Current Activities for the Development of a LOX/LCH4 Regenerative Thrust Chamber at the Italian Aerospace Research Centre

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ABSTRACT

The present paper reports the current progress on the development of a LOx/LCH4 rocket engine demonstrator in the framework of the HYPROB program.

The HYPROB program, a building element of the Italian Aerospace Propulsion Program, is carried out by CIRA under contract by the Italian Ministry of Research with the main objective to enable and improve National system and technology capabilities on liquid rocket engines (LRE) using propulsion systems for future space applications, with specific regard to LOX/LCH4 technology, in coherence with the long-term vision of the Italian Space Agency on Space Propulsion and the needs of industrial national stakeholders.

INTRODUCTION

Propulsion systems based on hydrocarbons, either liquid or hybrid, represent nowadays a major technology challenge for future launchers and space transportation systems.

Methane is one of the most interesting solutions as propellant for liquid rocket engines, in combination with Oxygen, due to good performances achievable in terms of specific impulse ($Isp_{ref} \sim 380$ s) combined with operation advantages, such as storability, low toxicity, availability and production cost, as compared to hydrogen. Additional features of methane regard its good cooling capability and well known material compatibility, that make it ideal for regenerative thrust chambers.

In a long term perspective, such a propulsion technology may encompass a wide range of propulsion systems, from launcher main stages up to small thrusters, but present envisaged applications regard mostly:

- upper stages of small launchers;
- primary propulsion systems for interplanetary missions, such as ascent ad landing modules.

HYPROB FRAMEWORK

With the aim of supporting and promoting the consolidation and the evolution of competences in the field by the national scientific and industrial community, an integrated national vision for mid-long term R&D activities has been defined, which takes the maximum benefit from both Ministry of Research and University initiatives and ASI on going and future programs, then preparing for the future technical challenges.

As far as chemical propulsion is concerned, the background earned by the Italian community is strongly based on solid rocket motors, that have mainly contributed to the success of Vega launcher development and early flights.

Significant programs have been started in recent years, mainly on LOx-Methane propulsion, with the main objective of developing an engine demonstrator for the upper stage of Vega evolutions, and the setup of a dedicated test facility. This activity has been complemented by the HYPROB program, the result being the acquisition of basic research competences, engineering design skills and enabling technologies maturation up to the fully national development of the entire combustion chamber.

The synergy among industry, research centers and university competences, skills and infrastructures is a key element of such vision, so as it is an adequate position in the international context.

In the aforementioned national framework, the HYPROB Program objectives and the overall development plan have been set in relations with the institutional, industrial and scientific stakeholders.

In the mid-term perspective, the focus is put on:

- development of technology demonstrators, including intermediate breadboards;
- development of R&D activities in relevant technology areas;
- improvement of test capabilities.

At system level, the mid-term objective is to design, manufacture and test, in a relevant facility, technology demonstrators of suitable class of thrust, with the main scope of validating critical design and technology features and then to assess technology readiness level of potential solutions for future engines.[1], [2].

CIRA aims at setting up a team where the best national competencies are involved at the maximum extent, from both industrial and scientific sides. AVIO, CRAS (University of Rome – La Sapienza), and Purdue University have involved in the industrial team so far, based on their background experience.

SYSTEM DEVELOPMENTS

The System line devoted to the LOx/LCH_4 technology aims at designing, manufacturing and testing a LRE ground demonstrator, representative of a 30 KN of thrust in flight conditions.

The architecture considered for the demonstrator is a regenerative cooled thrust chamber for ground testing[1]. Regenerative cooling is one of the most widely applied cooling techniques used in liquid propellant rocket engines. It has been effective in applications with high chamber pressure and for long durations with a heat flux range 1.6 to 160 MW/m^2. In particular, in expander engines, regenerative cooling enthalpy gain is used to move turbines for pressurizing pumps.

The study logic implemented in the present project is based on the following drivers:

- to design suitable intermediate breadboards to address the most critical design solutions, such as injection and cooling.
- to make use of existing know-how and design solutions for critical items;

That approach has been defined in order to proceed step by step, from the understanding of the basic physical processes, i.e. combustion and heat transfer, and then to validate design and analysis methodologies

Figure 1 shows the adopted study logic, whilst Figure 2 the related development plan.

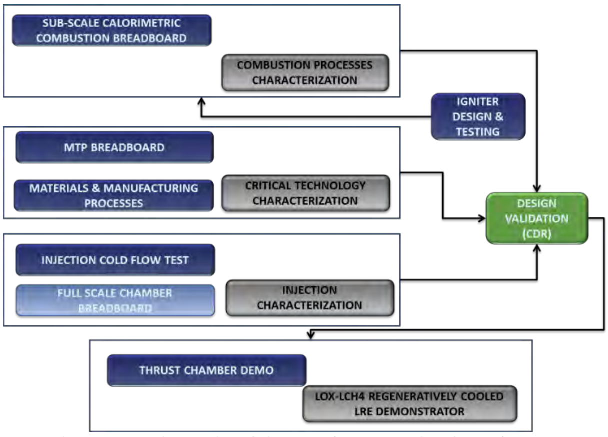


Figure 1 Study Logic of the LOx/LCH4 technology demonstrator

The Critical Design Review (CDR) is scheduled by mid 2014, after completion of the tests on the breadboards and the assessment of the design methodology. The delivery and test readiness of the demonstrator is scheduled by the end 2014.

