drastic changes in pressure and heat flux require a design of the porous material and the effusion cooling flow rate such that the injected coolant mass flow rate locally outbalances both, the heat flux and the total pressure. In order to develop numerical tools for a design of the cooling system and to layout, DLR has performed detailed studies of the coupled heat and mass transfer in such a thrust chamber taking into account the non-homogeneous transport properties for heat and mass of the ceramic material. These CFD analyses were performed with a standard compressible *k-e* - RANS code assuming 2D rotational symmetry. For the coupled analysis of hot gases and coolant („multi species“) was performed assuming ideal gas and the flow within the pores was modelled with a distributed resistance approach according to Forchheimer. The transport equation of the i^{th} species reads to

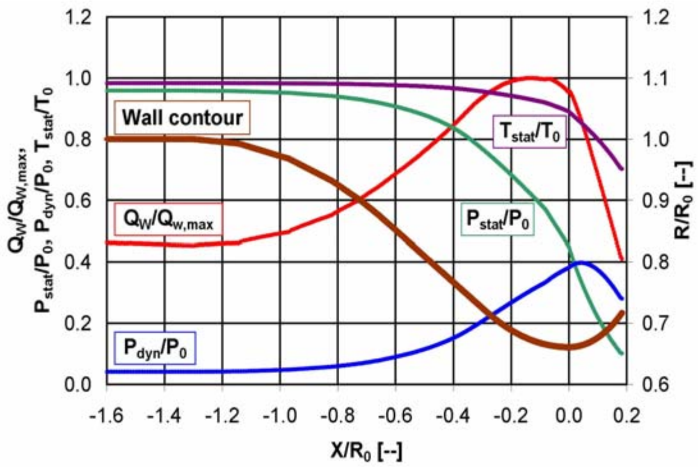

Figure 2: Axial distribution of typical thrust chamber parameterS

$$\frac{1}{r}\frac{\partial(r\rho\Phi_i v_r)}{\partial r}+\frac{\partial(\rho\Phi_i v_x)}{\partial x}-\frac{\partial}{\partial r}\left(\rho D_{m_i}\frac{\partial\Phi_i}{\partial r}\right)-\frac{\partial}{\partial x}\left(\rho D_{m_i}\frac{\partial\Phi_i}{\partial x}\right)=0 \qquad (1)$$

with Φ_i the mass fraction of the i^{th} species in relation to the total mass of species, n the number of species considered and D_{mi} the diffusion coefficient of species i. Assuming that energy sources are negligible, the energy equation can written as

$$\frac{\partial}{\partial x}(\rho v_x C_p T_0)+\frac{1}{r}\frac{\partial}{\partial r}(r\rho v_r C_p T_0)=\frac{\partial}{\partial x}\left(\lambda\frac{\partial T_0}{\partial x}\right)+\frac{1}{r}\frac{\partial}{\partial r}\left(r\lambda\frac{\partial T_0}{\partial r}\right)+W_V+E^K+\phi \qquad (2)$$

with T_0 the total temperature, W_v the volume change work, ϕ the dissipation energy and E^k the kinetic energy.

$$T_0 = T + \frac{v^2}{2C_p}\;;\; W_V = v_j \mu \left(\frac{\partial}{\partial x_i}\frac{\partial v_j}{\partial x_i} + \frac{\partial}{\partial x_k}\frac{\partial v_k}{\partial x_j} \right)\;;\; \phi = \mu \left(\frac{\partial v_i}{\partial x_k} + \frac{\partial v_k}{\partial x_i} \right)\frac{\partial v_i}{\partial x_k}$$

$$E^K = -\frac{\partial}{\partial x}\left[\frac{\lambda}{C_p}\frac{\partial}{\partial x}\left(\frac{1}{2}\left|v^2\right|\right)\right] - \frac{\partial}{\partial r}\left[\frac{\lambda}{C_p}\frac{\partial}{\partial r}\left(\frac{1}{2}\left|v^2\right|\right)\right] \tag{3}$$

With cryogenic hydrogen as coolant, temperature dependent fluid properties have to be considered and we followed the Sutherland approach for viscosity and thermal conductivity, with ς_μ the Sutherland coefficient for the viscosity.

