Potential Early Applications of Advanced Effusion Cooled Ceramic Combustion Chambers

Martin Sippel, Ingrid Dietlein
Space Launcher Systems Analysis (SART), DLR, 28359 Bremen, Germany

The paper addresses the question when and in which application an innovative ceramic combustion chamber can first be used in a launcher propulsion system. The introduction of this new technology in any existing European launch vehicle without prior successful space flight application is unlikely. A first potential future use of the effusion cooled engine is proposed in a complete new vehicle of smaller size.

The paper compares the different engine design configurations and delivers performance, size and mass data based on preliminary calculations. The influence of a potentially reduced mixture ratio and an impact of the effusion cooling on combustion efficiency are considered. All presented stages are able to deliver payloads between 50 kg and 250 kg into a polar orbit consistent with a small micro-launcher performance. The stage architectures, masses and overall dimensions are outlined in the paper.

Subscripts, Abbreviations

c*	characteristic velocity	m / s
I_{sp}	(mass) specific Impulse	s (N s / kg)
M	Mach-number	-
T	Thrust	N
m	mass	kg
α	angle of attack	-
ε	expansion ratio	-

C/C	Carbon Carbon
CFRP	Carbon Fiber
CMC	Ceramic Matrix Composites
GLOW	Gross Lift-Off Mass
LH2	Liquid Hydrogen
LOX	Liquid Oxygen
MR	Mixture Ratio
SiC	Silicon Carbide
SSO	Sun Synchronous Orbit
TRL	Technology Readiness Level
VEGA	Vettore Europeo di Generazione Avanzata
C	chamber
vac	vacuum

1 Introduction

Technologies for porous effusion or transpiration cooled ceramic combustion chambers are under investigation at DLR since a couple of years. Research concentrates on applying transpiration cooling to C/C liners as a candidate technology to fulfill the requirements of future cryogenic (LOX/LH2) liquid rocket engines, taking advantage of interesting features of C/C liners such as low weight as well as potentially increased reliability and life time of the rocket engine. A fully ceramic combustion chamber including nozzle is currently foreseen to be testfired in early 2011 (see Figure 1!).

The question when and in which application a ceramic combustion chamber can first be used in a launcher propulsion system becomes now more imminent. The introduction of this new technology in any existing relatively large European launch vehicle like Vega or Ariane without prior successful space flight application is unlikely. A first potential future use of the effusion cooled engine is more likely in a completely new vehicle of smaller size.

DLR has investigated a few design options of upper stages with low thrust (up to 10 kN) ceramic engine with hydrogen and oxygen propellants. Some of these types investigated in the multi-national study Aldébaran [1] or in a technology assessment together with Astrium for the German space agency [2] are described in this paper.

2 Status of ceramic thrust chamber development

A new approach proposed by the German Aerospace Center (DLR) since about one decade is the realization of a combustion chamber concept based on transpiration cooling and use of porous ceramic matrix composites (CMC) and high performance carbon fibre reinforced plastic composites (CFRP). [3]

Compared with metals, CMC materials show properties like low specific mass, high temperature stability, damage tolerance etc. which make them highly attractive for use in a thrust chamber environment. When considering this class of materials for thrust chamber applications, it is mandatory to apply structural design concepts which take the unique material characteristics of CMC's into account, such as anisotropy, very specific production processes combined with net shape techniques in design and manufacturing. [3]

Regenerative cooling requires a relative high value of thermal conductivity; the reason why copper is one preferred material choice for chamber walls. Typical materials based on either carbon or alumina fibres show thermal conductivity coefficients which are between one and two orders of magnitude lower than the ones of copper alloys. Thus, regenerative cooling is not an attractive cooling method for CMCs.

As an alternative method, transpiration or effusion cooling is well known since 50 years. Along with the US Space Shuttle development an entire transpiration cooled engine, the XLR-129, had been developed and successfully qualified showing at least the cooling potential of this method [4]. However, if a porous and permeable metallic wall material based on platelet technology created by sintering processes is used, a critical situation may occur during operation when local overheating melts the porous surface resulting in rapid further damage due to the interrupted local cooling. Today, CMCs with either no melting phase or very high operational temperature limits offer the potential to establish transpiration cooled concepts which avoid such disadvantages and associated operational risks inherent to metals. Furthermore, the significant pressure loss occurring in the regenerative cooling channels can be avoided or reduced when using a transpiration cooled system.