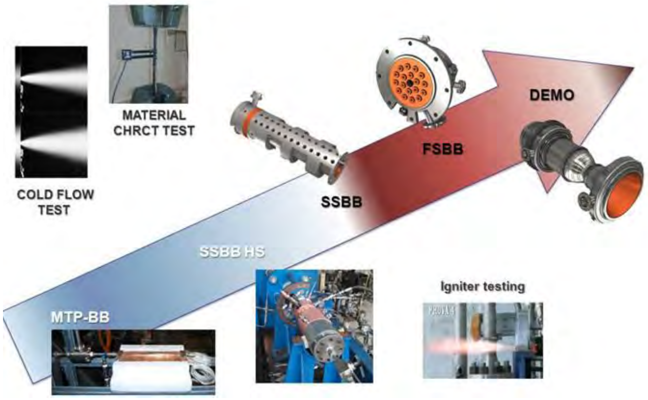


Figure 2 Project Time-Line

Next figure wraps up the current status of the project.

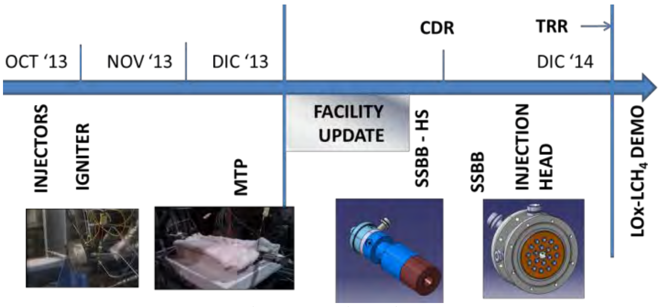


Figure 3 Project plan

HYPROB BREADBOARD

The HYPROB-BREAD project foresees, in the development of the complete system, the design of various breadboards aimed at the investigation of critical design aspects such as the supercritical behavior of the methane and the mixing and combustion processes of the propellants (gaseous methane and liquid oxygen). The above-said breadboards are described in the next paragraphs.

SUBSCALE COMBUSTION BREADBOARDS

In this section the status of the two subscale combustion chamber breadboards, namely SSBB-CC and SSBB-HS, are reported and briefly discussed.

The SSBB-CC is a segmented water-cooled single-injector calorimetric combustion chamber, which has a twofold aim: 1) to investigate the single injector thermal behaviour, in term of heat flux distributions on the chamber wall and 2) to mitigate the risk in the development of the demonstrator TCA, through the validation of critical manufacturing processes, such as vacuum brazing. The SSBB-CC shall provide data to validate the design and prediction tools used to design the demonstrator. To this end, the design of this system has been optimized to allow accurate measurements of heat flux on the wall and pressure and temperature inside the combustion chamber.

The SSBB-CC baseline concept is shown in Figure 4. More details can be found in [3], and [4].

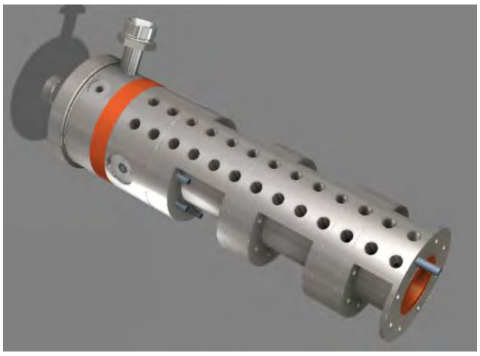


Figure 4 SSBB-CC concept

SSBB-HS is a simpler version of the SSBB-CC, without any water cooling system (HS stands for Heat Sink).

Main characteristics can be summarized as follows:

- Massive combustion chamber, without cooling system.
- The chamber is made of a Cu alloy while both throat and nozzle made of TZM.
- Equipped with thermocouples for heat flux reconstruction.
- Same injector head and igniter as SSBB
- Only short-time tests allowed (~2.5s)

The SSBB-CC manufacturing is currently on going, because of the set-up of the brazing process. First steps have been made in the definition of the thermal cycle, and now is on-going component burst testing and mechanical characteristic studies.

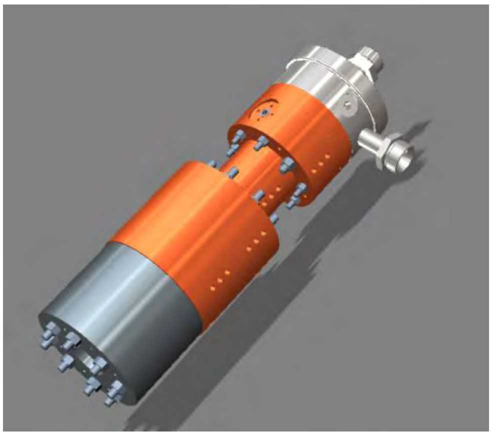


Figure 5 SSBB-HS concept

The SSBB-HS is currently Ready-To-Test.

Figure 6 SSBB-HS at Fast2 test bench

IGNITER

In order to fulfill SSBBs (CC-HS) system needs, the igniter has to be at the same time reliable, simple and cheap, customized on the BBs thrust chamber but easy to adapt for the installation on other small-scale chambers. Keep these requirements in mind a spark-torch igniter has been selected and developed. [4]

The spark torch ignition systems use two propellants (oxygen and a fuel) that are mixed in the igniter combustion chamber and ignited by a spark plug. In this way the igniter produces a torch flame that in turn is used to ignite the main combustion chamber of a rocket engine.

A large benefit of spark torch igniters is their ability to provide restartability to a space engine.

A spark torch system consists of three main parts:

- The igniter feed system in which the fuel and oxygen are stored under high pressure;
- The igniter, a small combustion chamber in which the igniter gasses are ignited by a spark plug;
- The exciter that delivers the energy to the igniter spark plug.

The architecture of the igniter is reported in the following Figure 7. The igniter is mainly made up of two main parts, the igniter head (1) and the torch outlet (2) with flanged interfaces sealed by metal O-rings. The fuel and oxidizer are injected via orifices..

Figure 7 Igniter schematics

The igniter is not actively cooled and is made by Molybdenum alloy. In this way high temperatures could be managed by radiation cooling for the necessary firing time, the stem part could be couplet with materials such as steel of Inconel as well as copper alloys. The inlets of CH4 and Oxygen are equipped with PT sensors; moreover a pressure sensor is installed in main chamber in order to monitor chamber pressure.

