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Black Engine Ceramic Rocket Propulsion

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Abstract

Today the field of space transportation is of high interest, as a result of the growing competition world-wide. Of particular note are the new Space Companies in the US, which have increased the pressure that the global space transportation industry is under. With this is mind, DLR proposes a new ceramic rocket engine concept, with the goal of increased efficiency and reliability. Additionally, weight, cost and operational risk will be reduced, aiming to be competitive on the global scale in the future.

Concerning these requirements, a specific rocket thrust chamber design, based primarily on the application of transpiration cooled porous and thermo-chemically resistant CMCs as inner combustion chamber liner material, is favored, aiming on the improvement of today's high performance standards, e.g. typical high performance main stage or upper stage propulsion systems.

After more than twenty years of intensive technology development, DLR offers within its *Black Engine* technology program a large portfolio of fiber reinforced structure systems for functional components in rocket thrust chambers or entire rocket engines respectively, using additionally CFRP for load carrying structure components or transpiration lubricated CMC journal bearings for lifetime increase of rocket TPs.

One major focus lies on the high ratio of thermal and mechanical load de-coupling capability, promising rocket engines with lifetime and maintenance standards comparable to aviation engines, which will be a significant progress for highly reusable main stages in the near future.

The exclusively transpiration cooled ceramic rocket thrust chamber system, leads to several additional system improvements. First, evaluations are ongoing considering a new injection cooling method, where the porous CMC wall takes inherently both the function of cooling and injection. This new method is applied in a novel design concept, reducing significantly typical pressure loss in the combustion chamber process and promising an overall engine efficiency increase of more than 5 %. Not only is the increase in efficiency important, but also the increase in reliability by applying higher blow rates for higher inner wall safety.

Technology approaches described above are a world first. Established international producers of launcher propulsion systems usually use metal designs for thrust chambers, with recent improvements coming from AM methods, increasing manufacturing efficiency. DLR's new system improvements using porous CMCs as key components in rocket thrust chambers as well as CMC journal bearings in rocket TPs predicts upcoming international competitiveness in the field of space propulsion.

Keywords: Innovative Rocket Propulsion Systems, Transpiration Cooled CMC Rocket Thrust Chambers, Long-Life Rocket TPs, Transpiration Lubricated CMC Journal Bearings, New Space, Space Transportation

Nomenclature

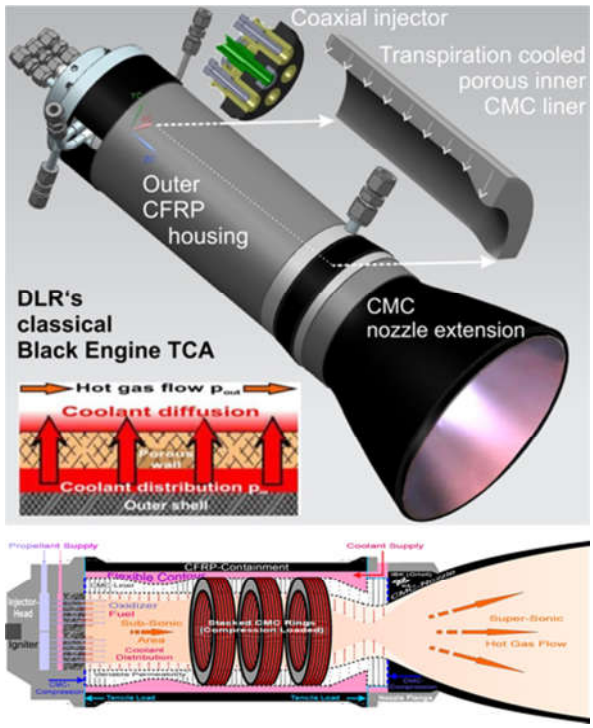
a	- Length of non-porous part
b	- Length of journal bearing
p	- Pressure
α	- Permeability
τ	- Coolant Mass Flow Rate
μ	- Dynamic viscosity
W	- Flow velocity
I_{SP}	- Specific Impulse
E	- Porosity
CTE	- Coefficient of Thermal Expansion
C_p	- Isobaric specific heat
Q	- Heat flux
A	- Area
T	- Temperature
$\dot{m}$	- Mass flow

Acronyms/Abbreviations

C/C	- Carbon/Carbon
C/C-SiC	- Carbon/Carbon-Silicon Carbide
C/SiCN	- Carbon/Silicon Carbon Nitride
C/SiCO	- Carbon/Silicon Carbon Oxide
CMC	- Ceramic Matrix Composite
LSI	- Liquid Silicon Infiltration
PIP	- Polymer Infiltration &Pyrolysis
SLM	- Selective Laser Melting
TCA	- Thrust Chamber Assembly
FKFL	- Transpiration Lubricated CMC Journal Bearing
TP	- Turbo Pump
LOX	- Liquid Oxygen
LCH4	- Liquid Methane
LH2	- Liquid Hydrogen
O/F	- Ratio of Oxidizer and Fuel
LPRE	- Liquid Propellant Rocket Engine
BE	- Black Engine
SME	- Small and Medium Enterprise
WEPA	- WEPA Technologies GmbH (SME)
TU-KL	- Technical University of Kaiserslautern
BEA	- Black Engine Aerospace UG (SME)
HYPE	- Hyperboloid Engine
TRL	- Technology Readiness Level
RPA	- Rocket Propulsion Analysis [1]
RLV	- Reusable Launch Vehicles
ELV	- Expendable Launch Vehicles

Subscripts

p	- Pressure
s	- System
c	- Chamber
sp	- specific
e	- Exit



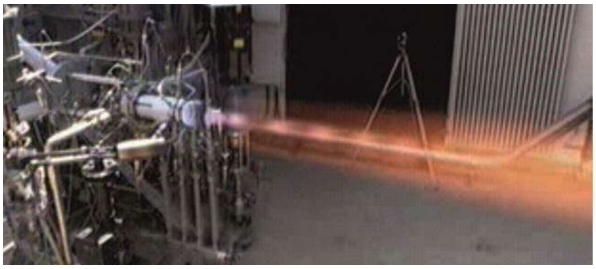


Fig. 1. DLR's classical CMC-TCA concept (top), tested successfully at the European Research and Technology Test Facility P8 in 2010 (see ST5-demonstrator, bottom).