Figure 1: Entire thrust chamber assembly in CMC as foreseen to be testfired

Ceramic matrix composites are composed of ceramic fibers and a ceramic matrix. DLR development in the past focused mainly on two CMC routes. The first one is C/C-SiC which was initially mainly driven by thermal protection requirements of re-entry vehicles [3]. Carbon fibers are embedded in a SiC matrix. Another CMC is called WHIPOX and is based on oxides, i.e. alumina or mullite fibers (Nextel) and usually an alumina or mullite matrix [3].

Using CMCs in a combustion environment raises questions with respect to thermo-chemical stability and resistance against a variety of species present at the material's surface. High temperatures, pressure and flow velocities create a severe environment, where chemical reactions, oxidation, erosion, melting etc. can occur, depending on fuel/oxidizer and reaction products. Thus, the verification of the material applicability under relevant conditions of combustion environment is very important. [3]

Until now only porous carbon-carbon (C/C) has been used for DLR's development of transpiration cooled cryogenic CMC thrust chamber which is derived from DLR's C/C-SiC LSI process. Figure 2 shows a typical microstructure of C/C which is obtained during the pyrolysis of the 'green body' virgin CFRP material.

As the C-fibers act as a kind of shrinkage impediment, tension stresses in the matrix caused by the pyrolysis process lead to an internal crack and channel system with a typical pattern. Normally, the cracks serve as channels for the later on liquid silicon infiltration and conversion of Si and C to SiC, leading to C/C-SiC with low porosity (approximately 3%). Depending on the thermal treatment, this porous intermediate C/C shows an open porosity between 12% and 20%; it can be reinfiltrated with resin with a follow on pyrolysis, reducing the open porosity down to approximately 7 - 8%. For this reason, the permeability will be changed offering the possibility to adjust it to application driven requirements. [3]

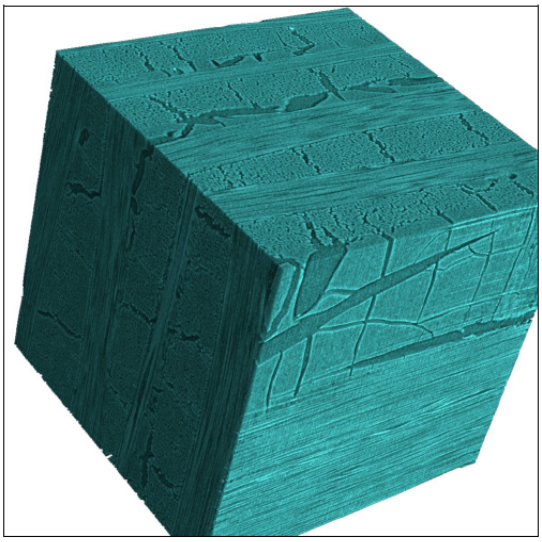

Figure 2: Microstructure of C/C material

2.1 Thrust chamber hotfire testing

Five CMC segments have already been hotfired as an axially stacked configuration with 50 mm inner diameter. Figure 3 shows a hot run of the integrated combustion chamber with an entirely transpiration cooled inner C/C liner structure and the advanced porous injector (API) from the DLR Institute of Space Propulsion. Several test campaigns including cold flow, ignition and hot tests at the test bench P8 in Lampoldshausen were performed in late 2008.

The tests demonstrated well the structural integrity of the design concept and the effectiveness of the cooling system. However, relatively high coolant mass flow rates had to be used to protect the C/C wall material especially in the vicinity of the injector. At this station traces of damage by oxidation, erosion or other effects occurred and are part of ongoing investigations. [3]

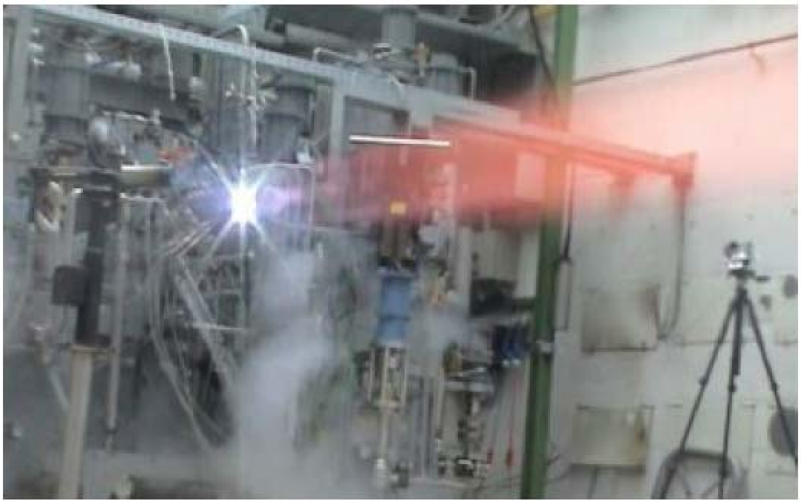

Figure 3: Hot run #2 at P8 (December 2008) with approximately 9 MPa chamber pressure

