UNIVERSITÀ DEGLI STUDI DI NAPOLI
FEDERICO II

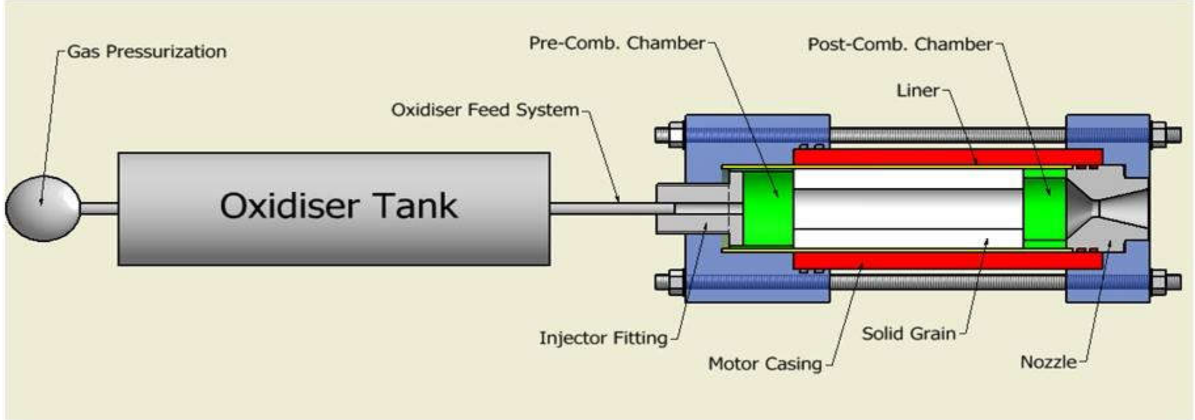
Dipartimento di Ingegneria Industriale –
Sezione Aerospaziale

Recenti sviluppi su endoreattori a propellenti ibridi

Introduction

Hybrid Rockets Propulsion – Introduction

Hybrid propellant rockets are chemical engines in which fuel and oxidizer are in different physical state



Advantages

- Throttling and restart capability
- Safety and reliability
- Economic sustainability
- Environmental sustainability
- Wide range of potential application:
  - Sub-orbital flight vehicles;
  - Launch vehicle lower and upper stages
  - Nano- and microsatellite launch vehicles

Disadvantages

- Low regression rate of the solid fuel (which means low thrust level)
- Complexity of the thermo-fluid dynamic behaviour in hybrid rockets (combustion instabilities, mixture ratio shifting, effect of operating parameters)

Introduction

Hybrid Rockets Propulsion – Historical perspective

- 1933: First hybrid rocket designed by Mikhail Tikhonravov and Sergei Korolev and their group and launched reaching 1500 m of altitude
- 1940s: First effort in hybrid rocket development in U.S., at the Pacific Rocket Society and General Electric
- 1960s: Flight test programs initiated both in the U.S. and in Europe; development of Marxmann's theory about hybrid rocket combustion process; attempts of scaling up
  - → Several successful firing tests performed, but the results highlighted an excessively low regression rate and volumetric fuel loading efficiency
- 1970s: interest in hybrid propulsion revived because of safety concerns about solid propellant segments of Space Shuttle booster

Tikhonravov and his Gird-09

Introduction

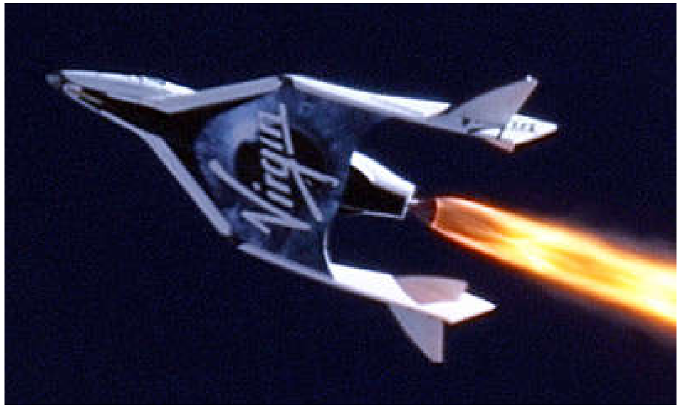
Hybrid Rockets Propulsion – Historical perspective

- 1980s-1990s: several development programs: AMROC (American Rocket Society) LOX/HTPB propelled launch vehicle and Hybrid Propulsion Demonstration Program (HPDP) for a 1100 kN thrust test bed


AMROC's SET-1 launch vehicle


HPDP engine firing at 1100 kN thrust

- 2000s: Successful flight of the reusable manned spaceplane SpaceShipOne; development of high-regression rate paraffin-based fuels at Standford University, which extend the potentiality for the application of hybrid rockets


Virgin Galactic SpaceShipOne

Introduction

Main challenges in Hybrid Rockets development

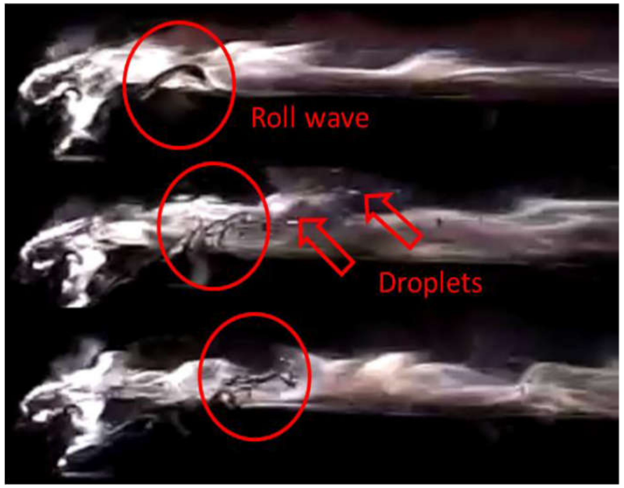
- **Low regression rate of classical polymeric fuels**
- Several proposed strategies leading to an increase of the system complexity without producing major improvements of the engine overall performance

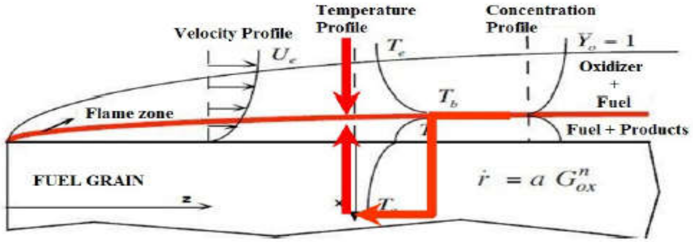
⇨ **Interest in the utilization of liquefying fuels, e.g. paraffin-based fuels**

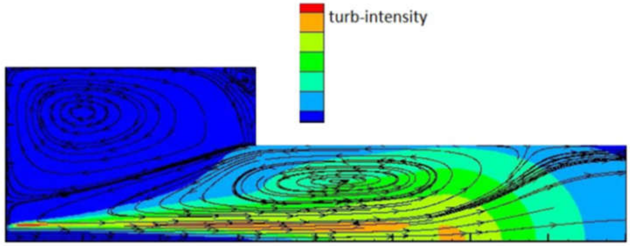
- Enhancement of regression rate due to the formation of a low-viscosity unstable melt layer on the burning surface and the consequent mechanical entrainment of liquid droplets

- **Complexity of the thermo-fluid dynamic behaviour in hybrid rockets**
- Behaviour can significantly change depending on the specific fuel formulation, the manufacturing process and the motor operating conditions

⇨ **Need for numerical modelling** to support the design and improve the performance prediction capabilities

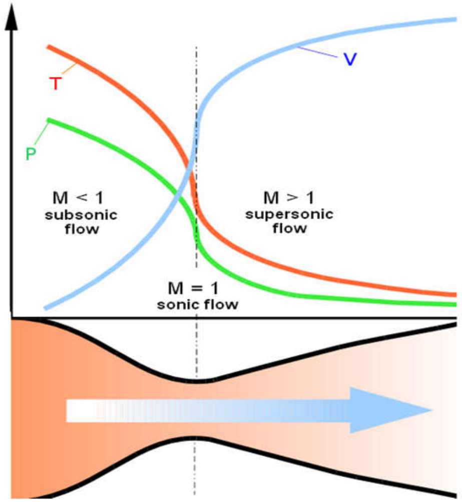




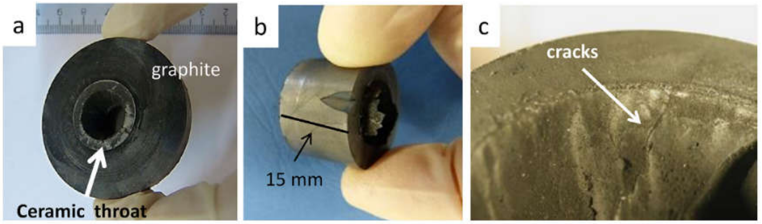


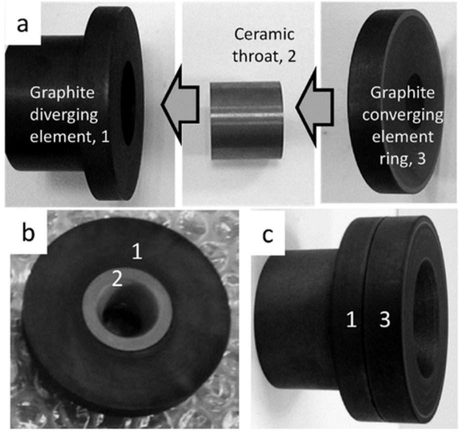
Introduction

Rocket nozzle materials in the harsh combusting environment



- The nozzle is subjected to the **highest heat fluxes and shear stresses in a chemically aggressive environment**
  - → Proper selection of suitable rocket nozzle materials
- Classical materials are subjected to **thermochemical erosion**
  - → Enlargement of the nozzle throat section and consequent decrease of rocket performances
- **Ultra-High Temperature Ceramics (UHTC)**
  - → Good erosion resistance
  - → Poor thermal shock resistance





- Recent interest in **Ultra-High Temperature Ceramic Matrix Composites (UHTCMC)** based on C or SiC fibres in UHTC matrix
- **Experimental testing and CFD simulations are needed** to improve the design and the current performance prediction capabilities

R.Savino, G.Festa, A. Cecere and L. P. a. D.Sciti., "Experimental set up for characterization of carbide-based materials in propulsion environment.," Journal of the European Ceramic Society, pp. 1715-1723, 2015.