Acceptance tests have been performed in AVIO Fast2 facility (qualification box shown in Figure 9).

Stable combustion occurs, reaching a firing time up to 2.2s, with a 1.5s in steady state, showing no plastic deformation.

As shown in Figure 8, the igniter has been installed over an Inconel disk to simulate the installation on the combustion chamber.

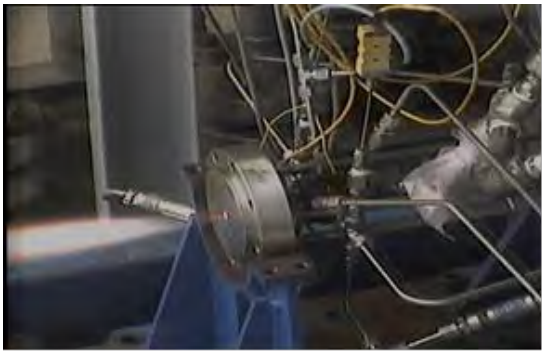


Figure 8 Igniter firing test

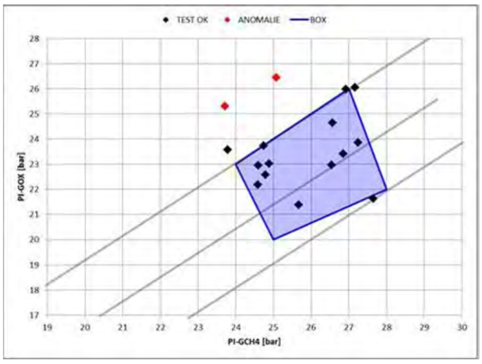


Figure 9 Qualification box

Temperature measurement on the body of the igniter and CFD analysis confirm that the flow is not symmetric, due to the injection system selected.
The temperature results are in line with FEM results, too.

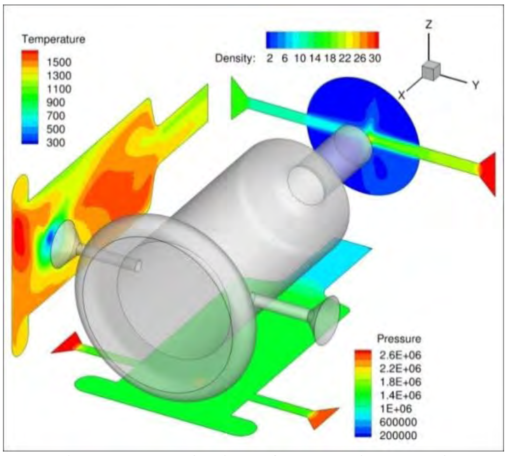


Figure 10 Sample view of CFD analyses result

Due to the peculiar assembly of the ignite (i.e. orthogonal to the SSBBs wall), in order to evaluate the impact on such heat flux on the breadboard structure several specific tests have been performed (Figure 11).

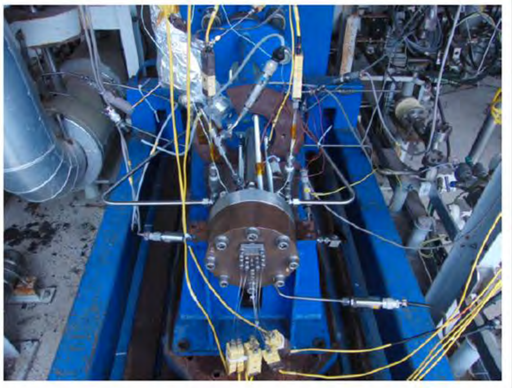


Figure 11 Equipment for the igniter thermal impact evaluation mounted on FAST2

INJECTOR

Regarding the injector design, a conventional shear coaxial injector has been designed and manufactured, as shown in Figure 12.

Figure 12 Shear coaxial injector

Eight type of injectors have been tested (performing a sort of scattering analysis on main design parameter) and the flow visualization has been obtained with a simple 8 MP camera. It has to be said that the results in terms of pressure drop (and so liquid velocity) are in line with design values, for what concerns the results with orifice, pressure drop are higher than expected and so lower liquid jet velocity are achieved. These results will be deeper investigated in future.

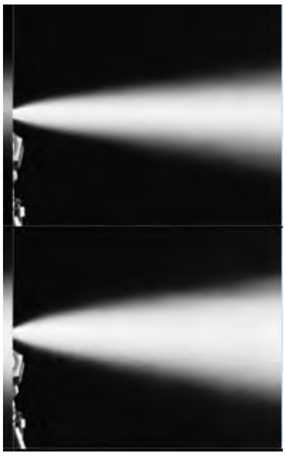


Figure 13 Jet spray from cold flow testing

METHANE THERMAL PROPERTIES (MTP) BREADBOARD

The MTP Breadboard is a test article conceived for the study of the thermal characteristics of the methane as a coolant, both for design validation and collect experimental data to share in the scientific community. Its concept is based on an electrical heating of a conductive material that transfers heat fluxes, similar to those experienced by methane in the regenerative cooling chamber of a rocket, to a channel (dimensions comparable with the HyProb demonstrator) in which methane flows at high pressure. The concept has to be correctly shaped in order to „drive" heat to the channel wall.

An important consideration has to be made about the thermal insulation of the model. On one side it is important in order to avoid losses of heat power throughout lateral conductor walls, on the other is important because the adiabatic condition applied at the channel wall is similar to the symmetry condition of the channel walls in a real rocket (Figure 15).