A fully ceramic thrust chamber including nozzle is currently foreseen to be testfired in early 2011 (see Figure 1!) which should not only include CMCs for the cylindrical chamber but also an effusion cooled throat from DLR and an uncooled ceramic nozzle section to be provided by Astrium.

3 Potential Early Applications of Effusion Cooled Ceramic Combustion Chamber

The introduction of the new ceramic combustion chamber technology in any existing relatively large European launch vehicle like Vega or Ariane without prior successful space flight application is unlikely. A first potential future use of the effusion cooled engine is more probable in a completely new vehicle of smaller size for which different technical options have been investigated in the Aldébaran project. One of the intentions of this multinational European approach is the flight test of advanced technologies in experimental stages [1]. DLR has further investigated a few design options of operational upper stages with low thrust ($\leq$ 10 kN) ceramic engine using liquid hydrogen and oxygen propellants [2, 5].

The objective of all proposed early applications of ceramic thrust chambers is not to demonstrate that the innovative technology would allow a unique mission which could not be achieved by conventional regeneratively cooled chambers or existing storable engines. Such a requirement would probably be too ambitious and by far too costly and risky. Therefore, the early applications are assessed on their capability of safely achieving their mission goals of delivering a considerable payload to a destination orbit. After successful demonstration of CMC thrust chambers in operation and showing superior characteristics, more demanding missions could be developed.

3.1 Engine and stage requirements

A few high-level requirements have been set before defining the early applications. As baseline only upper stages are looked at for which performance and inert mass are at acceptable values that delivery of a sufficient payload to LEO is achievable. The usual requirements on safety and reliability of operations are to be fulfilled. The propellants are to be delivered to the engine in acceptable conditions which will raise some challenges and will drive design solutions as discussed below.

The thrust level is between 1 kN and 10 kN as a maximum and mixture ratio of the cryogenic stages should be around 5. However, depending on the technological maturity of the effusion cooling process with CMC chambers even lower mixture ratios are to be considered. The possible nozzle expansion ratios are to be compatible with the dimensions of the small stage. Thus, exit diameter should be around 1 m or less. Burn times are quite short between 10 s and 30 s for kick-stages and a couple of minutes for the more ambitious pump-fed upper stages. Operational times including boost phases, ballistic phases, and de-orbiting could extent up to a few hours [5].

3.2 Application in an experimental kick-stage

An experimental kick-stage is the simplest option at lowest cost for an early application of CMC thrust chambers. Only one single ignition will be required with limited burn time. De-orbitation or transfer into a graveyard orbit could be achieved by cold gas thrusters using the residual gaseous propellant. The engine will be pressure-fed because the development of a small dedicated turbopump for an early unproven application might be too expensive in the beginning. A very simple engine design with gaseous injection at supercritical temperatures is an alternative requiring high pressure storage of oxygen and hydrogen.

During the Aldébaran study [1] the task of the kick-stage has been defined to circularize a 150 kg payload plus adapter from a 40 km x 800 km polar orbit into a circular 800 km SSO (800 km x 800 km x 98°). The main propulsion subsystem is designed to operate on gaseous propellant (GOX/GH2) only. Propellant is also stored in its gaseous (supercritical) state and remains gaseous throughout the complete propellant subsystem. The ΔV delivered by the main propulsion subsystem must be at least about 210 m/s to ensure circularization of the transfer orbit [6].

Main engine requirements are a thrust of approximately 1800 N, burn time of 28.1 seconds, and cooling of the thrust chamber is achieved by diffusion of GH2 through its porous walls. The proposed thrust chamber contour with a nozzle of an expansion ratio of 150 reaches an overall length of 0.93m. Table 1 presents the obtained performance mean values.

The propellant mass used to accelerate the stage from the transfer orbit into an 800 km x 800 km circular orbit amounts to 11.8 kg. This leads to a nominal burn time of 28.1 s. The total tank volume for oxygen is 0.1588 m³ and for hydrogen 0.1942 m³ [6].