Aerospace Propulsion Laboratory

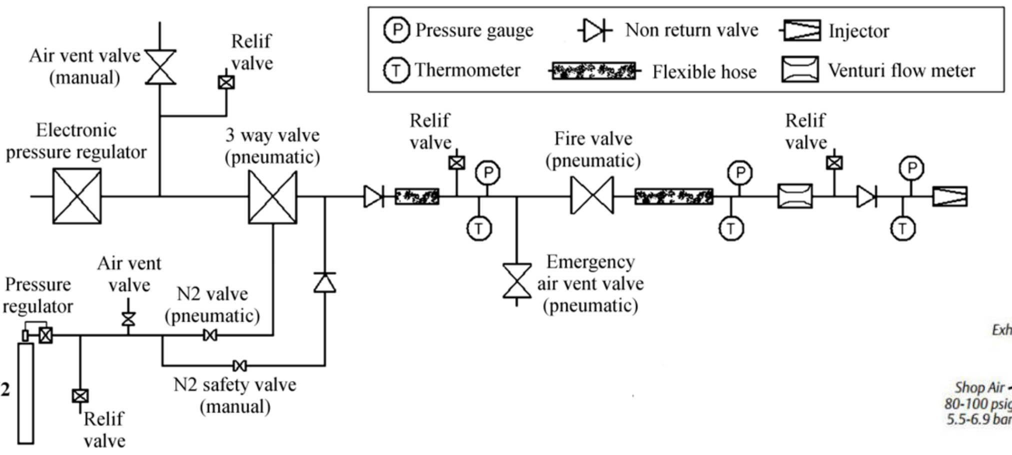
The experimental activities are carried out at the Aerospace Propulsion Laboratory located in the military airport "F. Baracca" in Grazzanise (CE)

Main purpose: testing of hybrid rockets

- Evaluation of **propellant performance** and **combustion stability**
- **Testing of sub-components** (nozzles, air intakes, catalytic devices, burners, ignition and cooling systems)
- **Testing of materials** for application in combustion environments

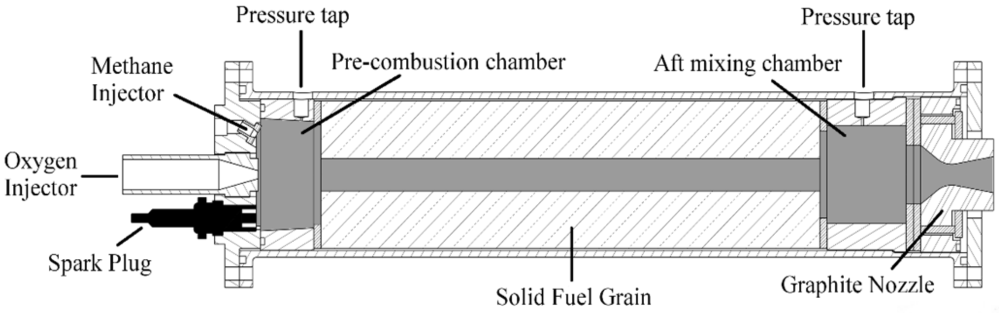
Main control and measurement techniques

- **Oxidizer mass flow rate**: regulated by a TESCOM electronically controlled pressure valve; measured by a Venturi-meter
- **Chamber pressure:** capacitive pressure transducers
- **Thrust:** load cells on the test bench
- **The analog signals generated by the sensors are processed and recorded by a NI PXI Express standard system**

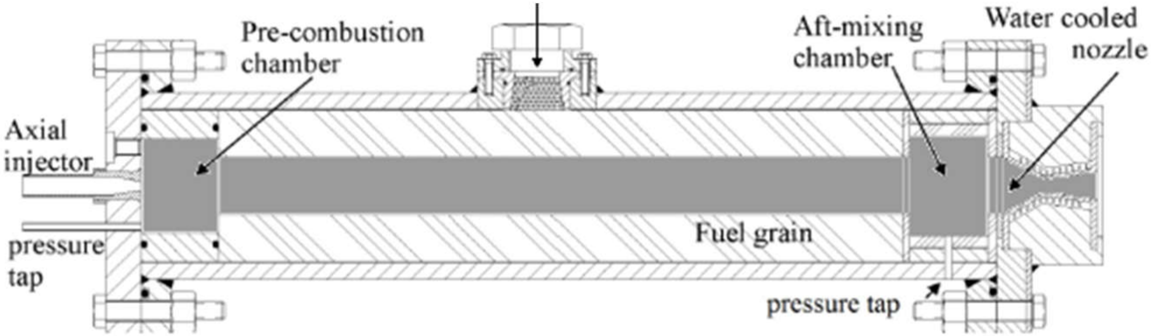


Experimental facility

Hybrid Rocket Engines



200 N-class hybrid rocket



1 kN-class hybrid rocket

- Axisymmetric combustion chamber
- Conical axial injector
- Upstream and downstream of the solid grain a dump plenum and an aft-mixing chamber are set up, respectively
- A graphite converging-diverging is employed in the 200 N-class engine
- Water-cooled, converging-diverging nozzle with 16 mm throat diameter and 2.44 area ratio was employed in the 1 kN-class engine

Motor dimensions for the definition of the computational domain

Engine class	Prechamber diameter	Prechamber length	Fuel grain length	Post-chamber diameter	Post-chamber length
200 N	46 mm	25 mm	220;240 mm	40 mm	58;38 mm
1 kN	80 mm	70 mm	430 mm	80 mm	200 mm