Figure 14 MTPB schematics

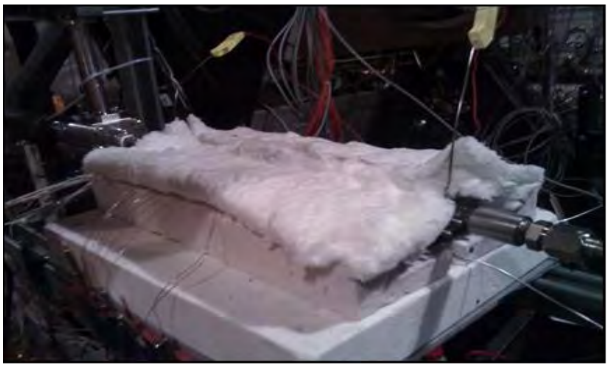


Figure 15 Insulated MTPB

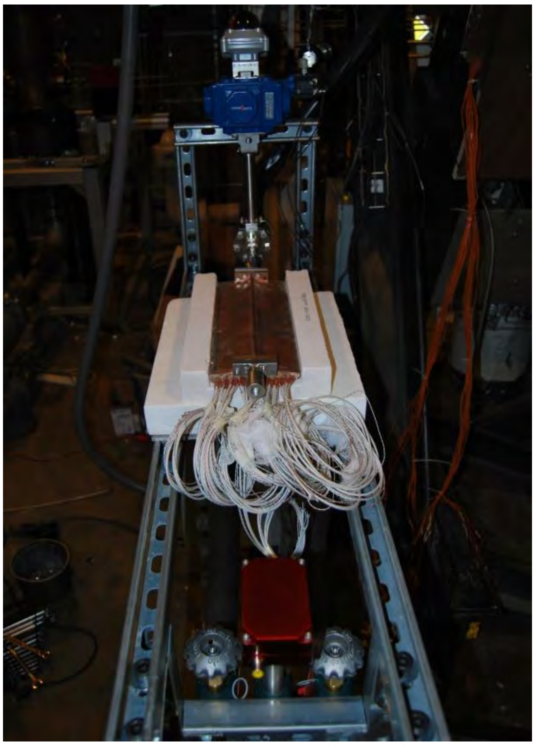


Figure 16 MTPB at Zucrow Lab Facility (Purdue University)

MTPB is equipped with 4 stations of three thermocouples each and the numbering is highlighted in Figure 17.

The fluidic lines are equipped at inlet and outlet with temperature and pressure measurements (see Figure 18, where also cartridges thermocouples map is shown). MTP is equipped also with 12 on-body thermocouples.

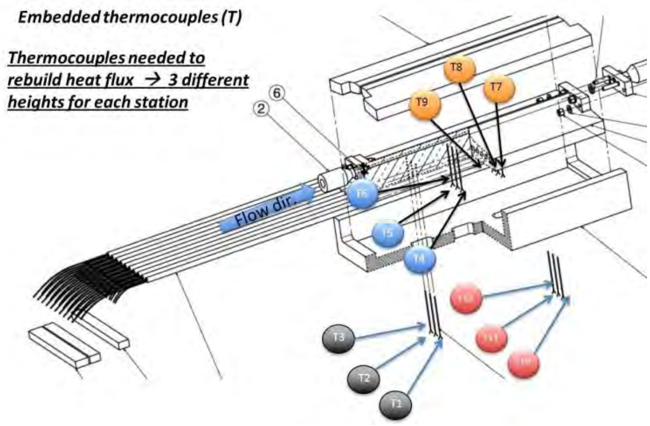


Figure 17 MTP PID – embedded thermocouples map

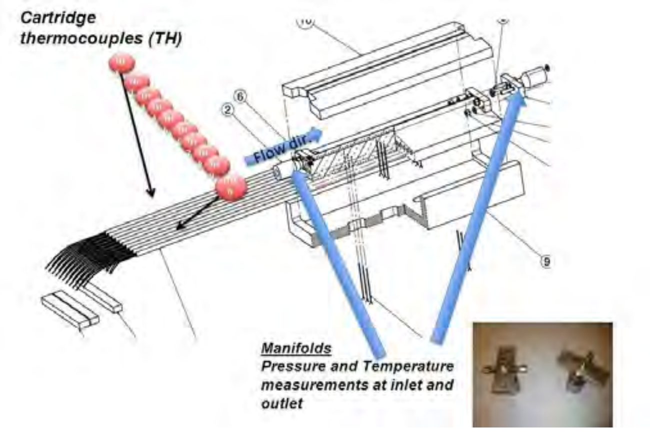


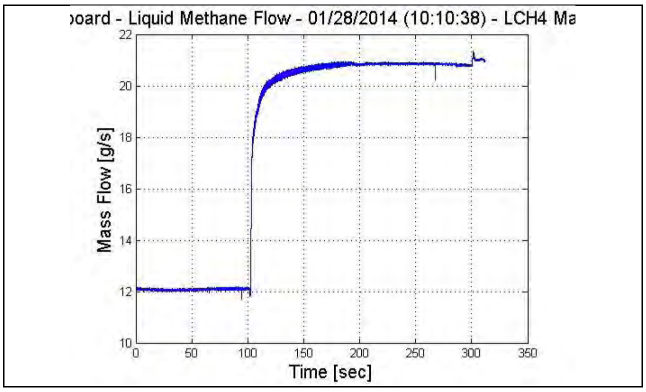
Figure 18 MTP PID – Cartridges thermocouples map

MTP breadboard has been tested at Zucrow Laboratories of the University of Purdue in West Lafayette (USA).

Test ID	Pex (bar)	mdot (g/s)	POWER IMPRESSED (kW)
1	150	20	12
2	150	25	12
3	150	15	12
4	120	15	12
5	120	20	12
6	120	25	12
7	100	15	12
8	100	20	12
9	100	25	12
10	80	20	12
11	80	15	12
12	80	25	12
13	80	10	12
14	80	15	0
15	80	25	0
16	60	20	12
17	80	20	20

Table 1 MTPB Qualification Test Matrix

Next Figure shows the a sample experimental results (case ID#10).