Mixture ratio [-]	5.0
Chamber pressure [bar]	10.8
Chamber temperature [K]	3113.5
Specific vacuum Impulse [s]	431
Vacuum thrust [N]	1775.2

Table 1: Table 1Engine performance characteristics of pressure fed experimental kick-stage engine [6]

In a pressure fed system with propellants stored in gaseous state not only pressure but also temperature is significantly decreasing during operation. As can be seen in Figure 4, the temperature at tank depletion of hydrogen is still above 80 K. This is compatible with common injection conditions aiming at preventing combustion instabilities. Considering that with adequate injector design, combustion stability may be achieved with hydrogen temperatures below 80 K, the current tank definition offers comfortable margin with respect to the danger of combustion instability.

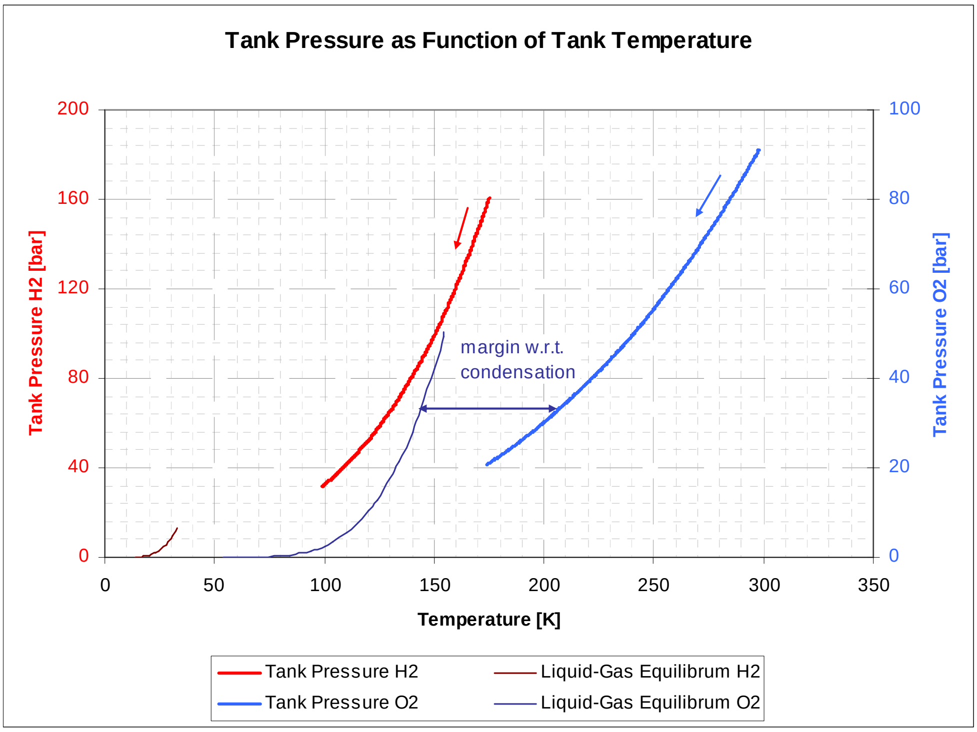

Figure 4: Tank pressure as a function of temperature for O2 and H2 in pressure fed kick-stage [6]

As shown by Figure 4, first depletion occurs on the oxygen side because the pressure in the tanks is no longer sufficient to sustain a positive pressure difference with the combustion chamber under the requirement of a minimum pressure difference of 10 bar between the tanks and the chamber. The figure furthermore demonstrates that no condensation occurs and that this aspect is potentially more critical for oxygen.

It can be concluded that the kickstage can be operated with sufficient margin with respect to condensation and minimum hydrogen temperature at injection for 32.8 s which offers a very comfortable margin for performance scattering and calculation uncertainties with respect to the required nominal combustion time of 28.1s.

The kick-stage design has been found in a multidisciplinary iterative process and the final configuration is presented in Figure 5. It consists of a stage adapter ring for attachment on the 2nd stage and two main beams combined to a cross structure. Four cylindrical tanks are attached to the ring and the beams at three points. They are made of an aluminium liner and composite over wrappings for load bearing. The stage dry mass is relatively high due to the need of high pressure hydrogen storage up to 180 bars. Therefore, the separated payload into an SSO is restricted to approximately 100 kg [6].