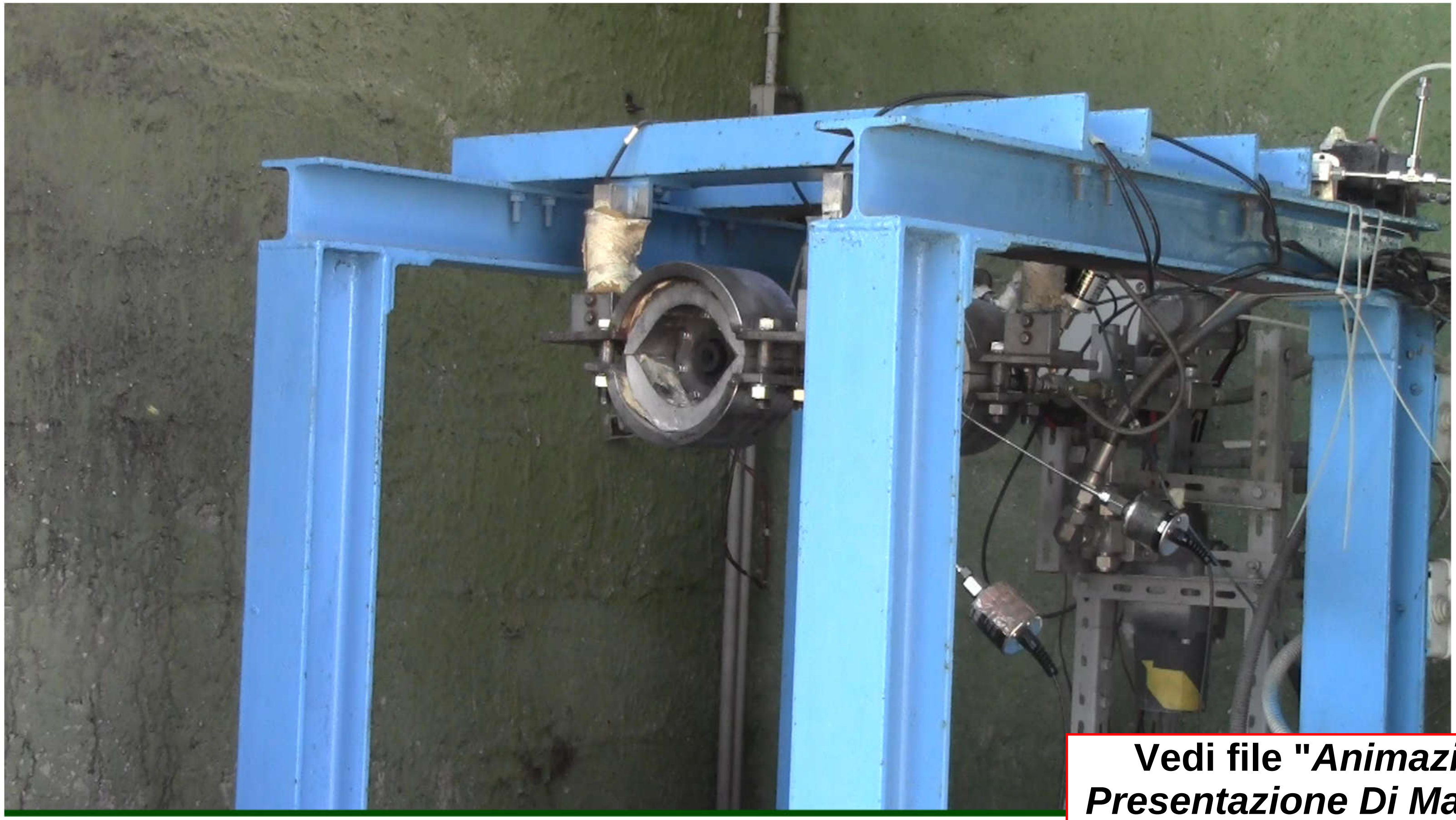
Experimental facility

Hybrid Rocket Engines

200 N-class hybrid rocket

1 kN-class hybrid rocket

Firing tests



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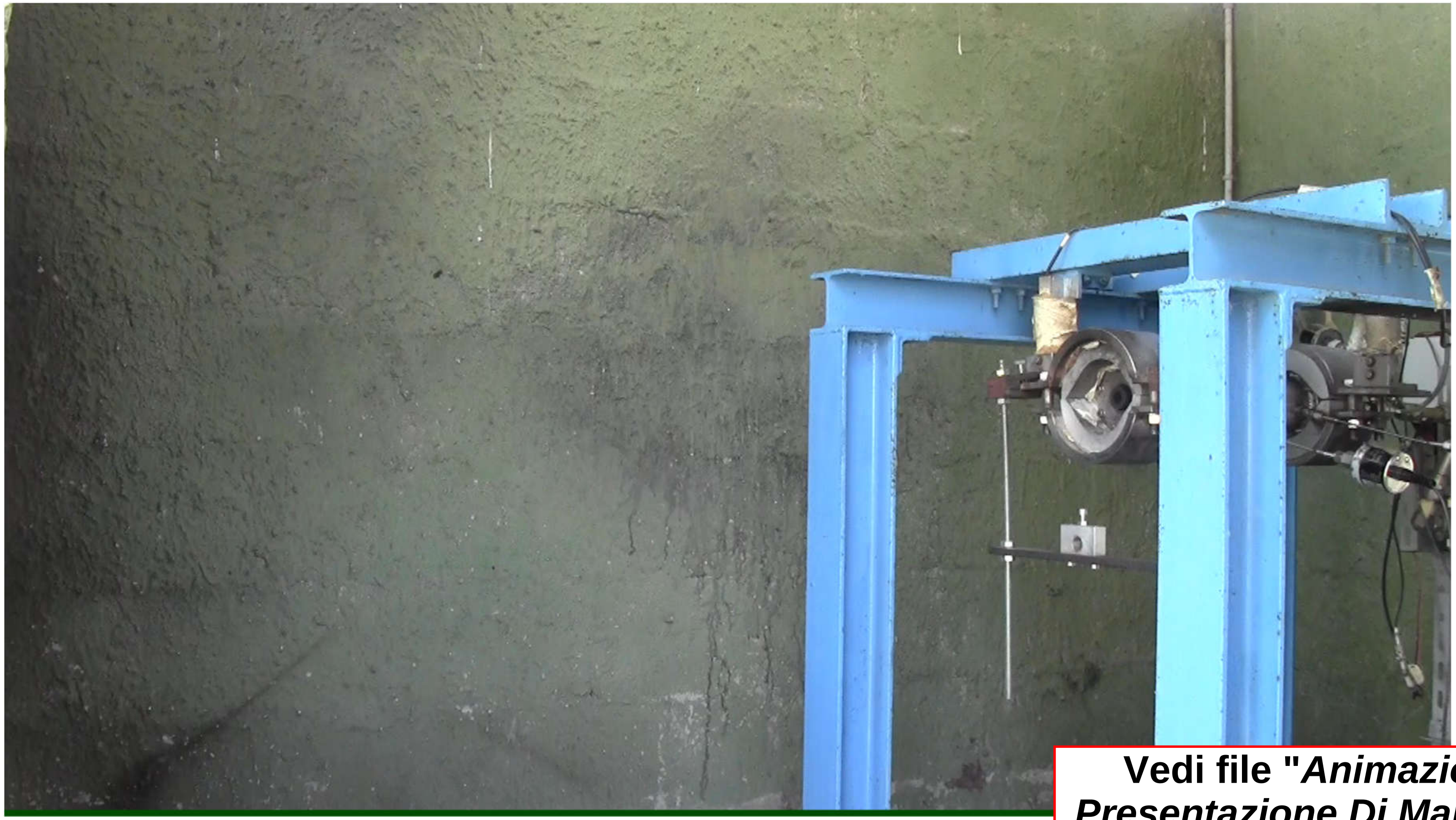
Firing tests



Vedi file "*Animazioni Presentazione Di Martino*", pag. 2

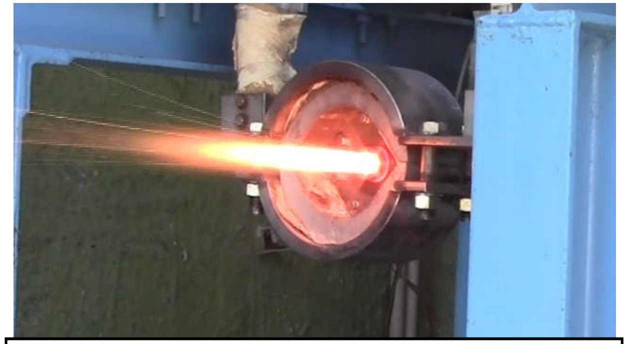
Firing tests

Vedi file "*Animazioni Presentazione Di Martino*", pag. 3

Firing tests with polymeric fuels

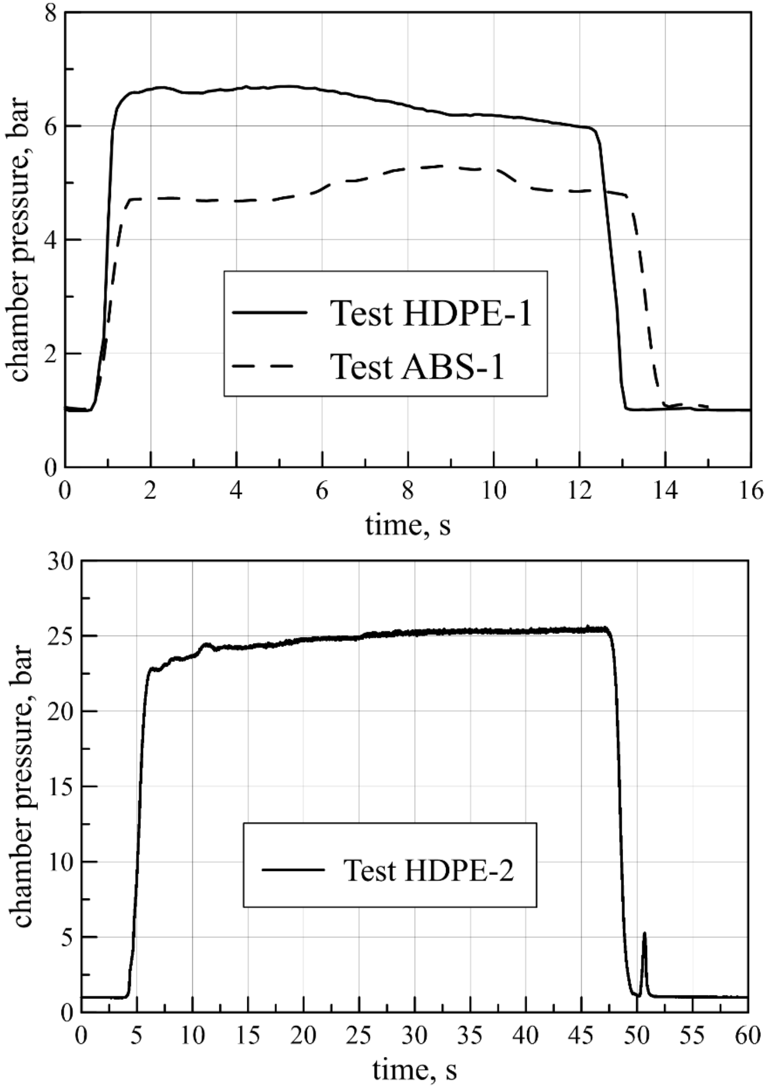
Firing test operating conditions

Parameter	Test HDPE-1	Test ABS-1	Test HDPE-2
Engine class	200 N	200 N	1 kN
Fuel	HDPE	ABS	HDPE
Time-averaged oxygen mass flow rate, g/s	27	27.5	208
Grain initial port diameter, mm	15	15	25
Grain length, mm	220	240	570
Test time, s	11.4	12	42.6
Average oxidizer mass flux, kg/m^2s	91.34	69.78	84.75
Time-space averaged regression rate, mm/s	0.39	0.61	0.73
Time-averaged aft-chamber pressure, atm	6.41	4.78	25.0
Time-averaged overall mixture ratio	5.63	2.62	3.02
Postburn space-averaged port diameter, mm	23.8	29.7	86.8
Time-space-averaged port diameter, mm	19.4	22.4	55.9
Nozzle throat diameter, mm	9.6	12	16



Rocket exhaust plume during Test HDPE-1

Hybrid rocket exhaust nozzle after the engine burn-out for Test HDPE-1



Operating pressure vs time

HYPROB-new research project

The objectives of the HYPROB-new research project by CIRA is the design and test of a 1 kN-class paraffin-fuelled hybrid rocket demonstrator

University of Naples is responsible of small-scale tests for the paraffin characterization, support in the design and testing of the 1 kN Demo

Test ID	Effective oxygen mass flow rate, g/s	Effective burning time, s	Average oxidizer mass flux, kg/m^{2}s	Average regression rate, mm/s	Average chamber pressure, bar	Average thrust, N
0a	19.1	4.2	51.88	1.58	4.5	39
0b	26	5.6	54.50	1.74	7.0	63
1	16	3.5	48.38	1.63	4.9	39
2	25	4.7	50.52	2.15	8.0	73
3	29	4.8	66.26	1.80	8.5	82
4	39	4.9	77.11	2.08	11.5	118
5	29	5.4	59.22	1.79	8.0	80
6	38	5.6	67.83	2.04	11.2	114
7	38	5.1	56.28	1.83	11.1	112
8	44	4	68.88	2.11	13.2	136
9	50.2	4	75.90	2.28	15.7	162
10	55.5	3.8	83.75	2.41	16.9	178
11	60	3.9	85.23	2.6	18.8	200
12	59.5	4.5	96.76	2.73	18.4	201
1W	42	5.3	72.58	2.29	12.9	135
2W	60.5	4.1	105.22	2.96	19.1	209
*****	29	3.5	72.93	0.94	8.0	74