$$\mu = \mu_n \left(\frac{T}{T_{n,\mu}} \right)^{1,5} \frac{T_{n,\mu} + \varsigma_\mu}{T + \varsigma_\mu}\;;\; \lambda = \lambda_n \left(\frac{T}{T_{n,\lambda}} \right)^{1,5} \frac{T_{n,\lambda} + \varsigma_\lambda}{T + \varsigma_\lambda} \tag{4}$$

The non-uniform behaviour of the fibre reinforced material is taken into account in form of a orientation dependent coefficients of heat conductivity, head loss and permeability in the energy (2) and Forchheimer equations (5).

$$\frac{\partial p}{\partial x} = -\left\{ K_x \rho v_x |v| + C_x \mu v_x \right\} \qquad \frac{\partial p}{\partial r} = -\left\{ K_r \rho v_r |v| + C_r \mu v_r \right\} \tag{5}$$

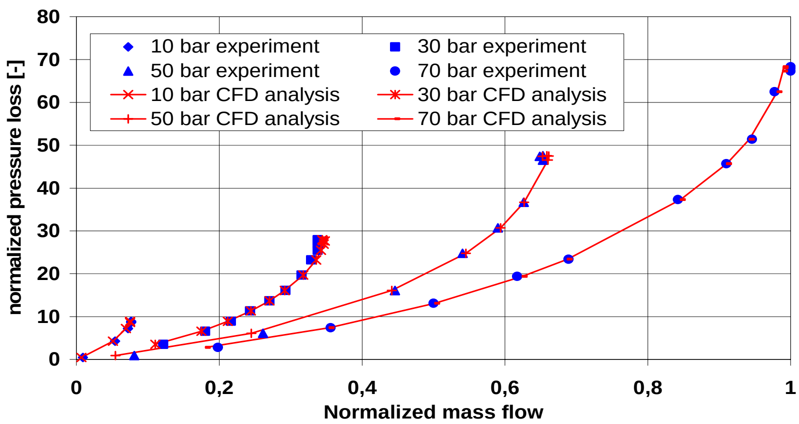

Figure 3: Comparison of experimental pressure results and CFD results with fitted Forchheimer parameters for cylindrical test sample for axial fluid flow parameters

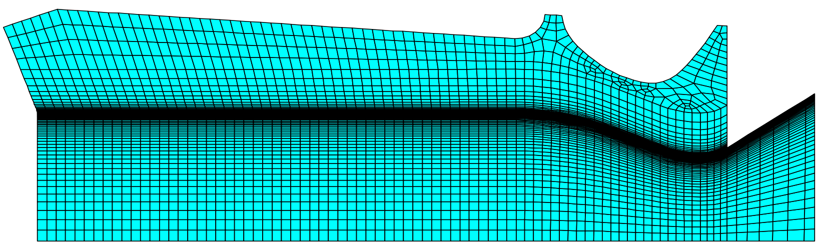

Figure 4: Meshing of the model in the area of the porous wall material

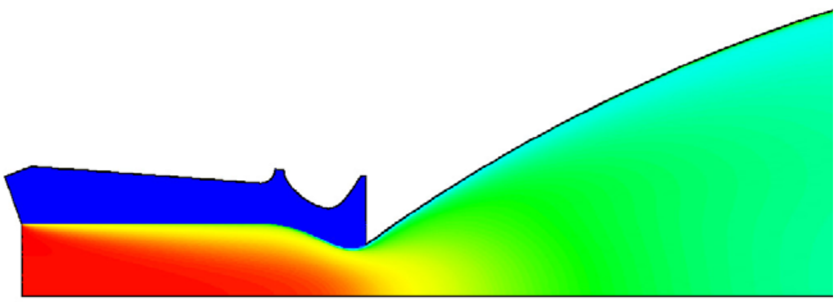

Figure 5: Temperature distribution along the liner wall

with K_x, K_r, the head loss parameters and C_x, and C_x, the permeability parameters in axial and radial directions and v_x, v_r the respective Darcy velocities. The parameters K_x, K_r, C_x and C_x, which describe the distributed resistance, are determined by means of least square fit analyses of experimental results, given for a series of pressure differences for a test sample (exemplary given for the axial flow direction).