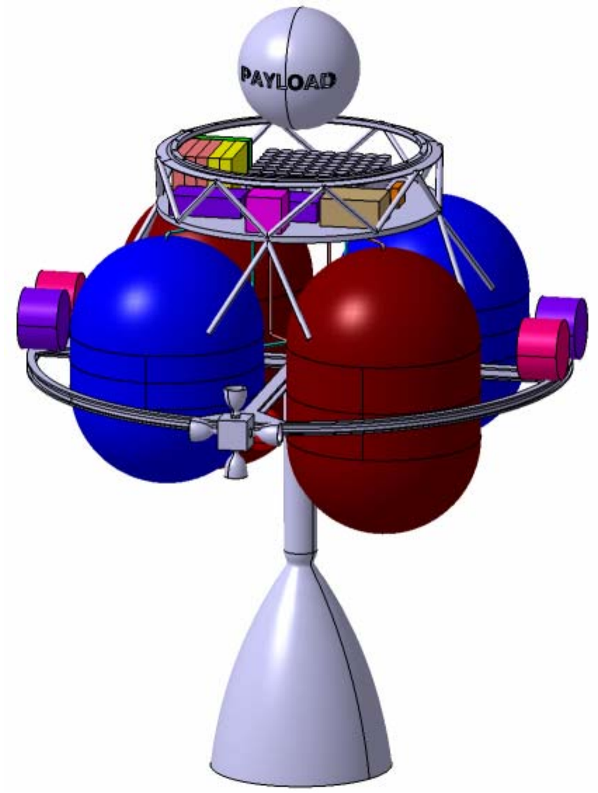

Figure 5: Final design of kick-stage at the end of phase 0 study [6]

One critical point raised is the filling strategy of the kickstage. Concerning the location where the filling of the tanks is executed, two major options exist with each of both have their drawbacks and advantages: Filling in the integration building with the tanks already under high pressure. Then a certain risk is expected due to the presence of gaseous hydrogen within a confined space. An alternative option is filling on the launch pad. Then special installations on the launch pad have to be provided. Furthermore, an interface in the fairing from the launch pad installations to the tanks is to be created. This is linked to additional infrastructure and operational costs.

As the presented gaseous kickstage is only for technology demonstrating purposes it may be preferable to accept the potentially significant dry mass increase of filling option 1 to the benefit of reduced launch pad and fairing complexity. Nevertheless, this approach is only feasible for a kick-stage with small Δ V.

3.3 Application in small cryogenic upper stages

The cryogenic upper stage mission requirements are more demanding as this stage should be able of injecting into transfer orbit and after a certain ballistic flight time reignite and perform the circularisation manoeuvre and afterwards the de-orbiting. The increased Δ V compared with the kick stage makes liquid storage of propellant under cryogenic conditions indispensable due to the otherwise outsize dimensions.

This fact raises a challenge of every cryogenic engine without regenerative cooling: the hydrogen injection at a suitable temperature necessary for achieving stable combustion. Pre-heating by a separate gas generator or the hybrid design of a small regenerative circuit combined with the effusion cooled section could be feasible options. However, an otherwise simple pressure fed engine becomes almost as complicated as a gas generator cycle engine without turbo machinery.

The hybrid design option in case of the pressure fed engine is disregarded here because a partially regenerative thrust chamber is not well suited to demonstrate effusion cooled chamber technology. Further, the integration of different materials within one chamber could become challenging. Thus, a separate gas generator similar in size and operation conditions to such a device used in open cycle turbo pump engines is to be included in order to sufficiently heat up the liquid hydrogen beyond critical conditions and also the tank pressurization gases (Figure 6 at left). The actual H_2 temperature required at the chamber head is subject to debate. Depending on the type of injectors used this is between 70 K or 80 K (coaxial) and might be reduced to 55 K for some advanced designs. In any case the temperature is far above the liquid hydrogen storage conditions which will not exceed 30 K even under relatively high pressures. The liquid propellant fed gas generator might be replaced by a small solid propellant gas generator in order to simplify the engine. However, the Isp engine performance of such a design is slightly reduced and reignitability becomes more difficult.