Rocket exhaust plume at different oxygen mass flow rate

Test 1, $\dot{m}_{ox} = 16\ g/s$

Test 12, $\dot{m}_{ox} = 59.5\ g/s$

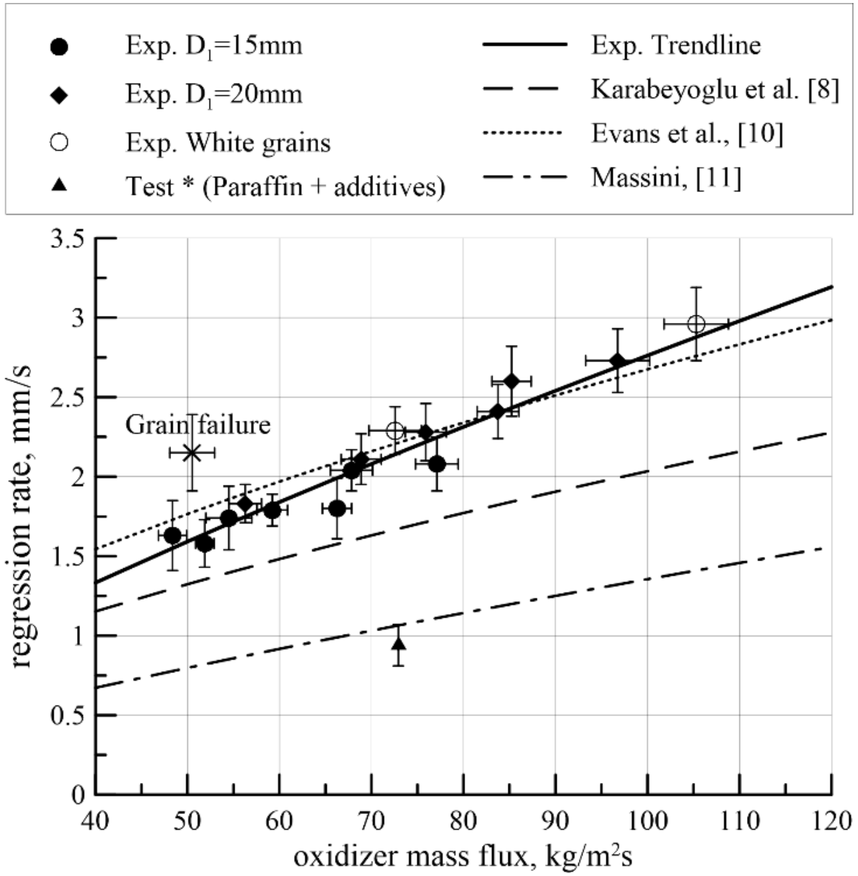
HYPROB-new research project

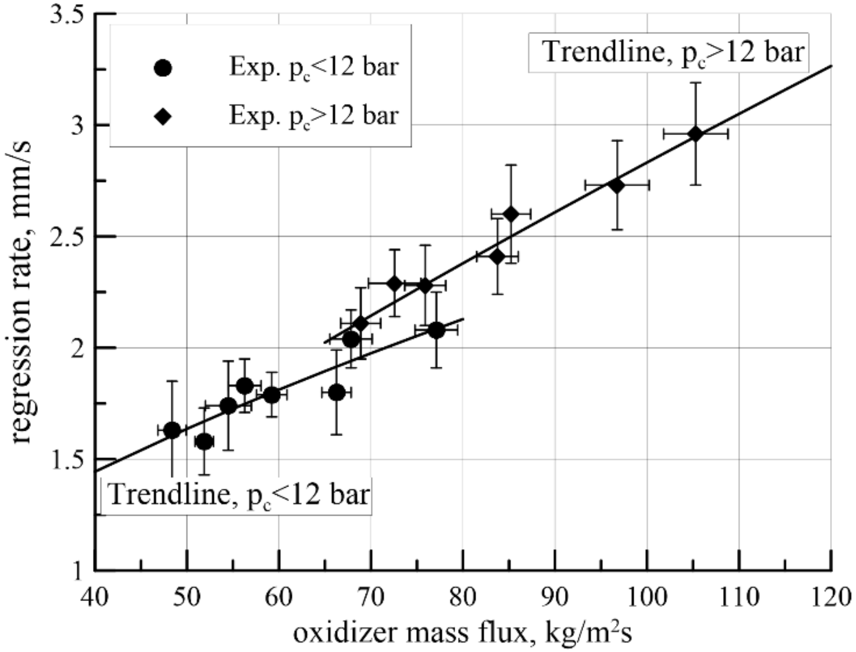
Regression Rate Laws

1) $\dot{r} = 0.071\, G_{ox}^{0.795}$

2) $\begin{cases} \dot{r} = 0.183\, G_{ox}^{0.56}, & p_c < 12\ bar \\ \dot{r} = 0.078\, G_{ox}^{0.78}, & p_c > 12\ bar \end{cases}$

3) $\dot{r} = 0.122\, G_{ox}^{0.58}\, p_c^{0.143}$





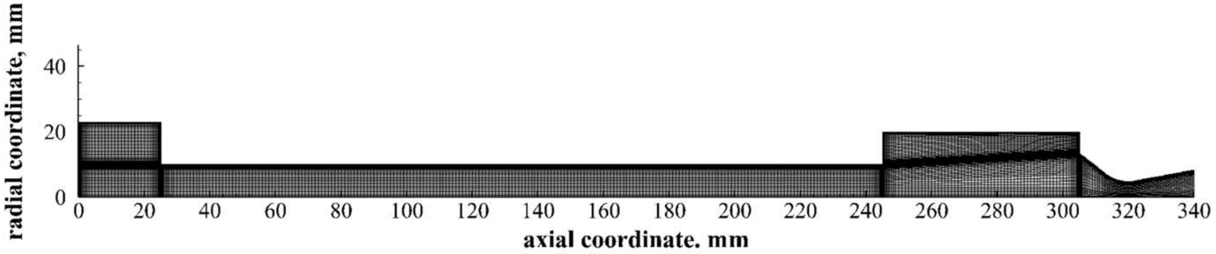
Fuel regression rate as function of the oxidizer mass flux

Modelling of hybrid rocket internal ballistics

Physical and numerical models for gaseous flowfield simulation

- **Reynolds-Averaged Navier-Stokes equations** for single-phase multicomponent turbulent reacting flows are solved with a control-volume-based technique and a pressure-based algorithm
- Turbulence model: **Shear Stress Transport (SST) k-omega**
- Combustion model: **Non-premixed combustion** based on the Probability Density Function (PDF) approach coupled to chemical equilibrium
- These models have been found the most suitable considering a large number of comparison with existing experimental data

- **The computational domain is based on the geometry of the engines** shown before

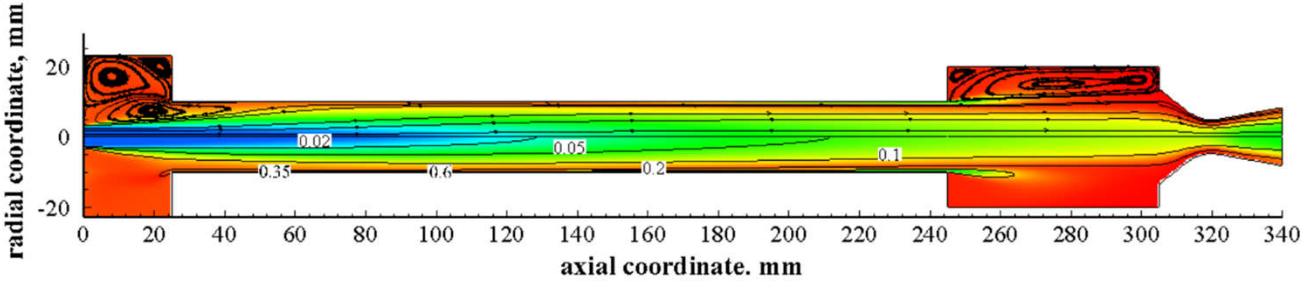


- **The boundary is divided in the following patches:**
  - **Oxidizer injector exit section** → BC type: mass flow inlet
  - **Pre-chamber, aft-mixing chamber and nozzle walls** → BC type: wall
  - **Fuel grain surface** → Dedicated treatment for fuel regression rate modelling estimated by means of an iterative procedure
  - **Nozzle outlet section** → BC type: pressure outlet

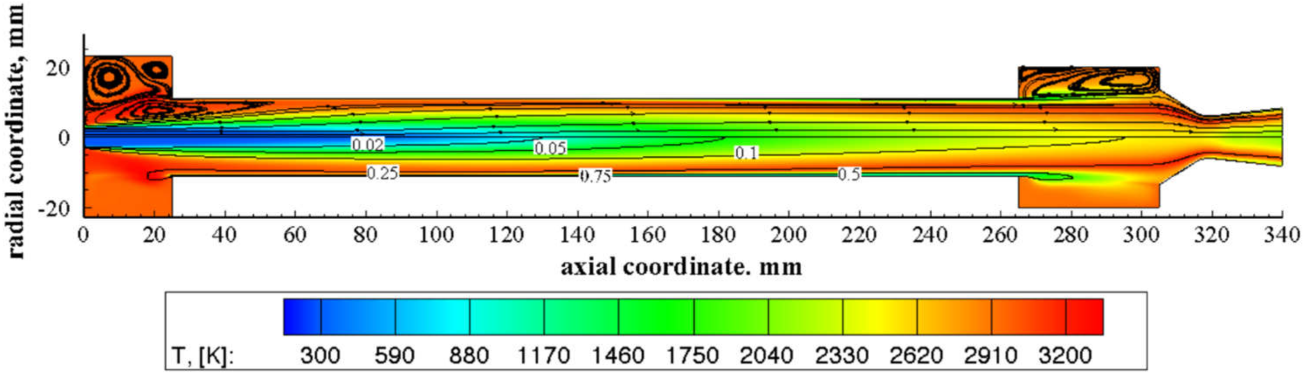
Numerical results of hybrid rockets internal ballistics simulation

The case of polymeric fuels: steady simulations

Numerical simulations at the conditions of the two test cases with polymeric fuels and the 200 N-class rocket shown before