1. Introduction

Continuing the development of the transpiration cooled CMC rocket thrust chamber technology (see Fig. 1) within the last two decades, which formerly used LOX and LH2 as cryogenic propellants [2÷13], the next goal will be to transition to the use of LOX and LCH4.

For this purpose, a TRL 5 demonstrator TCA is under construction, which will be up-scaled to 60 kN of thrust, compared to the former maximum demonstration level of about 30 kN (LOX/LH2). The upcoming 60 kN demonstration will represent an industrial transfer of a full-scale level targeting, focusing on micro and mini launchers in the New Space scene (see section 2.).

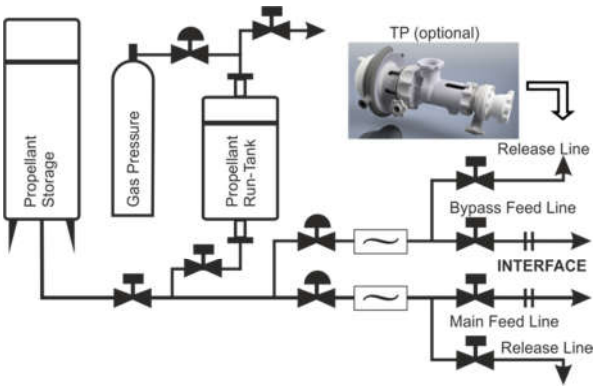


Fig. 2. Principle of the future BE-specific test set-up (only one propellant line is representatively depicted).

In collaboration with the industrial partners WEPA and BEA, who provide an adequate TP and a test facility, the demonstration of an entire rocket engine in 2020 is targeted (see sections 3. and 4.).Principles of the test environment are given in Figures 2 ÷ 4.

The available TP prototype, provided by the German WEPA, is a one-shaft TP-unit (see Fig. 3), performing 100 bar outlet pressure and 19.6 kg/sec overall mass flow at O/F = 2.8 ÷ 3.3. It will be driven by a gas generator, based on DLR technology. BEA plans an entire LPRE, where DLR's TCA technology and WEPA's TP shall be combined on BEA's PYROSTAR test facility (similar to Fig. 4). The latter is planned as a mobile rocket engine test facility, which can be transported by trailers and over-sea-containers.

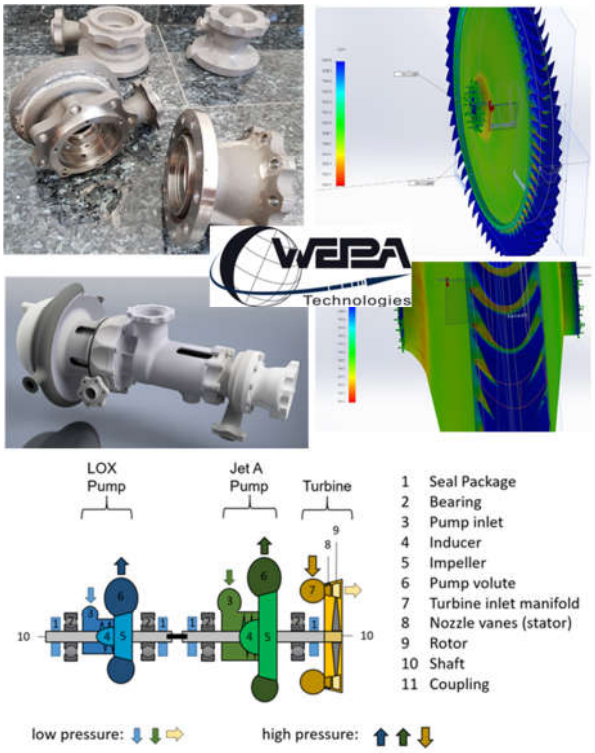


Fig. 3. Components of WEPA's one-shaft TP-unit (image source: WEPA).

BEA's PYROSTAR test facility provides up to 150 bar interface pressure, overall cryogenic mass flow up to 40 kg/sec as well as divisions in main feed lines and bypass feed lines for the propellants and coolants.

The next significant step in the innovation of DLR's ceramic thrust chamber development will be seen in the revolutionary sub-scale combustion chamber geometry, the so-called dual-shell hyperboloid thrust chamber (HYPE) design (see Fig. 5 and section 3). Combined with transpiration operated injection cooling, on the one hand the efficiency of the rocket engine shall be increased, and on the other hand the operation safety, lifetime and reliability shall be improved significantly.

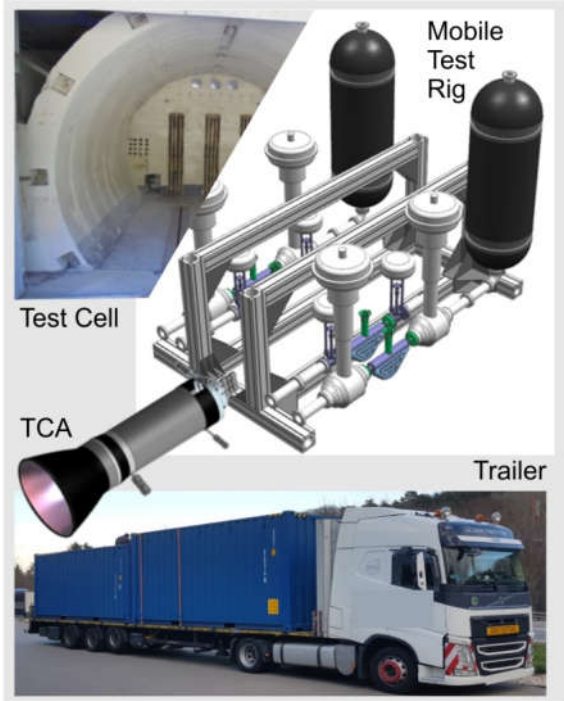


Fig. 4. Black Engine test environment.