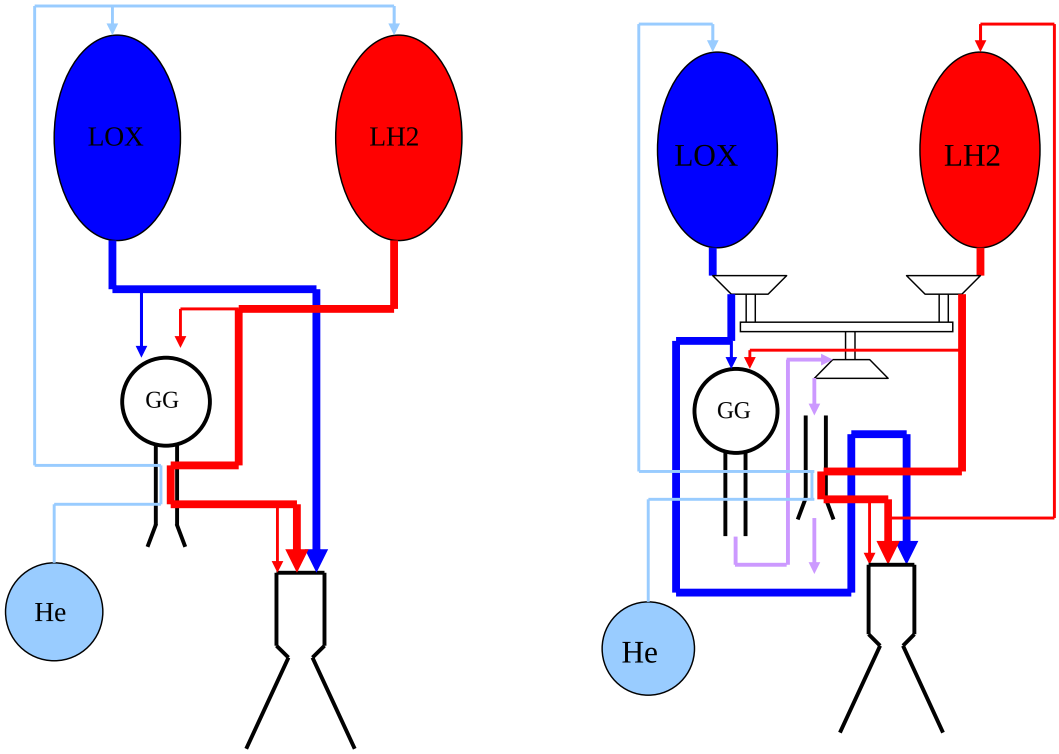

Figure 6: Cycle options of small cryogenic upper stage with fully ceramic thrust chamber (left: pressure fed, right: turbopump fed) [5]

The gas generator turbopump cycle with ceramic combustion chamber looks much more classical (Figure 6 at right). The only difference is the fact that turbine exhaust gas is not only used for heating the pressurisation gas but also for raising the temperature of the hydrogen to be injected into the main combustion chamber. The obvious challenge of this concept is the task to develop an efficient turbine and pumps of very small size only for the dedicated application. Unfortunately such cryogenic turbomachinery is not available off the shelf, at least in Europe.

The mixture ratios of these small engines have been defined at relatively low, fuel rich values based on results from the first P8 test campaign (see section 2.1 and [3]!) This experiment indicates that with the current design and material selection a high coolant mass flow will be required to avoid damage by oxidation in some regions close to the injector head. This assumption is a conservative approach in order to remain on the safe side. Several other technical options exist (e.g. change of injector placement or type, other wall materials) to improve the unfortunate situation of high coolant flow. However, effectiveness of these options is all not yet proven by experimental tests.

The chosen chamber MR values between 2.55 and 3.9 are based on an extrapolation done by Astrium from the experimental data at 9 MPa without damage and the heatflux to be expected at different chamber pressures and throat sizes [7]. The thrust range between 2 kN and 10 kN with chamber pressures of 1 MPa and 2 MPa has been investigated. The acceptable engine mixture ratio is depending on the expected heatflux on the wall: the lowest for a small engine with high chamber pressure and the maximum of RM= 3.9 for the largest engine with a pressure of 1 MPa. The nozzle expansion ratio is fixed for all engines at 150.

	MR_c [-]	P_c [bar]	F_{vac} [kN]	$I_{sp,vac}$ [s]
SK-2H-10 PF	2.75	10	2	424.8
SK-2H-20 PF	2.55	20	2	421.7
SK-5H-10 PF	3.4	10	5	431.1
SK-5H-20 PF	3.3	20	5	431.4
SK-10H-10 PF/TP	3.9	10	10	432.0 / 432.1
SK-10H-20 PF	3.75	20	10	433.5

Table 2: Estimated operational characteristics of different engines with effusion cooled thrust chambers [5]

The low mixture ratios have a relatively small and benign effect on the achievable Isp due to the shallow slope of the fuel-rich side of LOX-LH2 combustion looked from the optimum MR. The specific impulse values have been calculated taking into account a reduced c* due to effusion cooling and the additional gas generator required for fuel pre-heating. Basic data are from the DLR codes ncc and lrp and actual performance might slightly vary depending on the assumed gas property data.