Test HDPE-1



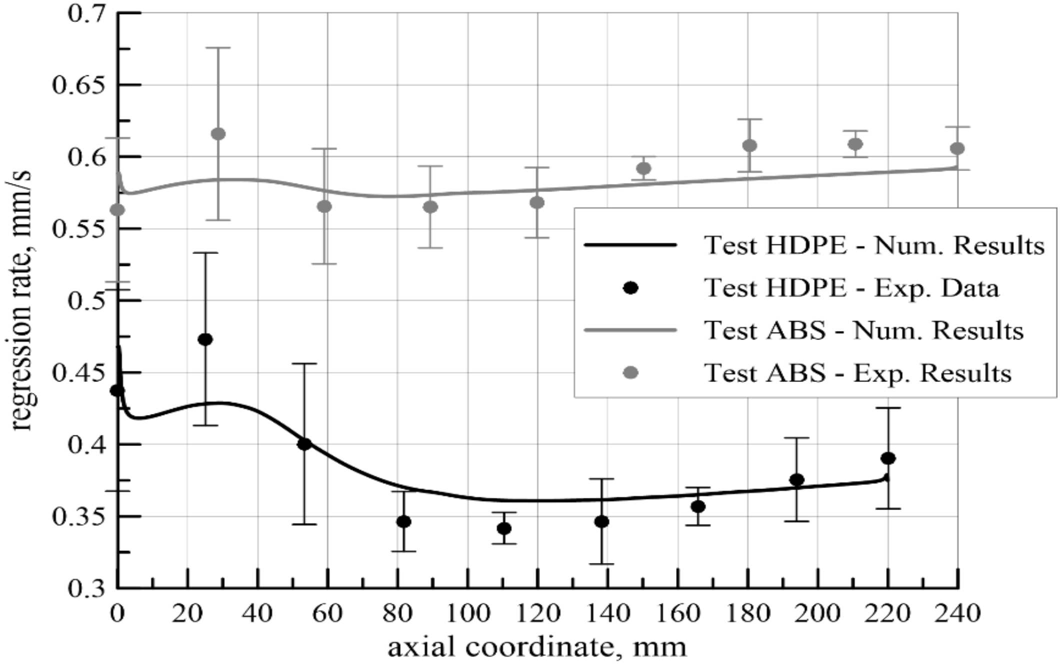
Test ABS-1

Temperature contour plot with overlapped streamlines (top half) and mixture fraction iso-lines (bottom half)

- Spreading of the oxygen jet up to an impingement point on the grain surface
- Extended recirculation region upstream of the impingement point
- Recirculation regions in the pre-chamber and in the aft-mixing chamber
- Temperature distribution reflecting the diffusion flame structure
- Propellant mixing promoted by the high turbulent kinetic energy generated in the vortices

Numerical results of hybrid rockets internal ballistics simulation

The case of polymeric fuels: steady simulations: Comparison of numerical results with experimental data



Regression rate distributions and comparison with experimental data

- Experimental points are obtained measuring the final port diameter in different cross sections
- Peak in the regression rate profiles due to the oxygen jet impingement
- Monotonically increasing trend of the regression rate profiles after a minimum point due to the effect of the mass addition, which is dominating on the effect of the boundary layer growth
- Good agreement between the computed and the experimental results both in terms of regression rate profiles and average pressures in the aft-mixing chamber

Computed average parameters and deviation with experimental data

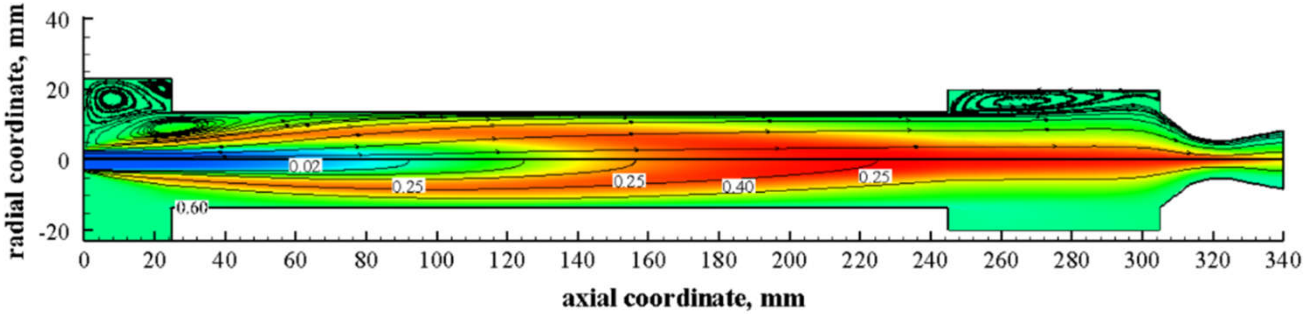
Test case	Computed averaged fuel regression rate (mm/s)	Regression rate relative error	Aft-mixing chamber pressure (atm)	Chamber pressure relative error
Test 1 (HDPE)	0.384	1.54%	6.52	1.7%
Test 2 (ABS)	0.581	4.75%	4.91	2.7%

Numerical results of hybrid rockets internal ballistics simulation

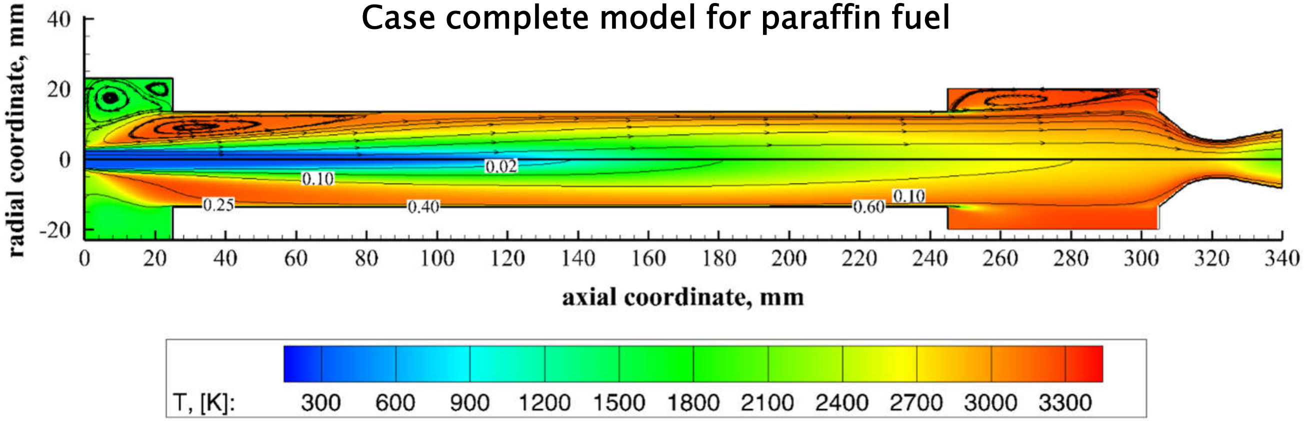
The case of liquefying fuels: comparison between the results with and without considering the entrainment

- Input conditions corresponding to Test P-4

Vaporization temperature equal to 675 K; Entrainment parameter equal to $2.1 \cdot 10^{-13}\ m^{8.5}s^{0.5}/kg^{3}$



Case complete model for paraffin fuel



Case HDPE fuel

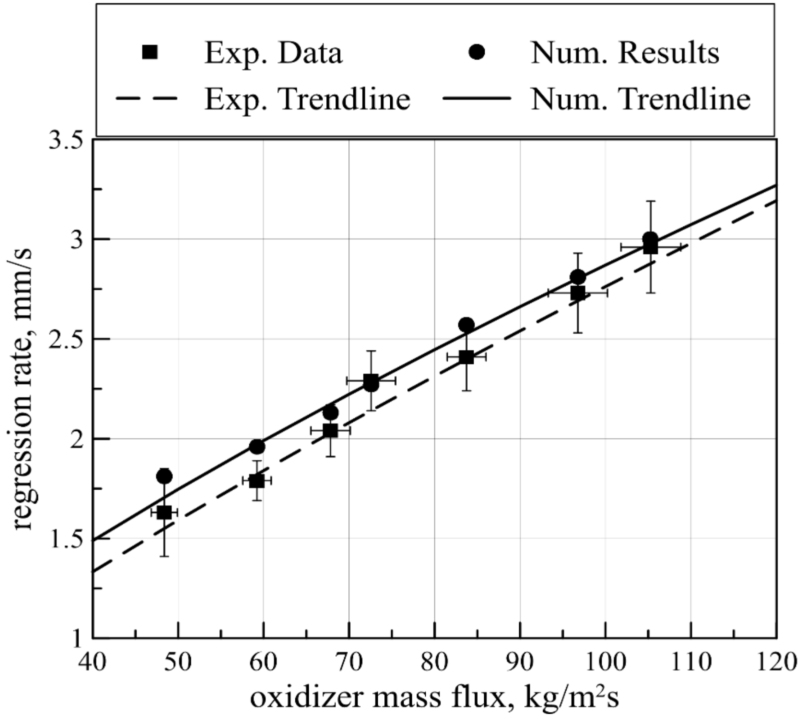
Temperature contour plot with overlapped streamlines (top half) and mixture fraction iso-lines (bottom half)

- Thermo-fluid dynamic flow field is similar to that shown before
- Temperature distribution reflecting the typical structure of a diffusion flame
- In the case with paraffin fuel, because of the significant fuel mass addition, due in great part to the entrainment and assigned in the whole port volume, the hotter region rapidly converge into the core flow

Numerical results of hybrid rockets internal ballistics simulation

The case of liquefying fuels: comparison of converged numerical results with experimental data

- Simulations with the oxygen mass flow rate and the average grain port diameter corresponding to the test cases shown before



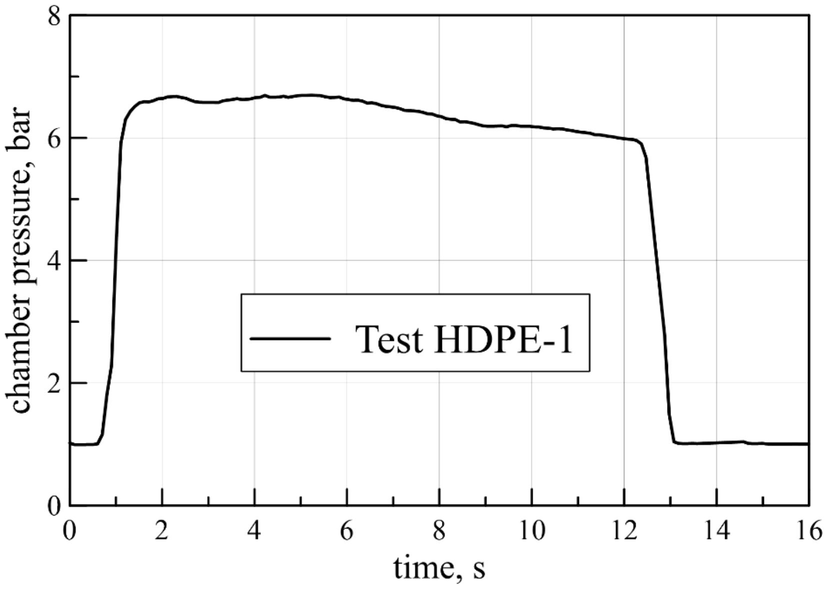
Computed regression rate deviations from experimental data