Exemplary a comparison between the results of the experimental determination of these parameters and numerical modelling is shown in fig.3 with quite reasonable agreement between experiment and model. In radial directions the agreement is similar well however the actual losses are larger. This non-homogeneity of material parameters depends on different parameters, i.e. initial fibre diameter, processing steps or a possible 3D reinforcement.

A zoom of the interesting part, the porous liner and the vicinity along the porous surface inside the combustion chamber is given in fig. 4 and a sample result of a temperature distribution in the thrust chamber is shown in fig. 5. For simplicity reasons and to allow for a concentration of all computational effort onto the areas of interest, atomization, mixing and combustion aspects have been neglected the temperature in the combustion chamber has been assumed at a temperature of 3700 K instead. The impact of the injected coolant along the porous liner is clearly visible. More recently, DLR started 3D simulation studies which include atomization and combustion as well to allow for more detailed studies of the interaction between injector flow and effusion boundary layer in the vicinity of the injector head.

2.3 Propellant injection and atomization

Figure 6: Porous penta-injector mounted in model combustor “C”

An essential function of the injector head assembly is to uniformly inject the propellants into the combustion chamber at the proper oxidizer/fuel mixture ratio, dampen secondary flows resulting from the turbopump or piping systems and at least partially decouple dynamically the propellant feed system from the combustion chamber. A major concern of any injector is the injector/wall interaction. In the vicinity of the face plate where propellant mixing is poor oxidizer-rich gases mixed with cryogenic droplets may get in contact with the combustion chamber walls. The result of this process, a combined physical and chemical attack, yields visible traces of material destruction at the liner surface called ‘blanching’. An entirely different approach compared to the high precision requirement coax or tri-coax injectors is an approach with a porous face plate through which all the gaseous propellants are fed into the combustion chamber and combined with a large number of small LOX injectors. The advantage of this approach is that the dimensions of each element are so small that deviations from the required dimensions are tolerable since each element delivers something in the order of 0.1% of the total oxygen mass flow rate to the chamber and deviations affect only negligible parts of the combustion chamber volume. It is worth mentioning that the atomization process with this kind of injector is different compared to the one with conventional shear coax injectors. Distribution of the gaseous fuel all over the face plate considerably lowers the hydrogen velocity which in turn reduces the available aerodynamic forces available for atomization which makes this approach less attractive.

In order to study this type of injection system in more detail, DLR followed an approach which aimed at first, identifying the near injector region of such a system using its model combustor ‘C’ which allows for optical access even at pressures which exceed 6 MPa. Test campaigns have been performed with single injector and multi injector assemblies, see fig.6 for the multi (penta) injector arrangement mounted in the combustor. Compared to a typical shear coax injector, the combustion with a porous injector setup seems to take place much nearer to the face plate as shown in fig. 7 which displays averaged OH-emission pictures of a shear coax and a porous injector for near critical pressure conditions. Although atomization and combustion are shifted towards the injector, the cooling capability of the gaseous propellant seemed sufficient since no signs of intolerable high temperatures could be observed. Obviously, this change in heat release will have an impact on the heat load distribution along the liner compared to conventional shear coax injectors. Hence, in a second campaign DLR applied its model combustor ‘B’, a segmented multi injector assembly with an inner diameter of 50 mm which is designed for combustion chamber pressures of 10 MPa, to study the head load distribution with a porous injection system. In order to verify the injector concept and to save time this study was performed with an injection system fabricated using a