The engine sizes and masses differ quite large due to the different thrust levels and chamber pressures [5]. The largest variant is the SK-10-H10 which slightly exceeds the requirement of an exit diameter of less than 1 m and which reaches a total length of about 1.7 m. The other engines with increased chamber pressure and lower thrust are significantly smaller [5]. The estimated masses of the 5 kN and 10 kN engines are between 20 kg and 25 kg.

A preliminary performance assessment of some low mixture ratio CMC chamber engines on a generic micro-launcher has been performed by DLR-SART within the BEAST study [2, 5]. A four stage vehicle has been defined with the lower three stages using existing solid motors: Z23 of Vega + the Brazilian S43 and S44 as 2nd and 3rd stage. The cryogenic upper stage is filled with 1000 kg of usable LOX and LH2 propellants. Staging parameters and propellant loading of this configuration are not optimized because this would have been beyond the scope of BEAST. However, the obtained results of stage mass estimation and ascent trajectory optimization show that such a vehicle at least is feasible in principle despite the relatively large hydrogen tanks required due to the low MR.

The SK-5H-10 PF and the SK-10H-10 PF/TP engines have been selected for this launcher performance assessment because both deliver slightly more than 430 s of vacuum I_{sp}. The separated payload masses into an 800 km 98° SSO achievable in simulation of the 4-stage vehicle are shown in Figure 7. The major difference is between turbopump fed and pressure fed engine and stage designs. While the former might achieve more than 400 kg payload, the latter are limited in the range 100 kg to 250 kg depending on the structural technology of the tanks.

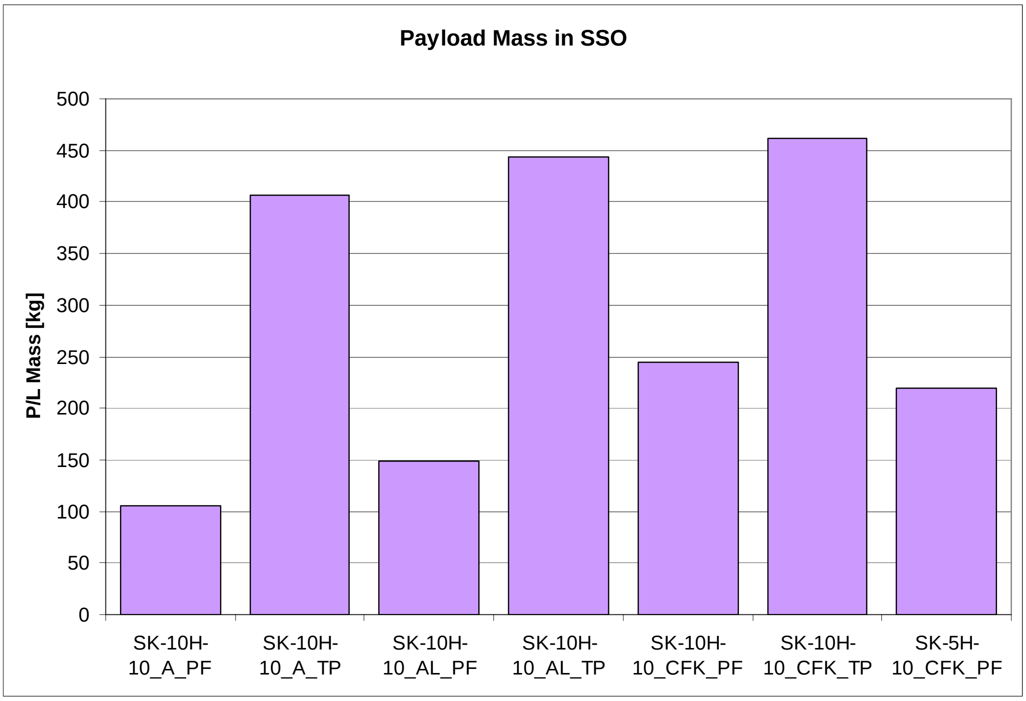

Figure 7: Achievable payload mass with different engines and upper stage architectures of micro-launcher, PF: pressure fed, TP: turbopump fed, tank materials A: aluminium, AL: aluminium-lithium, CFK: filament wound CFRP) [5]

Obviously, the investigated launcher configurations with assumed tank mixture ratios between 3 and 3.7 are not optimal for an operational system. However, the influence on the two engine cycles is much different. The negative impact on stage dry mass is between 50 kg and 100 kg in case of the pressure fed systems. Thus, an improved mixture ratio for these stages could deliver between 25 % and 100 % higher payload to orbit. This is far more relevant than the minor impact of MR on engine Isp. For the turbopump fed stages the negative impact of the comparably large hydrogen tank size is only about a few 10 kg and is not exceeding 10 % of the achieved payload values in Figure 7.