In parallel a new transpiration lubricated CMC journal bearing technology for long-life rocket TPs is under development (see Fig. 5 right and section 4).

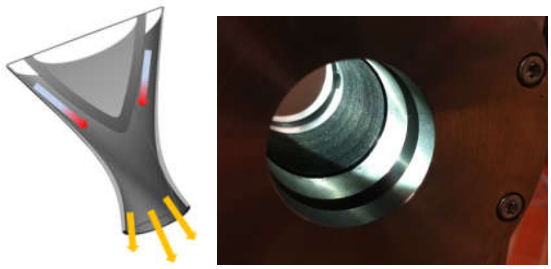


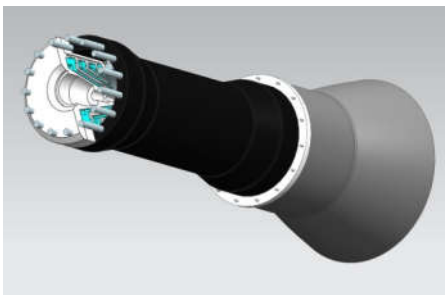
Fig. 5. HYPE design principle (left). Demonstrator for transpiration lubricated CMC journal bearings.

The goal of using all unique technologies mentioned previously is the implementation of highly reliable, safe and efficient future rocket engines at a low cost.

2. Six tons LOX/LCH4 TCA demonstrator

Since March 2019, the ongoing DLR internal Black Engine (BE) project is developing a 60 kN LOX/LCH4 full-scale TCA, with a transpiration cooled CMC design and targeting TRL 5 (see Fig. 6). It represents a directly extrapolated design (classical cylinder-Laval-design), derived from the LOX/LH2 demonstrator development in the past (see Fig 1), and combined, in the depicted version, with the optional new and up-scaled cone

injector component. The nominal chamber pressure amounts to 70 bars.



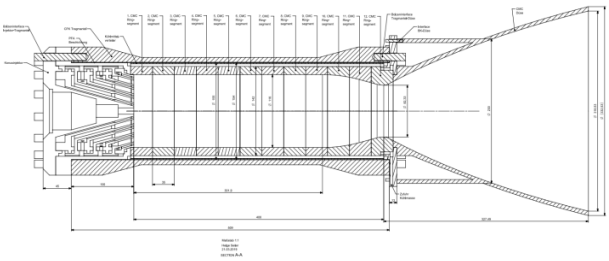


Fig. 6. Draft status of the TRL5-LOX/LCH4 demonstrator TCA (cone injector configuration).

The expansion nozzle exit pressure is adjusted to 0.5 bars, the nominal O/F to 3.1, whereas the starting O/F will be 2.8 due to operational safety reasons. The total TCA length amounts to 882 mm. In detail the combustion chamber length amounts to 400 mm, the injector to approximately 155 mm and the nozzle extension to approximately 340 mm. The operation settings have been derived by the RPA tool and are given in table 1, as well as the TCA geometry data, given in table 2. The magnitudes used are explained in Figure 7.

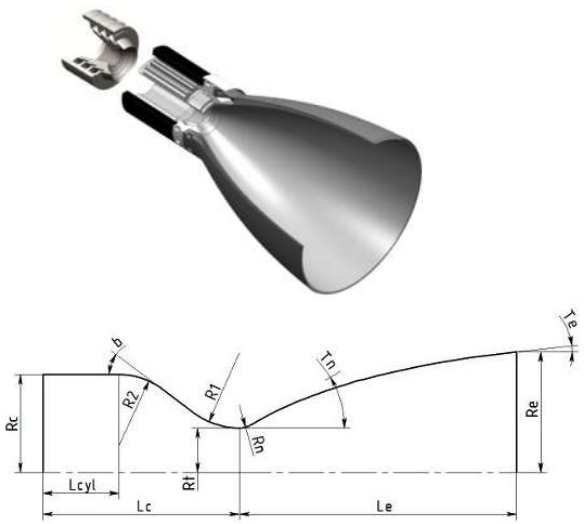


Fig. 7. Geometry principle and parameters of the TCA.

Table 1. Operational data

Thrust, mass flow rates and propellant temperatures		
Chamber thrust (vac)	64.38	kN
Specific impulse (vac)	331.55	s
Chamber thrust (opt)	60.30	kN
Specific impulse (opt)	310.58	s
Total mass flow rate	19.80	kg / s
Oxidizer mass flow rate	14.59	kg / s
Fuel mass flow rate	5.210	kg / s
LOX temperature	90.2	K
LCH4 temperature	111.6	K

Table 2. Geometry data

TCA geometry with parabolic nozzle					
D_c	116.00	mm	b	15.00	deg
R_2	216.73	mm	R_1	41.81	mm
L*	731.32	mm	L_{cyl}	305.53	mm
L_c	400.00	mm			
D_t	83.61	mm			
R_n	15.97	mm	T_n	28.14	deg
L_e	339.68	mm	T_e	13.78	deg
D_e	334.40	mm			
A_e/A_t	16.00	-			
Le/Dt	4.06	-	L_e/c15	72.26	%

2.1 Starting CFD analysis

As a first step, a fully coupled two-domain (see Figure 8) CFD simulation has been performed. This preliminary documentation gives a summary of the set-up and assumptions made, as well as the initial results and directions of further model developments.