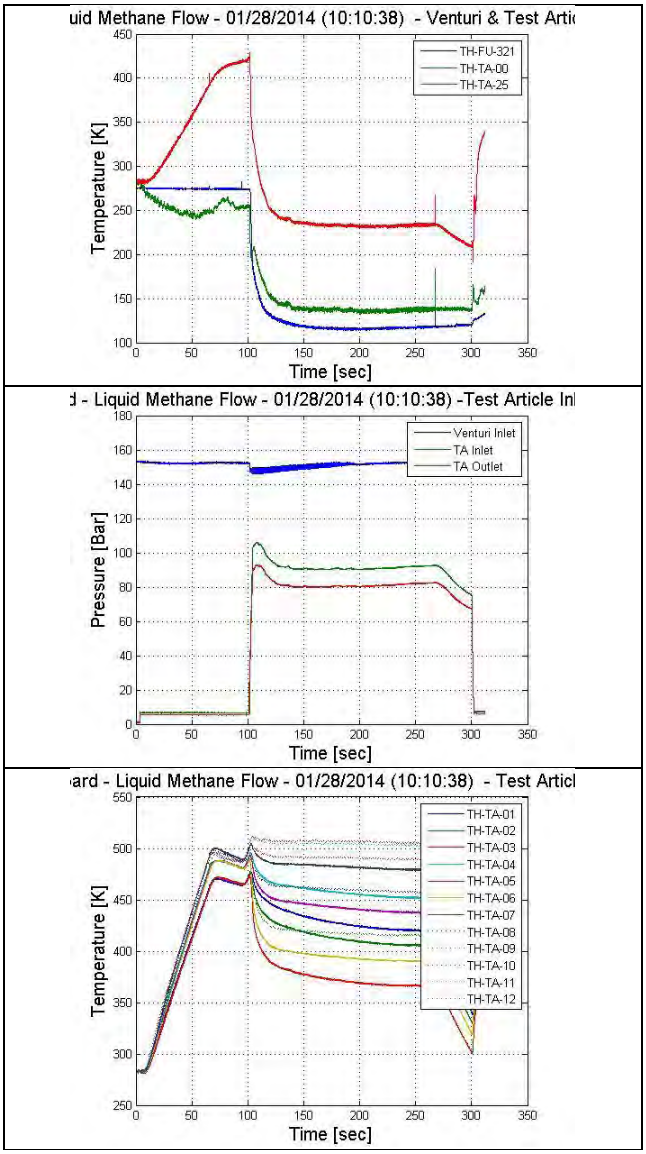


Figure 19: Sample experimental results (ID#10)

Experimental results have been compared with numerical results from a predictive tool developed by means of Ecosimpro [7] showing a good agreement.

Model is reported in Figure 20, while next figures show a comparison between experimental and numerical data (case ID#8) in terms of pressure (Figure 21) and temperature (Figure 22).

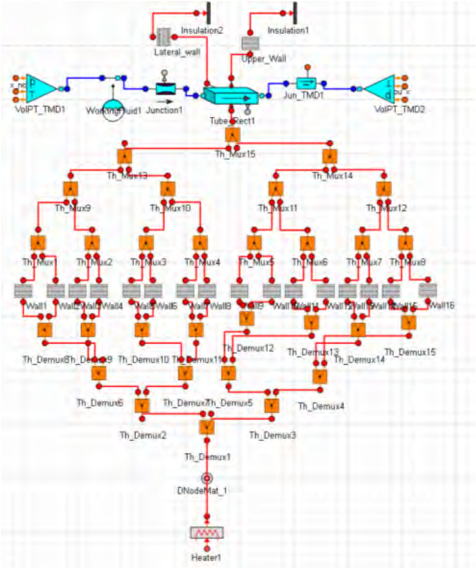


Figure 20 Ecosimpro model for MTPB

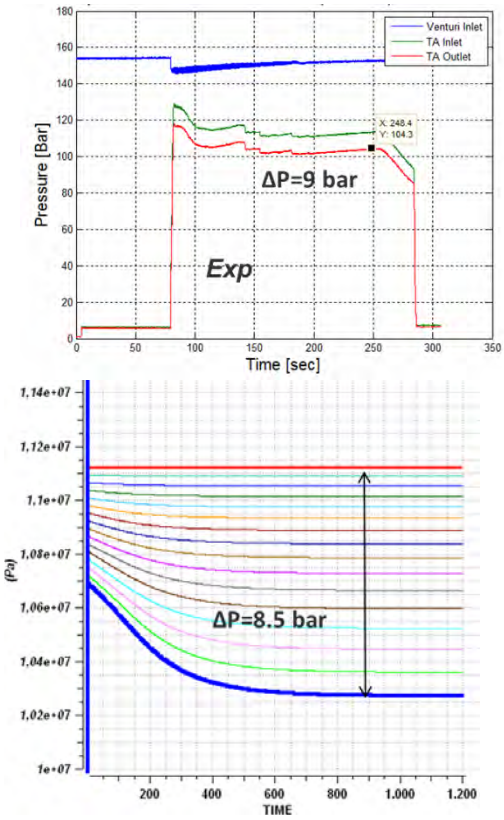


Figure 21 Pressure drop comparison (ID#8)

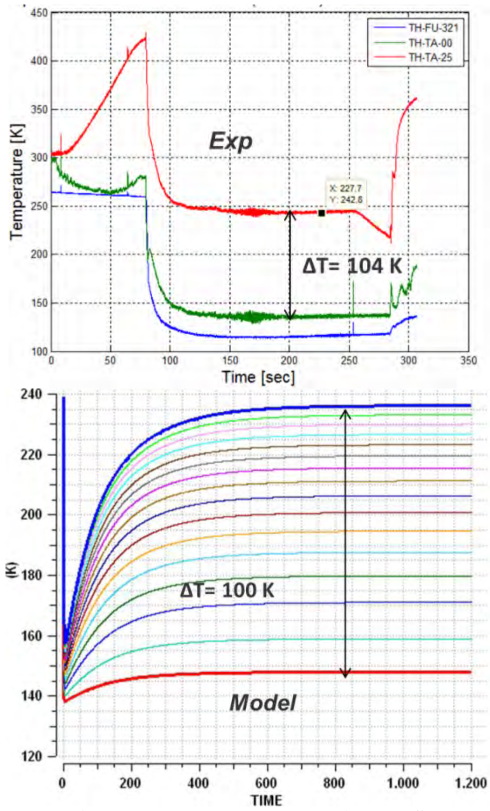


Figure 22 Temperatures comparison (ID#8)