Test ID	Calculated space-averaged regression rate, mm/s	Error relative to experimental data
P-1	1.81	11.0%
P-2	1.96	9.5%
P-3	2.13	4.4%
P-4	2.27	0.9%
P-5	2.57	6.6%
P-6	2.81	2.9%
P-7	3.00	1.4%

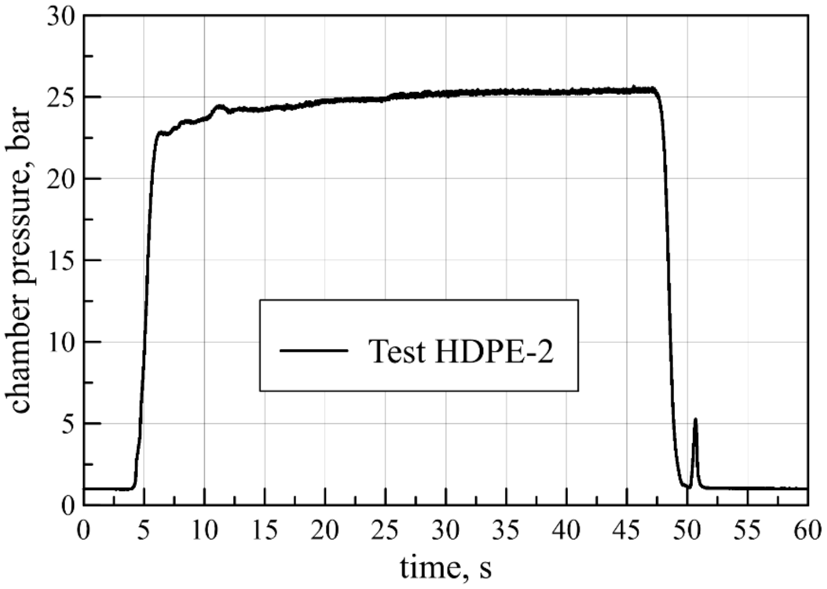
- Maximum deviation of 11% reached at the minimum mass flux
- Numerical prediction improves with higher mass fluxes showing excellent agreement at the largest mass fluxes where the deviation is around 1%

UHTCMC in hybrid rocket environment

Motivation for research project on new-class materials in hybrid rocket environments



Operating pressure vs time in the case of utilization of a classical graphite nozzle (significant throat section erosion detected)



Operating pressure vs time in the case of utilization of water-cooled copper alloy nozzle (constant throat section area)

UHTCMC in hybrid rocket environment

C^{3}HARME European Project

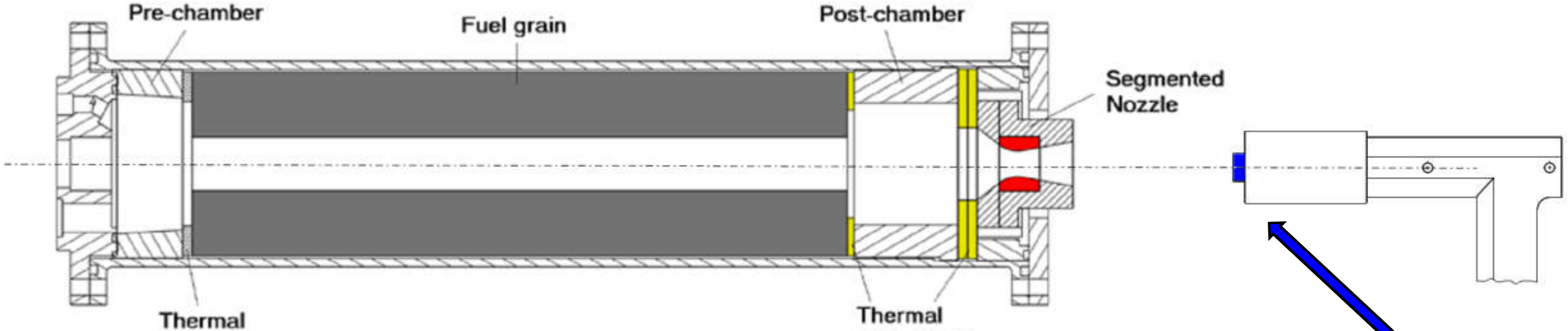
- NEXT GENERATION CERAMIC COMPOSITES FOR COMBUSTION HARSH ENVIRONMENTS AND SPACE
- Partners of the project: CNR, IN Srl, University of Birmingham, TECNALIA, UNINA, DLR, AVIO, NANOKER, HPS, Airbus, GMBH, Trinity College
- **Application: Near ZERO-Erosion nozzle** that can maintain dimensional stability during firing in combustion chambers of high performances rockets for civil aerospace propulsion

Activities of University of Naples "Federico II"

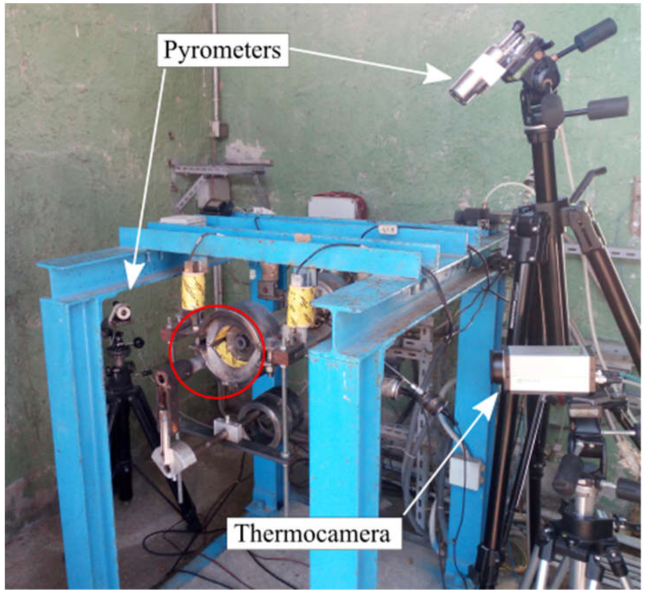
- **Design of prototypes and test conditions** for testing of new materials in hybrid rocket propulsion environment
- **First experimental tests on new UHTCMC materials for application in hybrid rockets**
  - Free jet tests on small material samples
  - Test of UHTCMC nozzle throat inserts
- **Definition of suitable numerical models** for supporting the experimental activities

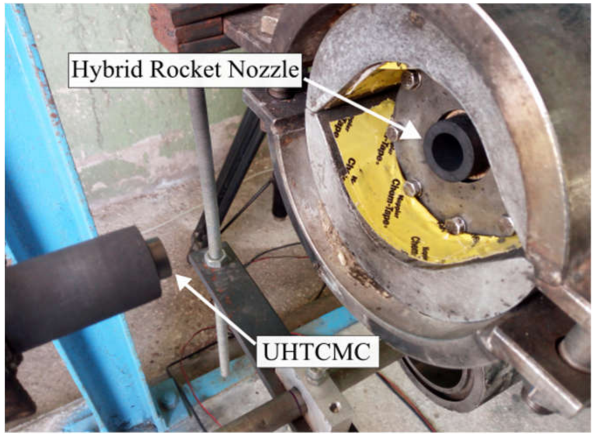
Characterization of UHTCMC in hybrid rocket propulsion environment (identification of the better candidates for nozzle)

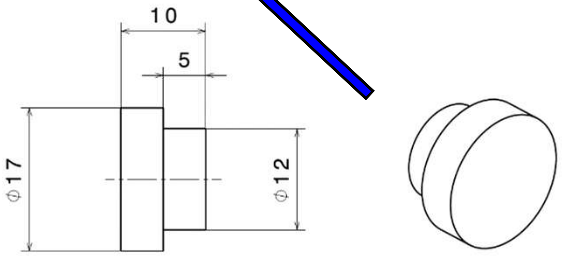
Experimental setups



200N-class Hybrid rocket motor and setup for UHTCMC free-jet testing





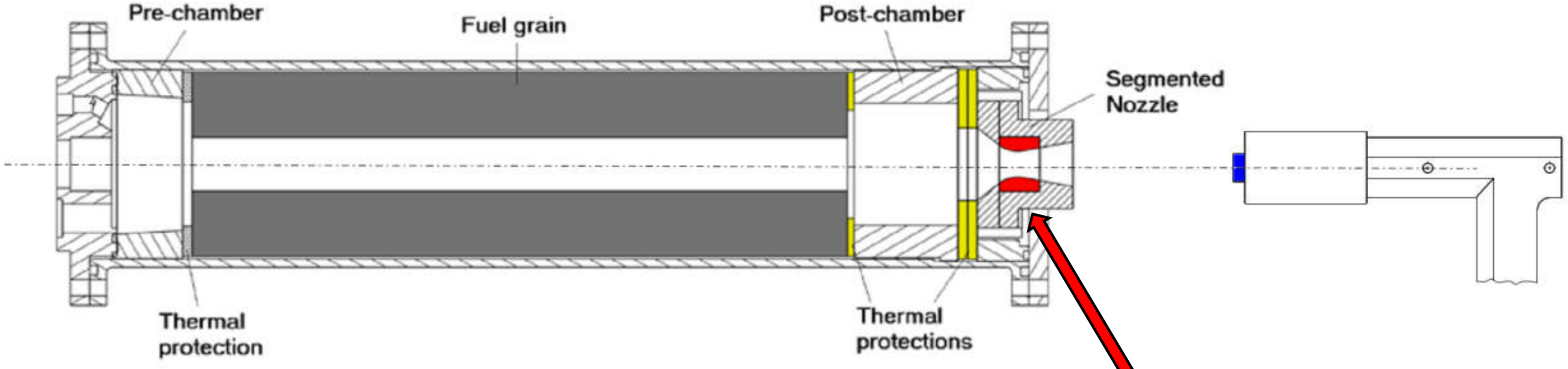


Nominal design of UHTCMC samples

- Gaseous oxygen axially injected
- Cylindrical grains of HDPE (220 mm long)
- Combusting flow expanded through converging-diverging graphite nozzle
- The sample is placed at a distance of 15 cm from nozzle exit