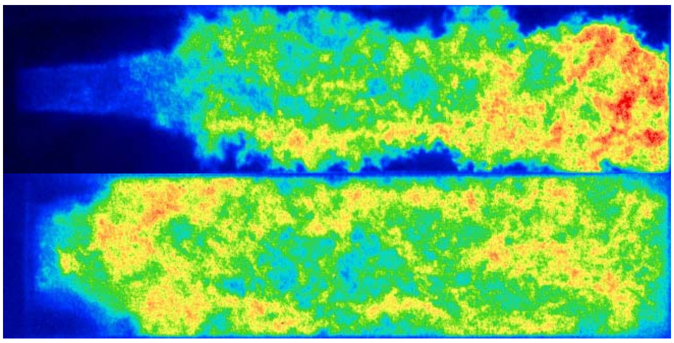

Figure 7: Comparison of the near face plate ragion of a shear coax (top) and porous (bottom) injector

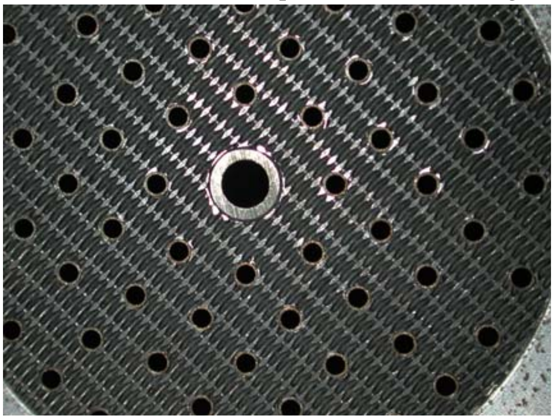

Figure 8: Sub-scale injector head for combustor ‘B’ with Rigimesh© face plate

Rigimesh© face plate, see fig. 8. The results confirmed the shift of the heat release towards the face plate observed with combustor 'C'. The differences in the local heat fluxes to the liner wall at a given downstream position may exceed 50%.

4. Nozzle Technologies

Over the past years, Astrium has, together with various partners, worked intensively on developing components for air-breathing and liquid rocket engines. Since this, various prototype developments and hot firing-tests with nozzle extensions designed for upper and core stage engines and combustion chambers of satellite engines were conducted. MBDA France and Astrium have been working on the development of fuel-cooled composite structures like combustion chambers and nozzle extensions for future propulsion applications. In view of the extreme thermo-mechanical loads in the combustion chamber of liquid-propellant rocket engines, previous developments at Astrium were focussed above all on the use of ceramic fibre composites for the less thermally loaded nozzle extensions. At Astrium, nozzle extensions made of SICARBON® have been developed for upper-stage (scale 1:1) and main-stage (scale 1:5) engines and successfully tested under real space conditions. Further test campaigns with radiation-cooled combustion chambers were carried out in the field of satellite propulsion, whereby the SICARBON® material was able to demonstrate its long-term stability and high chemical compatibility versus the propellants and combustion products in hot-firing tests carried out at sea level conditions [8], see fig. 9.

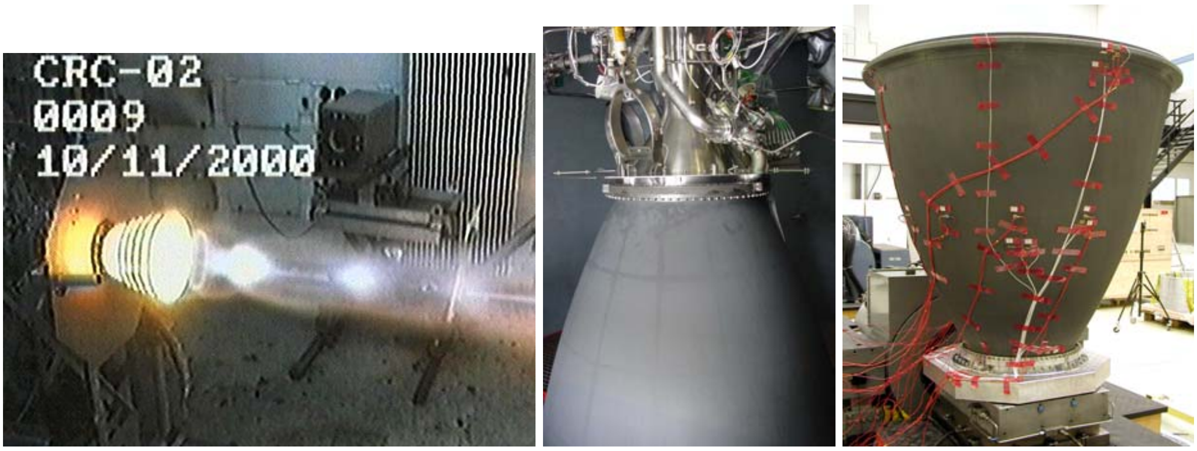

Figure 9: Hot test of nozzle extension (left picture), full-scale upper-stage nozzle extension on test facility (middle picture) and during vibration test (right picture)