DEMONSTRATOR

The architecture considered for the demonstrator is a regenerative cooled thrust chamber for ground testing. The main elements are the igniter, the injector head and the combustion chamber. A counter-flow architecture will be considered for the chamber cooling system, where the coolant (LCH4) is injected liquid into the fuel manifold and enters the cooling jacket counter flow with respect to the combustion gases (Figure 23). After heating it is injected directly in the fuel dome and then from the injector in the chamber where atomizes, mixes and burns with liquid oxygen. A demonstrator assembly view is reported in Figure 24.

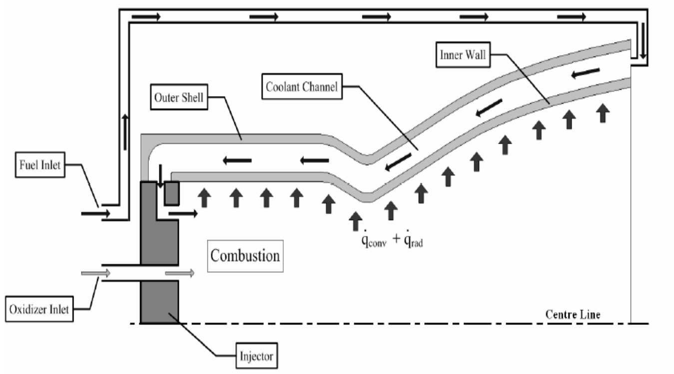


Figure 23 Counter-flow architecture of the cooling jacket

The combustion chamber has a cylindrical shape whose radius is 0.06m and it is long 0.192m. The nozzle‟s throat radius is 0.03m and the expansion ratio is 9. The overall length from the injection plate to the nozzle‟s exit is 0.44m.
A configuration with 96 channels has been selected, choosing a constant value for the channel rib width while a variable value of the rib height has been adopted in order to optimize the cooling performances in the nozzle, throat and chamber zone. The channel is defined by a liner, made up by a copper alloy, in the bottom part, and by a close-out, made up by Inconel, in its upper part.

Figure 24: Demonstrator view

DESIGN VERIFICATION

INJECTION HEAD

The injector head has 18 injectors with central igniter interface; the injectors are placed on two coronas with a 30° symmetry. The injector plate is made by Cu alloy and the main body is in Inconel, back the injector plate there is a back plate that ensures cooling of the injector plate.
Propellants are injected in the head throughout two manifolds, respectively, for LOX and GCH4 coming from cooling jacket. GCH4 manifold is connected to a distribution collector that injects methane in the CH4 collecting dome after a distribution grid.

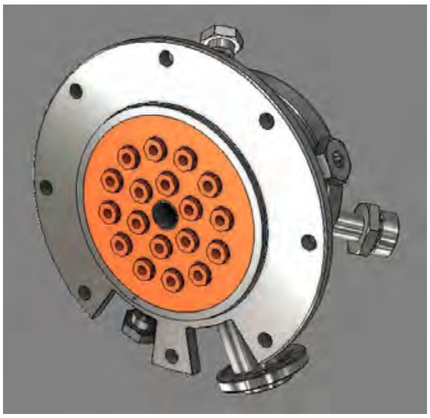


Figure 25 Demonstrator injector head

CFD and thermo-structural analyses have been performed on the injector head in order to evaluate thermal and mechanical loads on the injector plate and correctly design the active cooling of the back plate.

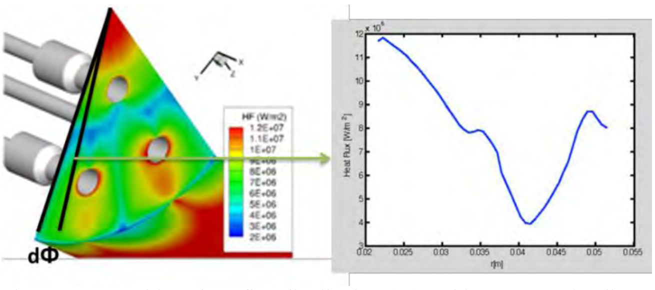


Figure 26.CFD driven heat flux distribution (sx) and its conservative linear extrapolation (dx)

The selection of heat flux has been represented in Figure 26: the distribution chosen (as reported on the left) is obtained as the most conservative possible from CFD results. On the right, the effective distribution is reported

Results are shown in the next Figure 27, Figure 28, Figure 29, and Figure 30.