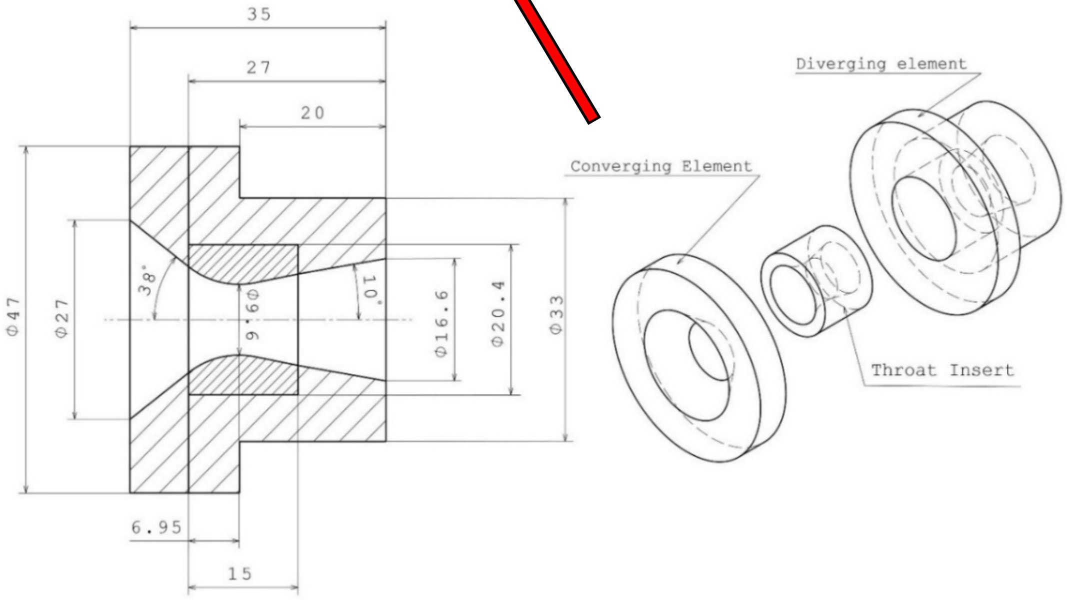
Characterization of UHTCMC nozzle throat inserts

Experimental setups

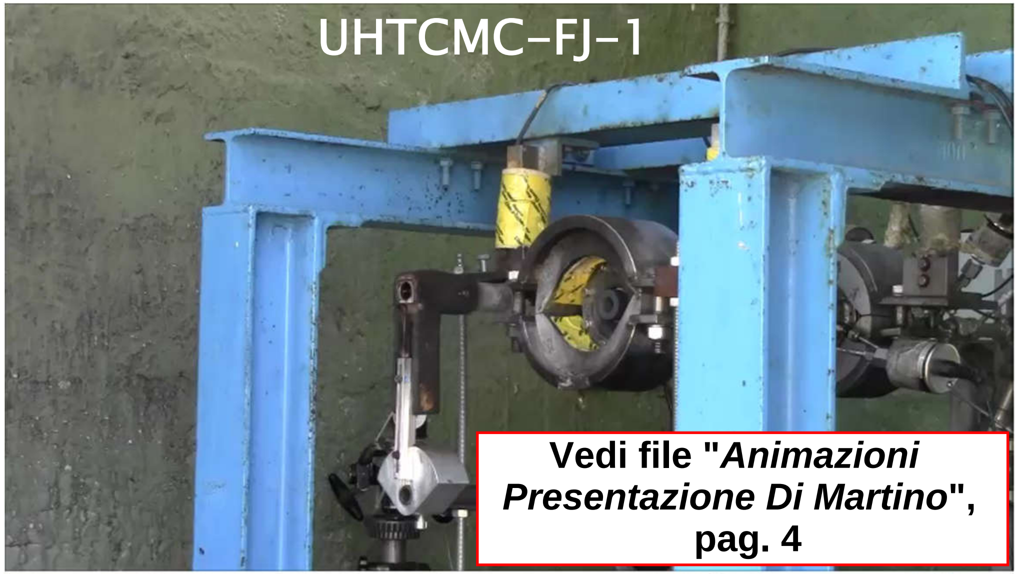


200N-class Hybrid rocket motor and setup for UHTCMC free-jet testing

- **Segmented-designed nozzle**:
  - Converging and diverging parts made of graphite
  - **Throat insert** made of the new materials to be tested
- Test the material in the most severe conditions, encountered right in the nozzle throat region



Free jet tests on UHTCMC samples


UHTCMC-FJ-1

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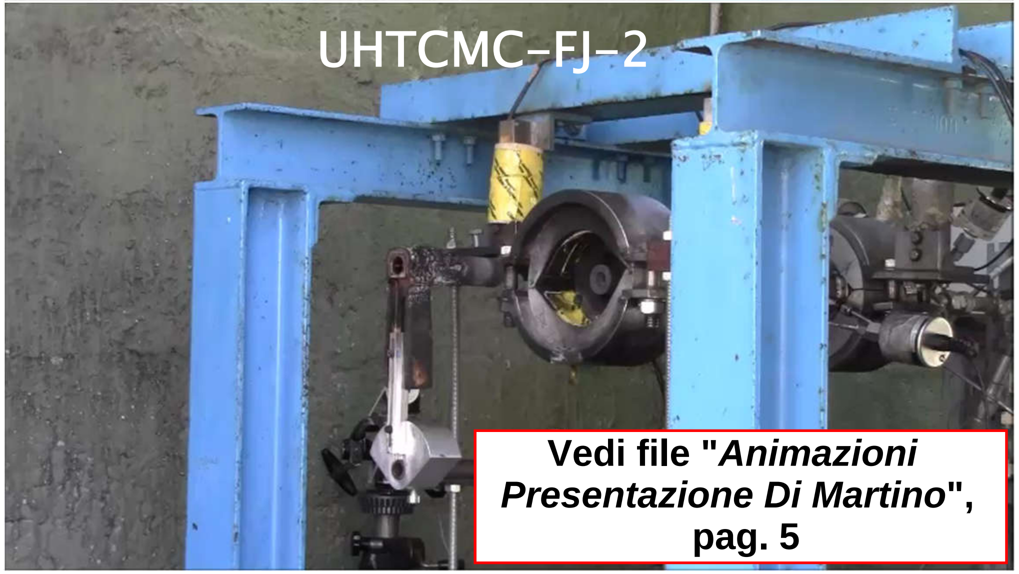
UHTCMC-FJ-2

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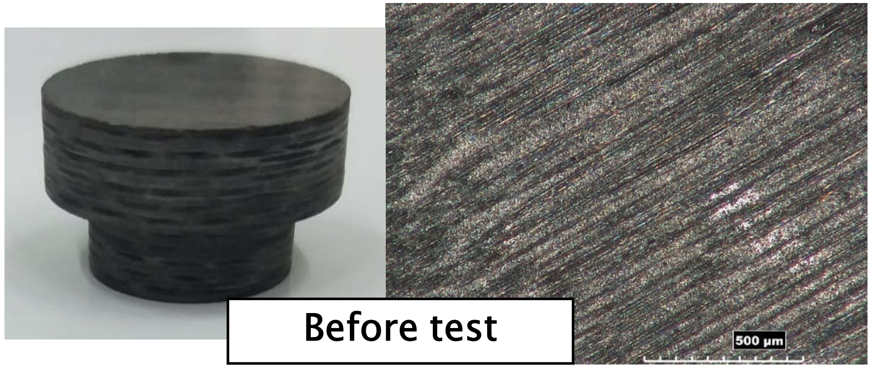
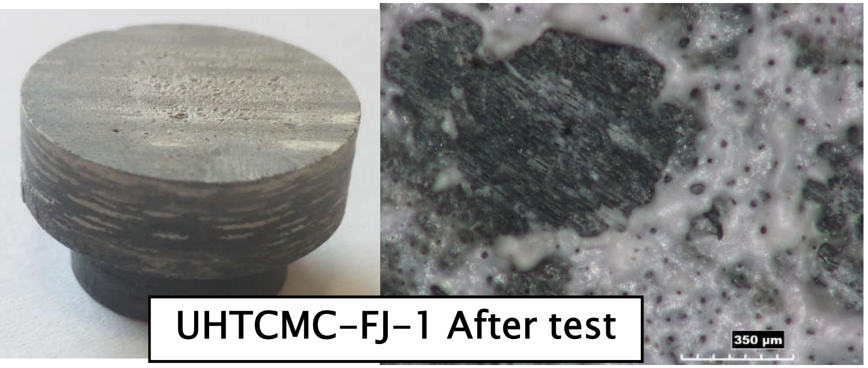
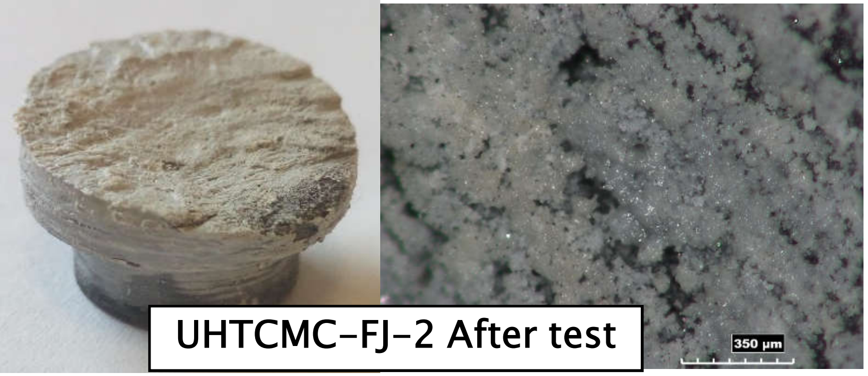
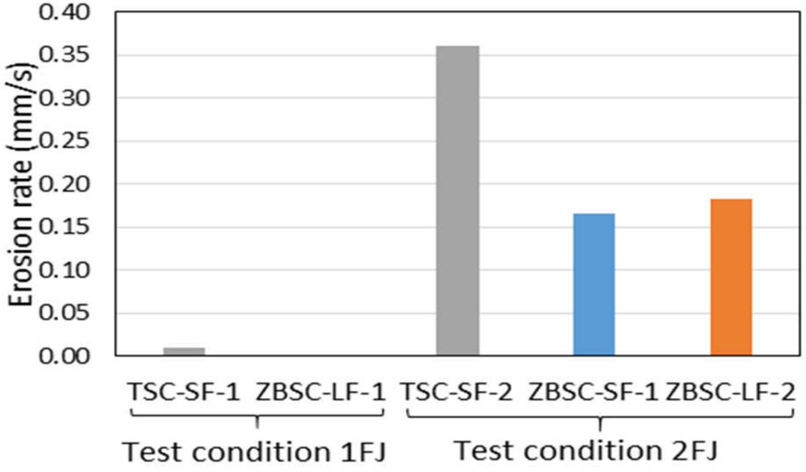