The achievable payload performance of the generic micro-launcher with low mixture ratio cryogenic upper stage is quite remarkable. However, the conservative assumption on acceptable MR based on the current status of chamber material development should not be misinterpreted as the upper limit of effusion cooled CMC chambers. Several promising approaches, in materials, injector types and patterns or cooling schemes are to be tested and qualified to improve the situation. A systematic research and testing campaign should be started soon.

4 Development Perspective and Conclusion

The current status of experimental research on ceramic thrust chambers with effusion cooling allows starting an assessment on potential first applications of this new technology in launcher stages. The need of continued fundamental research in the fields of CMC materials, cooling strategy, injectors etc. is identified in order to reach superior performance characteristics required for one day replacing today's chamber technology in larger stages.

However, an application oriented development approach should help focusing on increasing TRL in a medium term perspective. The only affordable way will be a roadmap based on low risk technology steps in engines and stages of small size. These applications cannot have overly ambitious requirements on re-ignitability or deep throttling capability to be successful. On the other hand, the early applications should already demonstrate sufficient operational benefit. Therefore, their performance has at least to match that of conventional storable propellant configurations. The results of the study presented in this paper demonstrate that this requirement is feasible, even under conservative assumptions of a high cooling flow resulting in unfavorable low engine and tank mixture ratios.

Keeping the above general statements in mind, the first flight demonstration could be on a micro-launcher kick-stage with a very low cost gas engine with less than 2 kN thrust. The development and qualification could be finished within 8 years by the end of this decade. Such a simple stage could already deliver a limited payload to orbit, but its overall performance is too low for a viable system operational over a longer period.

In case of an overall successful development perspective after qualification of the kick-stage, a more demanding small LH2-LOX upper stage including re-ignition capability is to be developed. The DLR-SART preliminary launcher system study shows that based on a generic four stage micro-scale launcher a gas-generator-cycle turbopump fed engine is far more attractive than a pressure fed system. This is mostly due to the fact that all effusion cooled engines with cryogenic propellants will require hydrogen pre-heating before injection which always makes a gas generator necessary. Then the gas generator can also be used for powering a turbine with attached pumps. The major challenge and hence cost factor of this approach is the development of yet non-existing efficient turbomachinery at this small scale. Identification of secondary applications for such small turbomachines would be helpful in justifying the cost of its dedicated development. The more complex approach of such small cryogenic upper stages with ceramic chamber will not see operation before 2022 as long as only moderate funding will be available.

Acknowledgements

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References

1. Louaas, E. et al.: Aldebaran: a "system" demonstrator project for new generations of space transportation, now entering in the phase A, IAC-09.D2.6.7, October 2009

2. NN: Analyse des Entwicklungspotentials der CMC-Faserverbund-Keramik für ihre Anwendung in Raketenmotoren, Astrium GmbH Angebot N° 08.2.0389, September 2008

3. H. Hald, A. Herbertz, M. Kuhn, M. Ortelt: Technological Aspects of Transpiration Cooled Composite Structures for Thrust Chamber Applications, AIAA -2009-7222, Bremen 2009

4. Herbertz, A., Systemanalytische Untersuchung einer Brennkammer in faserkeramischer Bauweise von Raketenantrieben, Dissertation, RWTH Aachen, Aachen 2008

5. Dietlein, I.; Sippel, M.: Systemaspekte des Einsatzes von Keramischen Brennkammern im Raumtransport, Anwendung auf einem Kleinträgersystem, SART TN-012/2009, Edition 1 / Revision 0, February 2010

6. Dietlein, I. et al: Kickstage for Aldebaran, SART TN-015/2008, Edition 1 / Revision 0, Bremen November 2008

7. Rackemann , N.; Preclik, D.; Keerl, P.N.: Effusionsgekühlte CMC-Brennkammer, Skalierung Betriebswerte für 2 kN ←→ 10 kN Referenz-Schubkammern aufgrund der KSK-KT Test Kampagnen Daten, Memorandum D2857-M-AMU-33, Astrium Ottobrunn, 14.08.2009

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