The development of ceramic nozzle extensions for rocket engines represents an increasingly recognizable problem: In the current design, the ceramic nozzles are carried out in single-shell design and as passively cooled structure. The exclusively used cooling concept of currently existing ceramic nozzles of upper stage engines (VINCI, RL10-B2) is, based on the today's knowledge level, however insufficient to guaranty the thermo-mechanical integrity of future high performance rocket engine structures. Due to the elevated thermal loads experienced by a main stage engine nozzle a design without an active cooling is not conceivable.

Based on the experiences with the development of fuel-cooled ceramic combustion chambers for air breathing engines for Dual-Mode Ramjets (DMR´s) [7, 8, 10, 11], preliminary investigations have been performed to outline the possibilities and interest of using the PTAH-SOCAR (Paroi Tissée Application Hypersonique- Simple Operational Composite for Advanced Ramjet) technology for actively cooled nozzles of Liquid Rocket Engines. System studies undertaken at Astrium as well as structural and thermal analyses promise, thanks to the use of CMC's in thrust chambers of liquid-propellant rocket engines, substantial advantages compared to metal materials, which are currently utilized for most launcher propulsion systems for cooled combustion-chamber structures and nozzle extensions. The main advantages comprise on the one hand the possible weight reduction and on the other hand the high resistance to thermo shocks as well as the resistance to chemical attack versus the liquid propellants used. A further significant advantage is the high creep resistance and the extraordinary resistance to high temperatures compared to metal materials.

However, the multi-axial states of stress occurring specially in actively cooled thrust chambers and nozzles necessitate a fibre composite that features sufficient shear strength in as many directions as possible (e.g. isotropic behaviour). The currently available 2-directional fibre composites would probably only have very limited lifetimes respectively damage tolerance. For this reason, some years ago the development of a new material system and manufacturing process, respectively, was commenced, with the objective of combining multidirectional (3D) textile structures with a cost-effective infiltration method. A cross section of the SICTEX® (C/C-SiC surface infiltrated) Fuel-Cooled Combustion Chamber (FCCC) is shown in figure 10.

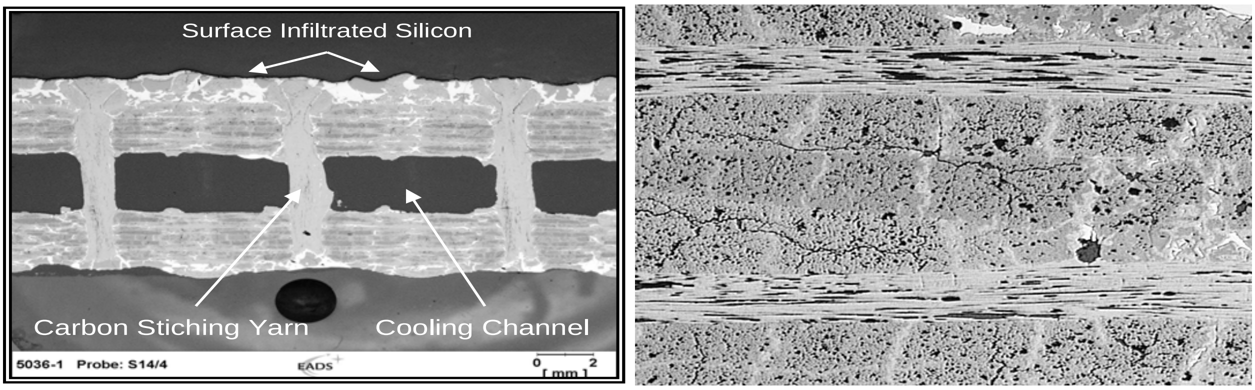

Figure: 10: Sandwich structure with cooling channels (left) and cross-section of SICTEX® SI material (right)