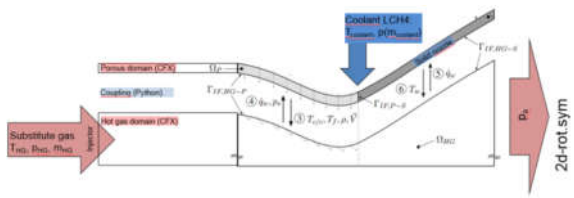


Fig. 8. Two-domain CFD set-up.

The reference test case that is simulated is defined by a coolant mass rate of 1.6 kg/s, hot gas inlet temperature of 3352.5 K, a hot gas mass flow rate of 18.3 kg/s. Furthermore, the exit pressure of the nozzle is 45700.0 Pa and the coolant temperature at the inlet to the porous liner is 111.6 K. The combustion products entering the combustion chamber at the inlet of the hot gas domain are the result of combusting LCH4 and LOX at an O/F of 2.8. The combustion product is modelled as an ideal gas for simplicity. Finally, to begin the simulation, the temperature of the porous wall and

flow entering the nozzle from the porous wall must be initialized. Currently a constant value of 1000 K is being used. The liner is modelled using the porous solver in ANSYS CFX (see Fig. 10).

Fig. 10. ANSYS CFX model of porous liner.

The porous liner is made from one of DLR's new and representative CMCs OCTRA 20 having a CTE of 15 W / mK and a c_p of 900 J / kgK. A porosity of 0.111 is currently used. The top surface of the liner is modelled as an inlet, having a mass flow rate of 1.6 kg/s. The heat flux into the incoming coolant from the relatively hot porous wall is given by:

$$Q_{in} = \frac{\dot{m} \cdot c_p}{A} (T_f - T_c) \qquad (1)$$

where $\dot{m}$ is the mass flow into the inlet, c_p is the average specific heat coefficient over the inlet, A is the area of the inlet, T_f is the average fluid temperature in the inlet and T_c = 111.6 K is the temperature of the coolant in the reservoir.

The bottom surface of the liner is modelled as an outlet, having a pressure and heat flux determined from the solution of the hot gas domain. Symmetry boundary conditions are used on the front and back surface, since only a slice of a complete chamber is modelled. Finally, the walls separating the porous samples are modelled as adiabatic walls.

The CFD model of the rocket nozzle is depicted in Figure 11. It is modelled using the fluid solver in ANSYS CFX.

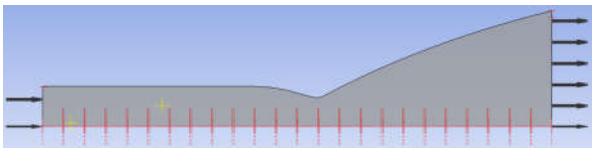


Fig. 11. ANSYS CFX model of rocket nozzle.

The fluid is modelled as the combustion product of LCH4 and LOX. The left-most boundary is modelled as an inlet, having a mass flow rate of 18.3 kg/s, static pressure of 70bars and static temperature of 3352.5K. The right-most boundary is modelled as an outlet, having a static pressure of 45700.0Pa. The front and back surfaces are modelled as symmetry boundary conditions, since only a slice of the complete geometry is modelled. The bottom surface is modelled as an axis boundary condition. Finally, the top surface is modelled as an isothermal wall with momentum sources, over the porous liner, and an adiabatic wall for the remainder of the nozzle. The temperature distribution of the isothermal wall is determined by the solid temperature of the outlet in the porous liner, except in the first iteration a *kick-off* value of 1000.0K is used. Furthermore, the temperature of the momentum sources is determined by the fluid temperature in the porous liner, again except in the first iteration a *kick-off* value of 1000.0K is used. The total mass flow rate of the momentum source is equal to the total mass flow rate into the porous liner. For all iterations after the first, the distribution of the momentum source mass flow rate is determined from the outlet distribution of the mass flow.

So far only the rocket nozzle has successfully converged. The temperature distribution in the rocket nozzle is illustrated in Figure 12.

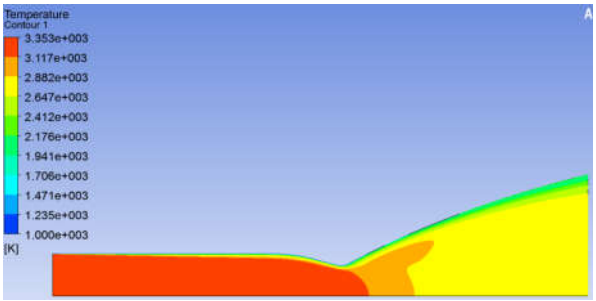


Fig. 12. Static temperature distribution in rocket nozzle.

The lowest temperature is 1000.0K, which is the boundary condition set by the initialization of the porous liner. Furthermore, the highest temperature is 3352.5K, set by the inlet value. The pressure distribution in the rocket nozzle is given in Figure 13.

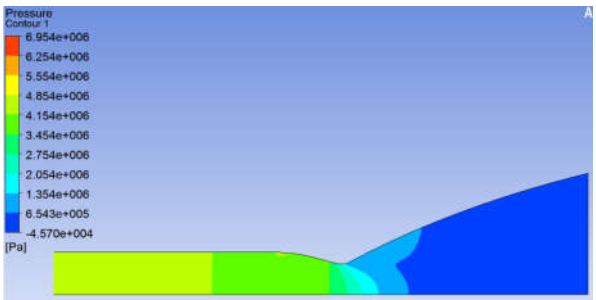


Fig. 13. Static pressure distribution in rocket nozzle.

Note the reference pressure is 45700Pa, therefore the maximum value is 70bars set by the inlet value. The Mach number in the nozzle is shown in Figure 14.