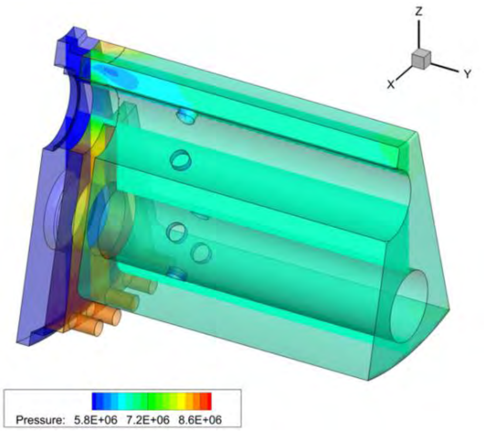


Figure 27 Injector Head: the reduced volume considered for CFD

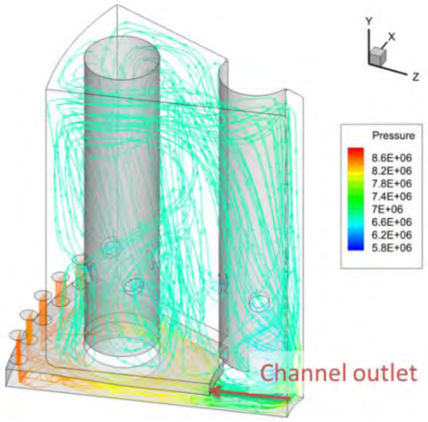


Figure 28 Stream-traces in computational-domain coloured by pressure value

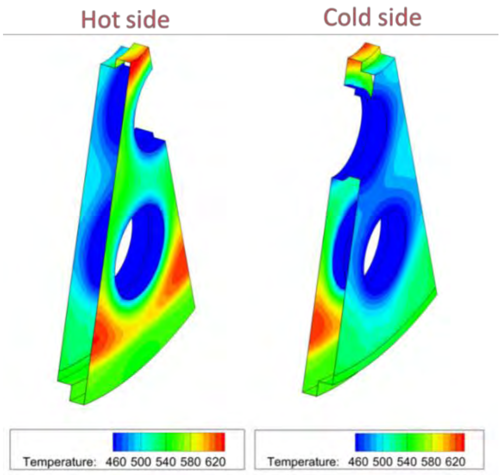


Figure 29 Firing-plate (solid part) – temperature contour for hot side and cold side

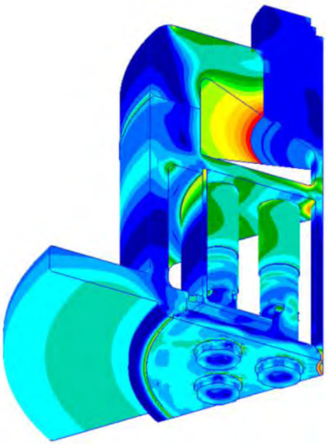


Figure 30 Injection head: Von Mises Stress

COMBUSTION CHAMBER

The combustion chamber has been designed according standard procedures in rocketry, it is composed by a cylindrical part and a conical nozzle and is reported in Figure 25.

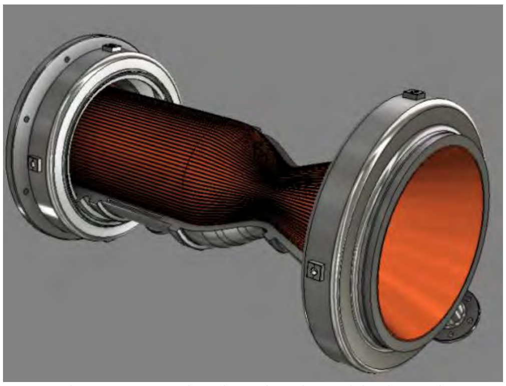


Figure 31 Combustion chamber with channels

Also for the chamber, CFD and thermo-structural analyses have been performed in order to verify the adopted design.

The computational domains adopted are displayed in Figure 32. A 60° periodic edge was considered for the reactive simulations and half of the cooling channel, made by two solid parts between which there is a fluid zone, was modeled in order to decrease the computational effort [6].

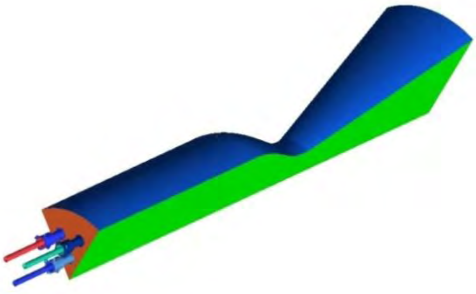


Figure 32 Computational domain

A sketch of the grid mesh is given in Figure 33.

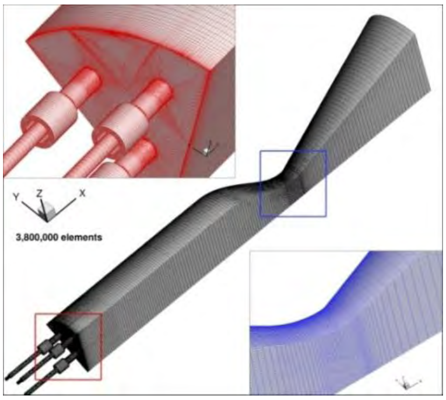


Figure 33 Sketch of the grid mesh

CFD results, adopting the grid mesh shown in Figure 33, are given in next Figure 34.

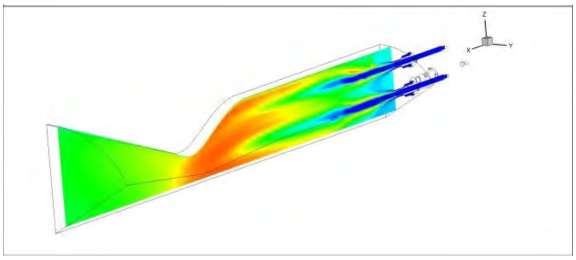


Figure 34 Static Temperature Contour Plot

For the cooling channel analysis, a “weak thermal coupling” between hot gas side and cold side of the chamber has been adopted. The input heat flux is given by the CFD on the “hot side” (Figure 34), final loop is reached after a series of thermal values updating.

Results are given in Figure 35 and Figure 36.

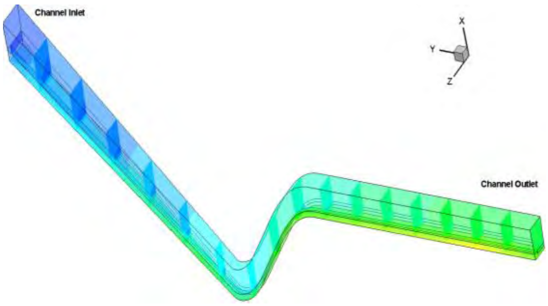


Figure 35 Temperature distribution on the cooling channel

The final configuration has been verified under the thermo-structural point of view, by considering a 2-D FEM models (3-D model shall be finalized soon).