The potential interests of using CMC's for rocket nozzles are an increased component lifetime since thermal cyclic sensitivity is negligible compared to current metallic structures, the light weight design capability, increased cooling concept flexibility (regenerative, dump, film, effusion) and a performance gain due to less weight and reduced cooling requirements. Astrium's part within the 'Propulsion 2010' project bases on the PTAH-SOCAR design and composite material technology. Figure 11 shows a sketch of the fuel cooled CMC subscale nozzle design. The geometry and contour of the nozzle bases on a VULCAIN nozzle down-scale and is modified into a dual bell shape. While the fuel cooled section extends to an area ratio of $\varepsilon = 32$, an exchangeable ceramic skirt up to an area ratio of $\varepsilon = 60$ secures the performance target. The nozzle layout was driven by envisaged LOX/H_2 hot firing tests operated at the P8 test facility in Lampoldshausen with a chamber pressure of $p_c = 100$ bar and a mixture ratio of O/F = 6. The interface to the combustion chamber at an area ratio of $\varepsilon = 5$ has been designed to be compatible to an existing Astrium 4 kN subscale thruster. This gives the opportunity to test the nozzle device even prior to the ceramic motor test campaign as a passenger. To approach as best as possible the conditions of potential full scale applications hydrogen has been selected as coolant. Moreover, the specified hydrogen inlet pressure of 8 MPa bar is fully representative for a dump cooled nozzle system as currently used in the VULCAIN 2 engine.