The exit Mach number is just over 2.9, a value of 3.56 was reported by RPA without injection. The initial pressure distribution on the outlet of the porous liner is depicted in Figure 15. The pressure in the constant radius section is approximately 43bars, before dropping rapidly to about 0.75bars at the throat section. The initial heat flux distribution applied to the outlet of the

porous liner is illustrated in Figure 16. As expected the heat flux starts high, before dropping off as a coolant film is laid. The heat flux increases slightly in the convergent section of the nozzle, before reducing as the flow expands in the divergent section.

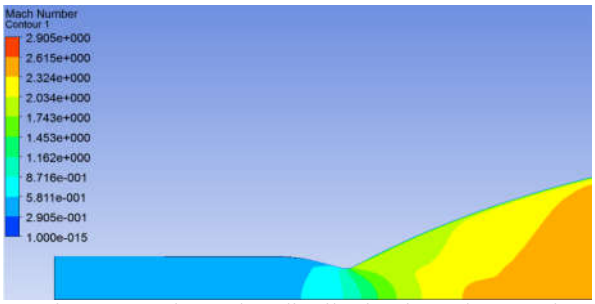


Fig. 14. Mach number distribution in rocket nozzle.

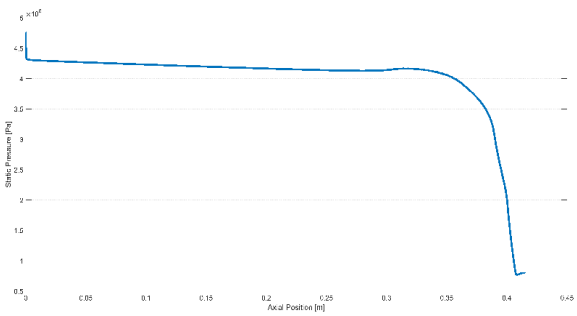


Fig. 15. Pressure distribution applied to outlet of porous liner.

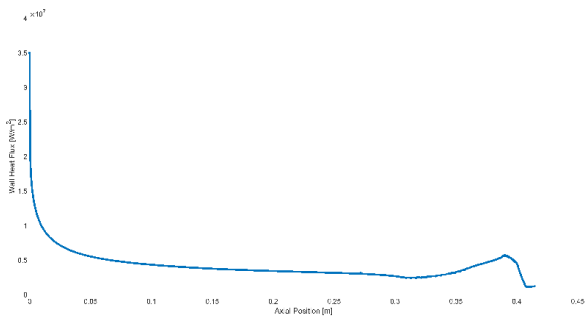


Fig. 16. Heat flux distribution applied to outlet of porous liner.

So far a Python script has been developed that can set-up the model for a two-domain simulation of the BE rocket nozzle. The hot gas domain converges to an initial solution; however, currently the porous domain is not converging with the given boundary conditions. From numerical experimentation it seems either the mass flow rate is too low or alternatively the outlet pressure is too high. Nonetheless this needs to be confirmed in the further modelling improvement.

3. Injection cooled hyperboloid TCA technology

Within the long lasting transpiration cooled CMC rocket thrust chamber development one potential and absolutely significant improvement became more and more visible. Particularly in conjunction with the unconventional cone injector design, that creates a hyperboloid injection spray shape, it became beneficial to derive the design of an innovative dual-shell hyperboloid combustion chamber geometry (see Figure 17 and Figure 5 left) [14÷18]. The requirement of a quasi-cylindrical cross-section-area-constant combustion zone leads mathematically to a two-shell hyperboloid contour at the inner core (or insert).

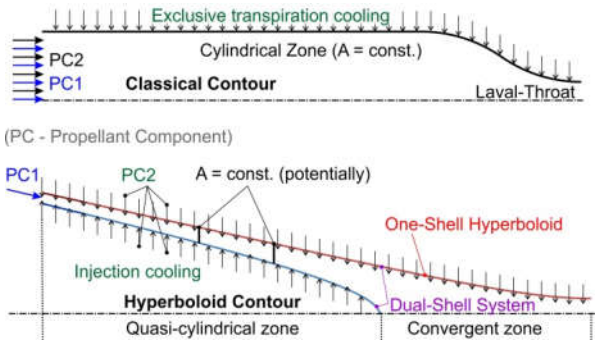


Fig. 17. Transition from classical to dual-shell hyperboloid rocket combustion chamber geometry.

The basic idea consists of the transition to the combustion chamber geometry, where the chamber wall gets the function of the injector, so that typical pressure loss, generated principally by the traditional high performance regenerative cooling method, can be significantly reduced. The latter opens the potential for further engine efficiency increase.

The injected fuel is now simultaneously the coolant. This is practically feasible because of the creation of a narrow ring-shaped combustion chamber zone directly in the mixing section close to the injection face-plate (see Fig. 18).

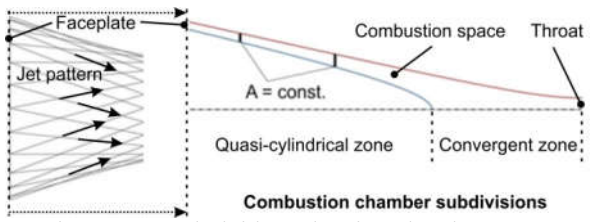


Fig. 18. Hyperboloid combustion chamber zones and oxidizer injection pattern for diffusion jet flaming.

Using the principle of diffusion jet flames, where a multitude of very thin oxidizer jets will be shot into a plenum of fuel created by the wall injection, highly efficient combustion can happen. Figure 19 shows the temperature plot of a combustion process in full-scale configuration of a potential gas generator design (see Fig. 22, for the TP drive of the 60 kN rocket engine) under 60bars hot-gas pressure, using LOX and LCH4 at O/F of approximately 0.27.