In particular, the most stressed section have been considered, such as the throat one, and the thermal most stressed section in the cylindrical part.

CFD simulations allowed to determine the input information for the FEM thermo-structural simulations in terms of fluid bulk temperature, fluid pressure and average heat transfer coefficients.

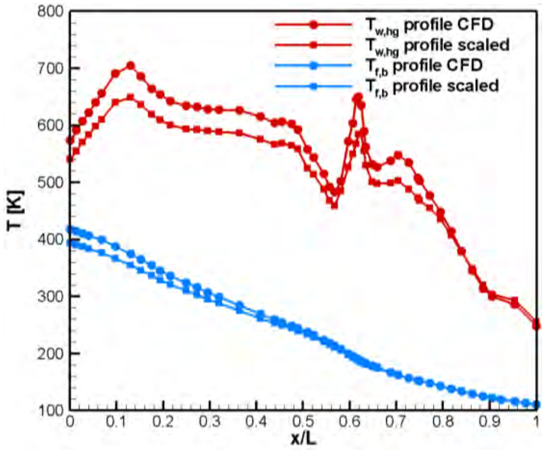


Figure 36 Temperature axial profiles: wall temperature profile of the liner (in red) and fluid bulk temperature (in blue) .

Figure 37 Equivalent plastic strain contour plot after 3 cycles, throat section

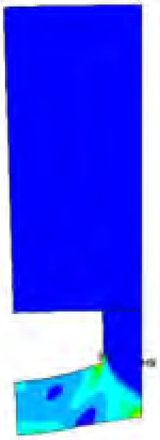


Figure 38 Equivalent plastic strain contour plot after 5 cycles, chamber section

Fatigue life has been calculated by adopting a safety factor of 4 on the material and after-brazed properties have been taken into account.

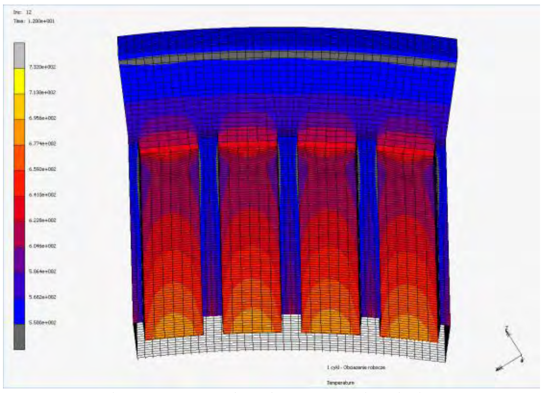


Figure 39 3D Chamber FEM Simulation

COMBUSTION STABILITY

Combustion instabilities are a result of the resonant interaction between two or more physical mechanism. A driving process generates the perturbation of the flow, a feedback process couples this perturbation to the driving mechanism and produces the resonant interaction which may lead to oscillatory combustion. For the development of the system stability analysis are crucial in order to ensure that the designed system is stable [8]. In the framework of the HYPROB BREAD project the stability analysis have been performed by means of ROCCID2000X® code [9]. Both low frequency and high frequency analyses have been performed for all the experimental operative box points.

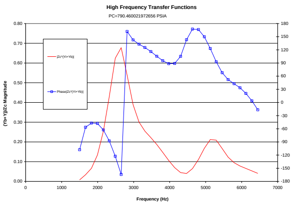


Figure 40 1L High Frequency Transfer Functions at nominal operating condition t

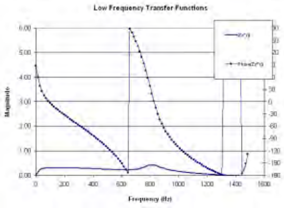


Figure 41 Low frequency transfer function at nominal condition

FAST2 Test Bench Facility

According to the synergic approach adopted in the program, the FAST2 facility, running at the Colleferro (Rome) premises of AVIO, has been selected as the main experimental asset to carry out tests at system level on the demonstrators and associated combustion breadboards. The facility has been developed in the FAST2 program, funded by ASI some years ago in support to space transportation technologies and is presently co-owned by ASI and AVIO.
The main characteristics of the present configuration of the facility are hereinafter described:

- LOx feeding line up to 10 kg/s @ 200 bar (max pressure tank)
- GCH4 feeding line up to 2 kg/s @ 200 bar (max pressure tank)
- Cooling water feeding line up to 20 kg/s @ 140 bar
- Test cell for combustor testing
- Command and control capability provided by two redundant units
- Data acquisition unit with 82 channels

The facility has already been extensively used in recent programs to test thrust chambers representative of 30 kN thrust (vacuum) class.
A major update of the facility will be carried out in late 2014 to provide testing with liquid methane, according to the requirements of the HYPROB program.

Conclusions

In the present paper the present design status of the HYPROB-BREAD Project, along as the development plan, has been presented and discussed.
Following an incremental approach in terms of system complexity, two different breadboards has been designed and will be used to test critical sub-system components of the demonstrator and to validate critical design aspects and analysis tools.
Verifications on demonstrator configurations by means of FEM, CFD and engineering methodologies have been briefly discussed.
Moreover first manufacturing and testing outcomes have been presented

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Acronyms

BB	Breadboard
CC	Combustion Chamber /Cooling Channel
CFD	Computational Fluid Dynamic
EOS	Equation Of State
FEM	Finite Element Method
GCH4	Gaseous Methane
HS	Heat Sink
IH	Injection Head
LCH4	Liquid Methane
LOX	Liquid Oxygen
LRE	Liquid Rocket Engines
MTP	Methane Thermal Properties
NS	Nozzle Convergent Part start
TC	Thrust chamber
TCA	Thrust chamber assembly
TS	Thermo structural

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