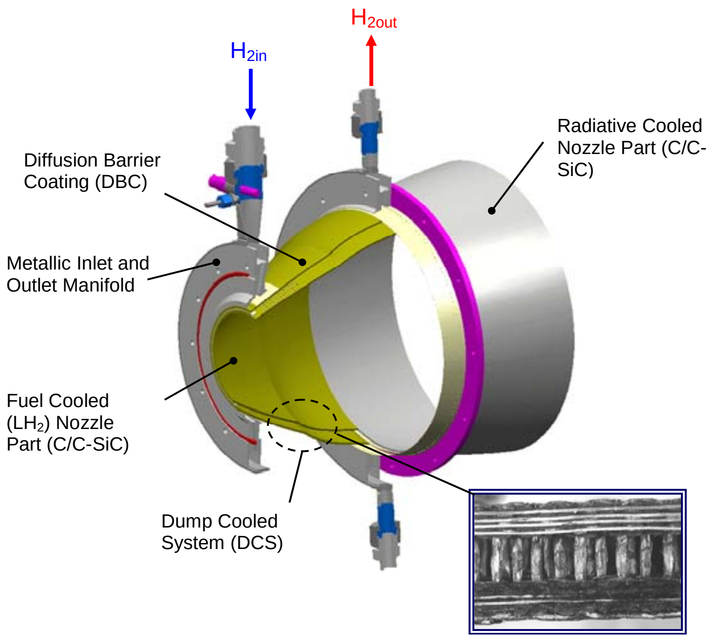

Figure 11: Fuel cooled CMC dual bell nozzle

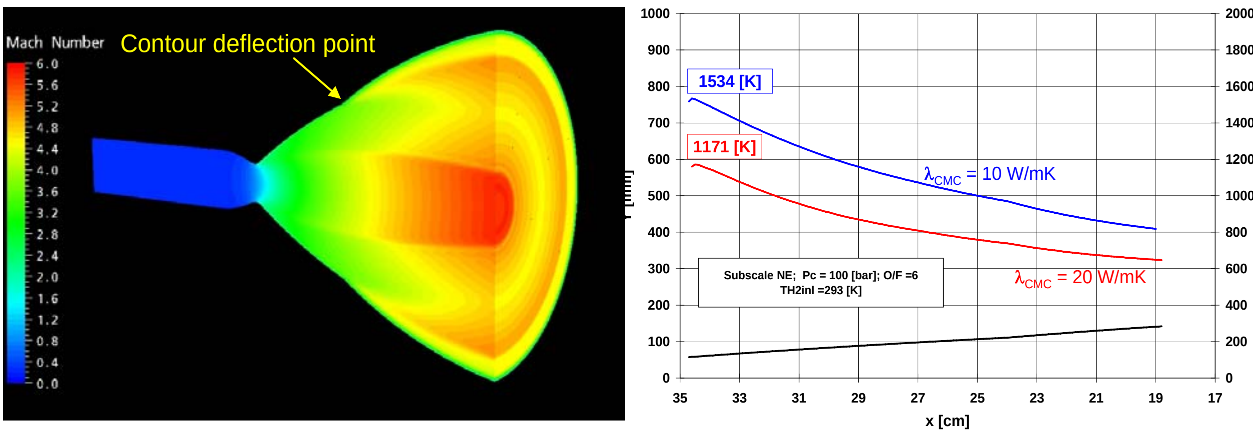

Figure 12: The thermal layout of the subscale FCCN has been conducted with Astrium's advanced heat transfer tools [15]

From thermal layout point of view two challenges had to be mastered. Firstly, the heat flux distribution along a hot wall in a supersonic flow had to be determined very precisely and secondly, the coolant efficiency in a pin fin structure with rough ceramic surfaces had to be evaluated. For the latter an advanced Nusselt-type correlation including surface roughness effects has been applied. Additional pressure drop and heat transfer enhancement induced by the pin fins have been estimated. The heat flux profile imposed by the hot gas flow has been assessed using CFD with detailed chemistry modelling. Before, the methodology had been anchored to the aforementioned SICARBON® nozzle tests. In particular the predicted gas composition close to the hot ceramic wall was found to crucially determine the transport properties and hence the heat fluxes. Figure XX displays the simulated Mach number distribution within the dual bell nozzle and two typical wall temperature profiles depending on the realized radial thermal conductivity of the CMC material. Apparently, the assumed coolant flow rate can further be reduced to reach the temperature limit of the ceramics.

In a first approach the design work and the prototype manufacturing of a Fuel-Cooled-Ceramic-Nozzle Extension has taken place. The investigation of heat transfer and pressure drop of the sandwich structure has shown that the effect on wall temperatures because of a reduction of the channel height is not significant but in opposite there is a severe influence on pressure loss. Further on an essential reduction of the coolant mass flow rate is possible and there is no critical heat conductivity influence of pin fins on the wall temperatures to be expected. A first nozzle prototype has been produced by textile technique and fig. 12 shows the braided and densified prototype.

Figure 13: Braided and densified nozzle extension prototype

4. Conclusions

The development of an actively cooled nozzle structure a special challenge represents to CMC structure regarding thermal conductivity, damage tolerance, tightness and joining technique. Besides the development of the joining of the metallic manifolds with the ceramic structure, the setting of a defined porosity on hot gas wall side to the influence of the permeability is solved and in certain limits reproducible. Using liquid hydrogen as coolant, absolute tightness has to be achieved to fulfil the system requirement and special coating systems are under development to seal the outer shell of the nozzle.

Besides the material related problems mentioned above, the major technical challenges for the development of the propellant injection system and the combustion chamber liner are the compatibility of the spray pattern with the requirements of the liner in the vicinity of the injector, the cryogenic temperatures of both propellants for gas generator applications and the local adaptation of the coolant requirements to the varying thermal loads. The technical challenges must be solved and verified in the coming years to test the ceramic nozzle and the ceramic combustion chamber on the test bench in 2010.

5. Summary and Outlook

Within the MoU 'Propulsion 2010' Astrium and DLR have agreed to cooperate in the field of development of CMC components for the application of liquid rocket engines by in combining their competences and resources with the aim to develop a subscale thruster based on CMC components. Although key technologies have already been developed and verified during the last years, there are still technical challenges which remain to be mastered. However, in the upcoming years Astrium and DLR will continue their cooperation and work closely together to reach the goal and have a thrust chamber assembly made of CMC materials ready for a test campaign at the European R&D facility P8 at DLR Lampoldshausen.

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