Before test


UHTCMC-FJ-1 After test


UHTCMC-FJ-2 After test

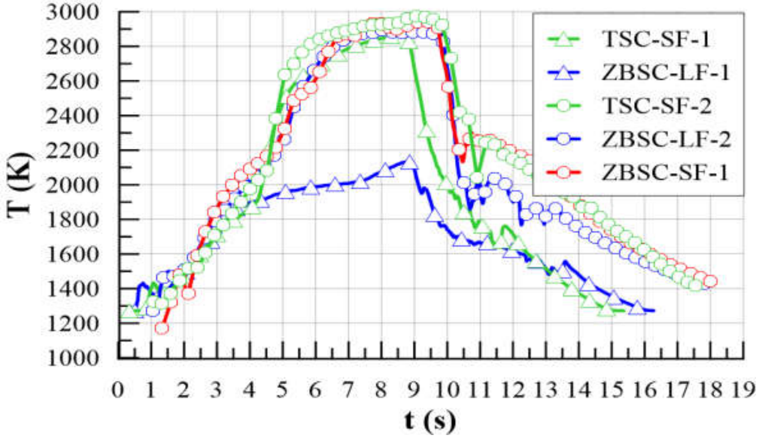
Free jet tests on UHTCMC samples

Experimental results

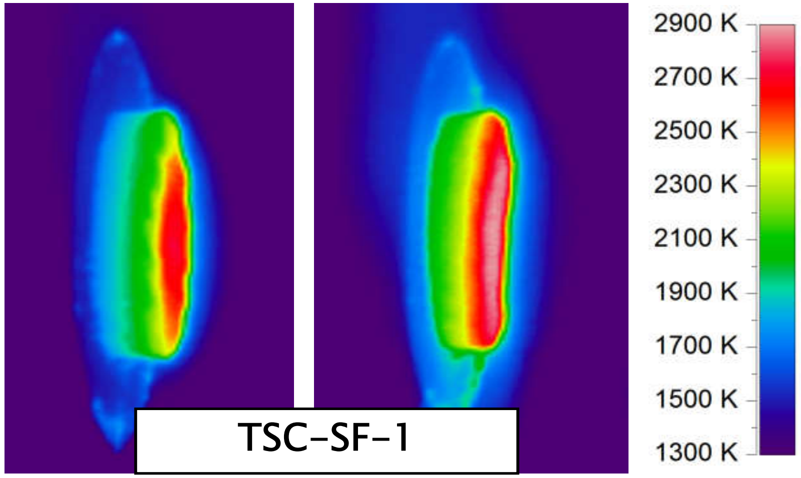


Erosion rates of UHTCMC samples in free jet test

Pictures of test on TSC-SF-1 sample, at beginning (left) and end (right) of the test



Thermal histories of UHTCMC samples in free jet test

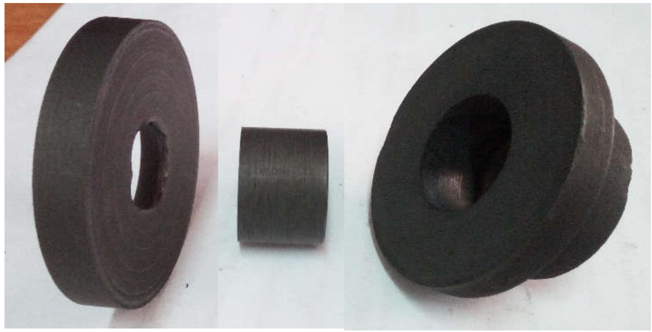


Test of UHTCMC nozzle throat inserts

UHTCMC samples and test conditions

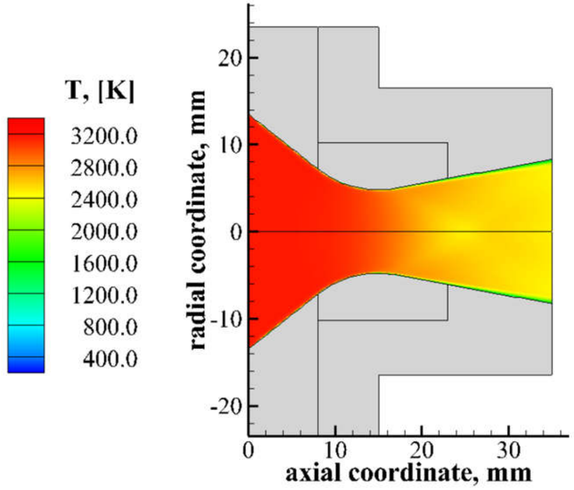
UHTCMC nozzle throat inserts

UHTCMC sample ID	Matrix composition	Carbon fibers
ZBSC-SF-TI	ZrB_2/SiC	Chopped
ZBSC-LF-TI	ZrB_2/SiC	Continuous Unidirectional



Nominal test conditions for throat insert tests

	Test condition 1TI	Test condition 2TI
Oxidizer mass flow rate [g/s]	25	40
Oxidizer-to-Fuel ratio	5.13	6.50
Chamber pressure [bar]	6.49	5.65
Combustion temperature [K]	~ 3200	~ 3200
Nozzle exit pressure [bar]	0.42	0.46
Nozzle inlet CO_2 mass fraction	0.32	0.32
Nozzle inlet H_2O mass fraction	0.16	0.14
Nozzle inlet O_2 mass fraction	0.30	0.41
Shear stress [hPa]	3.2	4.8
Average cold-wall surface heat flux [MW/m^2]	17.0	20.0



Example of temperature distribution through the nozzle

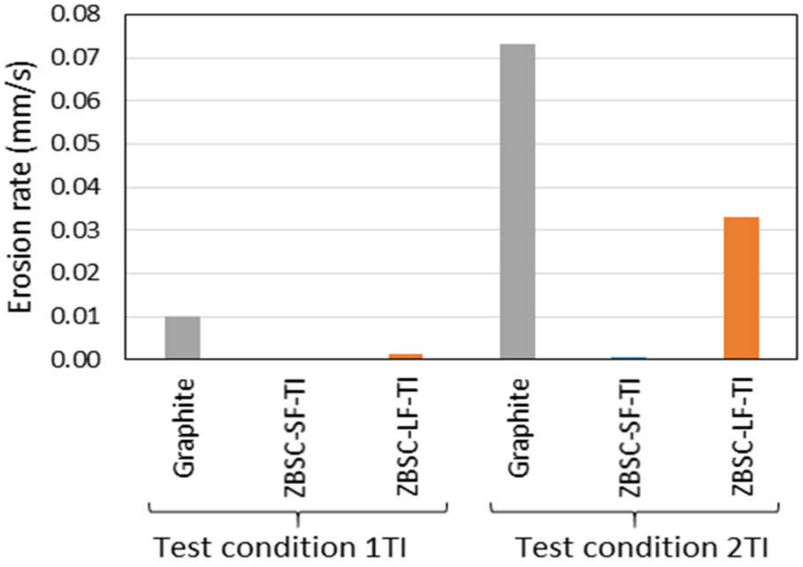
Test of UHTCMC nozzle throat inserts

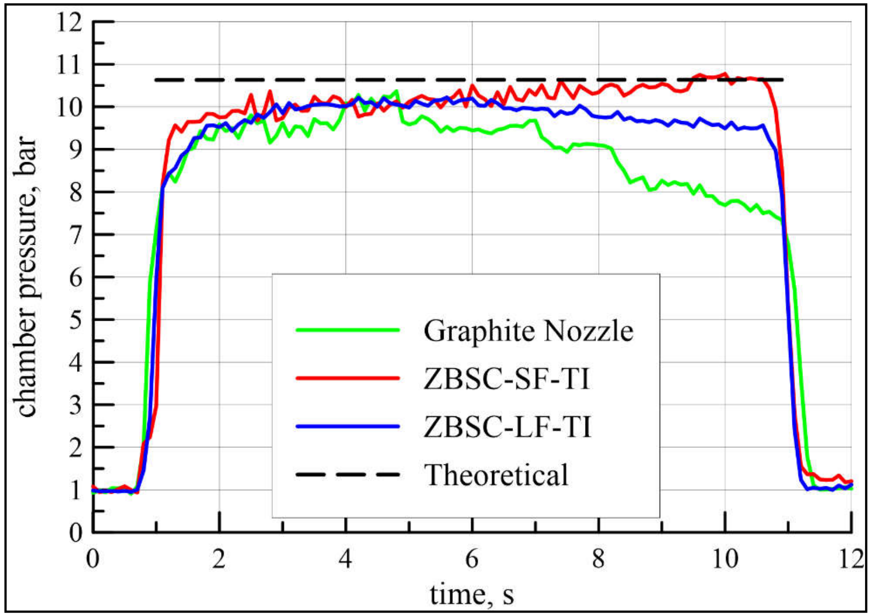
Experimental results

- Graphite nozzle:
  significant erosion during test in both conditions 1 and 2TI
- UHTCMC nozzle throat inserts:
  no significant erosion during test in conditions 1
  better erosion resistance than graphite nozzle also during test in conditions 2TI

Direct effect on rocket performance

- Strong decreasing trend of the chamber pressure in the case of graphite nozzle
- More stable behavior in the case of UHTCMC nozzle throat inserts





Theoretical and measured chamber pressures vs operating time for tests in conditions 2TI

THANK YOU
FOR YOUR ATTENTION!