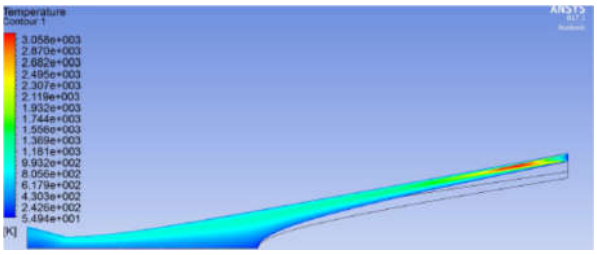


Fig. 19. HYPE Temperature plot in the TP gas-generator configuration.

Even in the main chamber configuration, the combustion promises to be highly efficient and, in parallel, the fuel diffusion blow rate at the inner wall surface will be so high, that the wall itself is over-proportionally cooled. This leads directly to a thermally uncritical and safe inner wall operation. Additionally the boundary layer, especially in the near-throat-region, shows significantly higher homogeneity (as given by the viscosity plot in Figure 20), which leads, as a direct consequence, to lower surface temperatures (see Fig. 21).

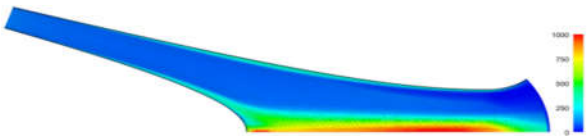


Fig. 20. Viscosity plot in the dual-shell hyperboloid combustion chamber geometry.

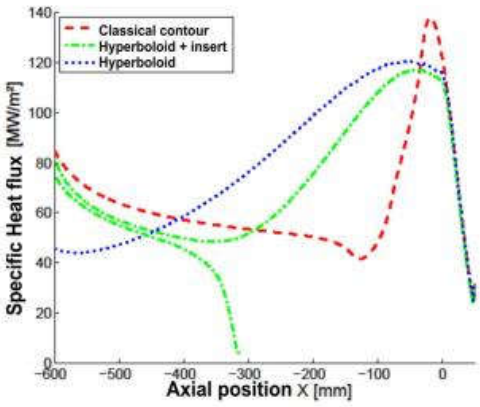


Fig. 21. Viscosity plot in the dual-shell hyperboloid combustion chamber geometry.

The summer 2019 first cold spray tests have been performed with the Injector of the new gas generator component (see Fig. 22).

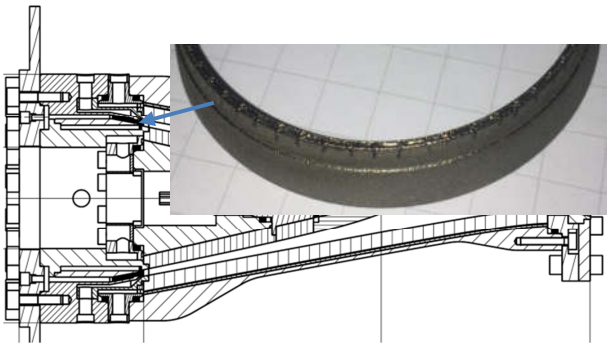


Fig. 22. Demonstrator component for the full-scale gas generator function and sub-scale HYPE main chamber investigation incl. SLM-printed LOX injection element.

Within DLR's SeLEC project, using water and gaseous nitrogen as propellant replacement media (replacing LOX and LCH4), the function principal was tested. Validation, on the one hand, the accurate flow through SLM printed LOX injector elements (see Fig. 22) and, on the other hand, the functionality of the overall jet-pattern in the hyperboloid spray-shape (see Fig. 23).

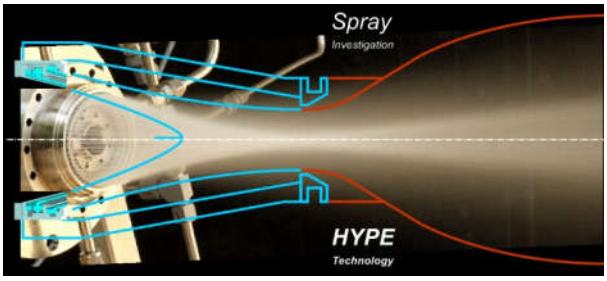


Fig. 23. Cold spray investigation at the hyperboloid gas generator component, using water and gaseous nitrogen as replacement media.

Finally, it can be stated that the HYPE technology promises significant improvements for high efficiency and high-performance rocket engines at very safe and long-life operation conditions. Regarding these properties, a competitive technology option for future space transportation applications is represented.

4. Transpiration lubricated CMC journal bearings for future rocket TPs

In terms of the demand of future RLV technology, not only thrust chambers of the rocket engines, but also TPs must have the capability of high cycle resistance.

The cryogenic conditions and the fast-rotating shafts in the TP for rocket engines initiated an investigation of

a new hydrostatic transpiration lubricated journal bearing, consisting of a microporous CMC. The idea of using a microporous CMC material, such as C/C, C/C-SiC, C/SiCN or C/SiCO arose from the experience with the transpiration cooled CMC rocket thrust chambers at DLR, and it will be presented more in form of a short overview in the following section.

One significant advantage of a journal bearing over other types of bearing systems is that journal bearings have a better damping effect on the shafts, consequently the reusability of future rocket TPs will be guaranteed. Most notably, the very first test in 2015 at the TU-KL has shown an excellent performance prediction of the new hydrostatic journal bearing technology [19÷21]. WEPA has shown high interest in the use of this sophisticated technology in its own TP development.

The structural principle of the new FKFL technology is similar to the material and design basics of the transpiration cooled CMC combustion chamber technology (see Fig. 24).

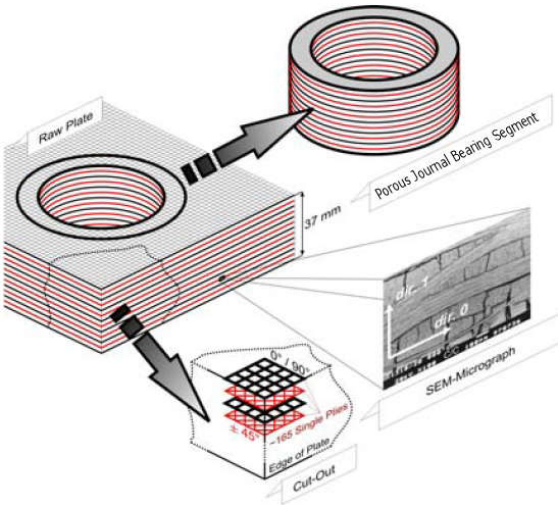


Fig. 24. Basic manufacturing process of the CMC ring elements for the FKFL.

The CMC bearing sleeves will be extracted from flat CMC plates, so that the fibre ply lay-up is oriented axis-normal to the longitudinal direction of the rotating bearing shaft. Two major advantages are given in this design approach: a) the fibre plies generate flow baffles for the preferably radial diffusion direction of the lubricating medium and b) in case of inner contact surface degradation the load carrying fibres won't be cut, but only the end of load carrying fibres will be shortened (see Fig.25, case B). The schematic of the journal bearing is given in Figure 26. TU-KL, represents German academic excellence in turbo pumps and flow technology of journal bearings, currently develops numerical tools for the functional bearing design in a common project (DLR's FKFL project) for the planned industrial technology transfer, funded by DLR's technology marketing.

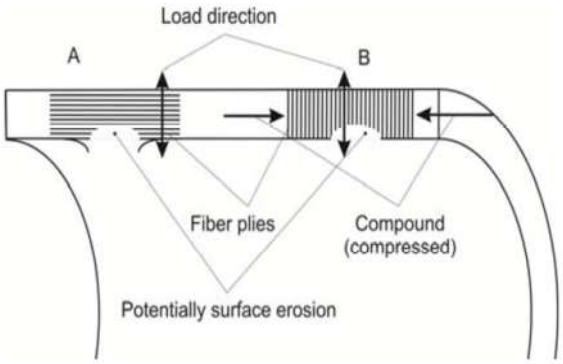


Fig. 25. Composite-specific functional principle of the CMC bearing sleeves.

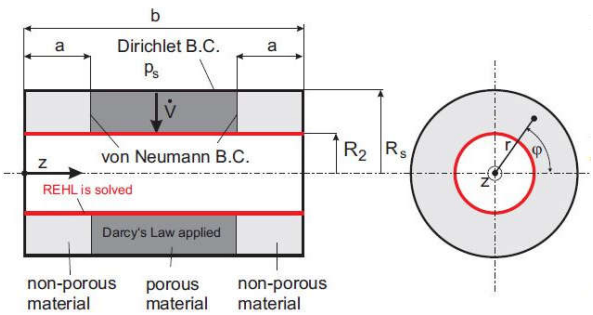


Fig. 26. Schematic of the ceramic bearing sleeve.

Compared to a non-porous bearing with only one hole, the hydrostatic journal bearing with micro-porous CMC material allows an increase in the load capacity by up to ten times higher and the stiffness is up to eight times higher (see Fig. 27 and 28).

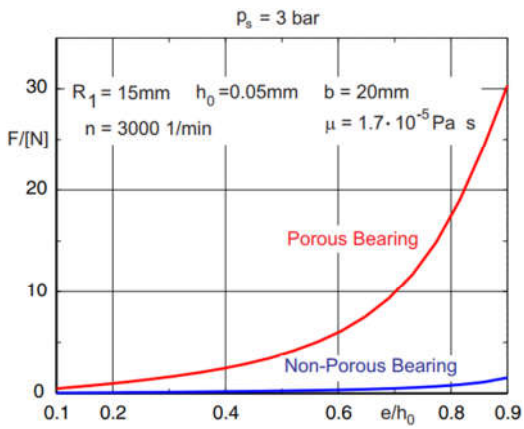


Fig. 27. Load capacity comparison between standard and micro-porous CMC journal bearing.

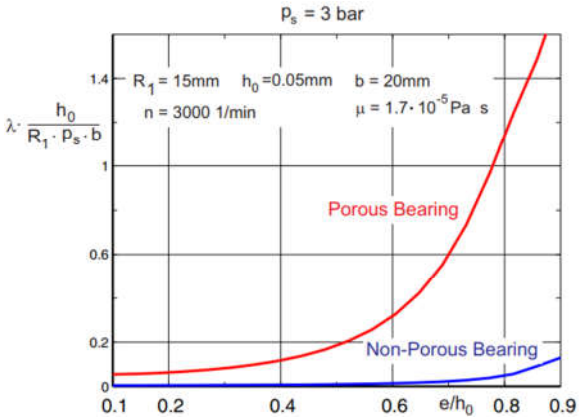


Fig. 28. Stiffness comparison between standard and micro-porous CMC journal bearing.

The industrial partner WEPA plans for the optional implementation of this technology in its TP system for the 60 kN LOX/LCH4 application (see Fig. 3). After basic pre-testing programs at DLR this application offers an investigation environment under original operation conditions.

5. Conclusion

Summarizing the activities in the field of innovative CMC high performance rocket engine development, currently a multitude of single technology elements fit together in a very constructive manner. First of all the transition from TRL 5 of the LOX/LH2 operation on up to 30 kN sub-scale level, to TRL 5 on 60 kN full-scale level using LOX and LCH4. Secondly further investigation and development of the improved HYPE technology is very promising, targeting future competitive rocket engines world-wide. The implementation possibility of unique TPs, given by WEPA, into the entire development program is of significant advantage. Specifically the development and implementation capability of the new CMC journal bearing technology, promising potential long-life applicability for TPs and supported essentially by TU-KL, suits the request of reliable future RLVs. Finally the availability of the accompanying construction of a suitable test capacity, supported by BEA, underlines the general overall capability of bringing forward successfully innovative and competitive future space transportation technology, developed by DLR in Germany.

6. Acknowledgement

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References

[1] A. Ponomarenko. 2017-03-30. RPA – Tool For Rocket Propulsion Analysis. *URL: www.propulsion-analysis.com.*

[2] M. Ortelt, A. Herbertz, H. Hald I. Müller. 2013. Advanced Design Concepts for Ceramic Thrust Chamber Components of Rocket Engines. *5th EUROPEAN CONFERENCE FOR AERONAUTICS AND SPACE SCIENCES, Munich, Germany.*

[3] A. Herbertz, M. Ortelt, I. Müller, H. Hald. 2012. Transpiration-Cooled Ceramic Thrust Chamber Applicability for High-Thrust Rocket Engines – Scaling of KSK Test Results. *48th AIAA/ASME/SAE/ASEE Joint propulsion Conference & Exhibit, Atlanta, Georgia.*

[4] M. Ortelt, A. Herbertz, H. Elsässer, H. Hald I. Müller. 2012. Structural Investigations on Cryogenically Operated and Transpiration Cooled Fiber Reinforced Rocket Thrust Chambers. *48th AIAA/ASME/SAE/ASEE Joint Propulsion Conference & Exhibit, Atlanta, Georgia.*

[5] M. Ortelt, F. Breede, A. Herbertz, D. Koch, H. Hald. 2013. Current activities in the field of ceramic based rocket engines. *ODAS 2013- 13th ONERA-DLR Aerospace Symposium, Palaiseau, France.*

[6] A. Herbertz, M. Selzer. 2013. Analysis of Coolant Mass Flow Requirements for Transpiration Cooled Ceramic Thrust Chambers. 29th *International Symposium on Space Technology and Science, Nagoya, Japan.*

[7] M. Selzer, S. Schweikert. 2013. Transfer of an analytical transpiration cooling model to the cooling analysis of rocket combustion chambers made of ceramic matrix composites. *5th European Conference for Aeronautics and Space Scoences (EUCASS), Munich, Germany.*

[8] S. Gordon, B.J. McBride. 1994. Computer Program for Calculation of Complex Chemical Equilibrium Compositions and Applications. *Techn. Rep., NASA, USA.*

[9] T. Linder. 2014. Numerische Untersuchung von Transpirationsgekuehlvorgaengen mit Augenmerk auf die Kopplung zwischen Heiss- und Kuehlgasstroemung. *Bachelorthesis Institut fuer*

Thermodynamik der Luft- und Raumfahrt,. University of Stuttgart, Germany.

[10] H.W. Coleman, B.K. Hodge, R.P.Taylor. 1984. A Re-Evaluation of Schlichting's Surface Roughness Experiment. *Journal of Fluids Engineering, Vol. 106.*

[11] M.M. Pimenta, R.J. Moffat, W-M. Kays. 1975. The Turbulent Boundary Layer: An Experimental Study of the Transport of Momentum and Heat with the Effect of Roughness. *Interim Report, Stanford University, CA.*

[12] Ansys, Inc. ANSYS CFX-Solver Modelling Guide 17.1. April 2016.

[13] M. Ortelt, H. Hald, I. Mueller; 2014. Status and future perspectives of the CMC rocket thrust chamber development at DLR. *65th International Astronautical Congress, Toronto, Canada.*

[14] S. Soller, B. Kniesner, C. Maeding, R. Kaess, R. Behr, A. Dahlaus, R. Blasi. 2013. Astrium's LOX/LCH4 Gas Generator Development Programme. *5th EUROPEAN CONFERENCE FOR AEROSPACE SCIENCES (EUCASS 2013).* München, Germany.

[15] Educational Manuscript. Fluidreibungswiderstand an Oberflaechen. *URL: http://hydro.ifh.uni-karlsruhe.de/download/Kap07ps.pdf. University of Karlsruhe, Germany.*

[16] S. Michaelides, 2016. Ansys-CFX-Simulation einer keramischen Gasgenerator-Brennkammer fuer einen LOX/LCH4 Raketenantrieb der 35 kN Klasse. *Internal report ILR-RFS GB 16-07, Professur fuer Luft- und Raumfahrt at the Technical University of Dresden and Institute of Structures and design at DLR Stuttgart, Germany.*

[17] T. Schleutker, 2014. Analytischer und numerischer Vergleich des Wandwaermestroms bei konventionellen und hyperbolischen Brennkammern. *Diplomarbeit, Institute of Space Systems at the Technical University of Stuttgart and Institute of Structures and design at DLR Stuttgart, Germany.*

[18] H. Seiler, 2017. Structural design of the hyperboloid gas-generator combustion chamber for a 35 kN LOX/Methane rocket propulsion system. *Master work IRS-17-SO48, Institute of Space Systems at the University of Stuttgart and Institute of Structures and design at DLR Stuttgart,*

[19] B. Gwiasda, S. Fleder, M. Appel, M. Ortelt, M. Boehle, 2016. Design and testing of radial gas bearings with porous media, using fiber-reinforced C/C composites. *3rd International Rotating Equipment Conference IREC, Duesseldorf, Germany.*

[20] M. Boehle, Y. Gu, A. Schimpf, Two flow models for designing hydrostatic bearings with porous material, *AJKFLUIDS 2019, Fluids Engineering Divisions Summer Meeting, ASME Paper, AJKFLUIDS 2019-4657, August 2019 (to be published).*

[21] H. Seiler, 2019. Development of a hydrostatic journal bearing with micro porous CMC material. *54th AIAA/ASME/SAE/ASEE Joint propulsion Conference & Exhibit, Atlanta, Indianapolis, USA.*