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An Italian network to improve hybrid rocket performance: Strategy and results

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Abstract

The new international attention to hybrid space propulsion points out the need of a deeper understanding of physico-chemical phenomena controlling combustion process and fluid dynamics inside the motor. This research project has been carried on by a network of four Italian Universities; each of them being responsible for a specific topic. The task of Politecnico di Milano is an experimental activity concerning the study, development, manufacturing and characterization of advanced hybrid solid fuels with a high regression rate. The University of Naples is responsible for experimental activities focused on rocket motor scale characterization of the solid fuels developed and characterized at laboratory scale by Politecnico di Milano. The University of Rome has been studying the combustion chamber and nozzle of the hybrid rocket, defined in the coordinated program by advanced physical-mathematical models and numerical methods. Politecnico di Torino has been working on a multidisciplinary optimization code for optimal design of hybrid rocket motors, strongly related to the mission to be performed. The overall research project aims to increase the scientific knowledge of the combustion processes in hybrid rockets, using a strongly linked experimental–numerical approach. Methods and obtained results will be applied to implement a potential upgrade for the current generation of hybrid rocket motors. This paper presents the overall strategy, the organization, and the first experimental and numerical results of this joined effort to contribute to the development of improved hybrid propulsion systems.

Nomenclature

A_b sample burning area, m^2
B boundary layer blowing parameter
$c*$ characteristic velocity, m/s
E_j activation energy, kJ/mol
H thickness of the melt layer at the fuel surface, mm
G mass flux, kg/m^2 s

m mass, g
m_{en} entrainment component of mass flux from fuel surface, kg/m^2 s
p_d dynamic pressure, Pa
r_f solid fuel regression rate, mm/s
t_b combustion time, s
T temperature, K
M cinematic viscosity, mPa/s
P solid fuel density, kg/m^3
Σ surface tension, mN/m
Φ equivalence ratio (O/F)/(O/F)$_{stoichiometric}$
CB carbon black
GOX gaseous oxygen
HP hydrogen peroxide, H_2O_2
GW- gel wax-based fuels group
HTPB hydroxyl-terminated poly-butadiene
LOX liquid oxygen
H- HTPB-based fuels group
KER kerosene
MA maleic anhydride
MMH mono methyl hydrazine, $CH_3N_2H_3$
MO mineral oil
PE polyethylene
NTO nitrogen tetroxide, N_2O_4
PUF poly urethane foam
SEBS styrene–ethylene–butadiene–styrene
SW-solid wax-based fuels group
TPE thermoplastic polymers
UDMH unsymmetrical dimethil hydrazine, $(CH_3)_2$ N_2H_2

1. Introduction

The aim of this research project is to build an Italian scientific community focused on theoretical, experimental and numerical activities aimed to the improvement of hybrid rocket propulsion technology for the space access and exploration. The project consists in a two-year long work developed in such a way as to optimize the available budget.
Combustion processes represent the key problem in the development of the hybrid rocket design. The growing international attention to hybrid propulsion points out that the hybrid rocket propulsion system design needs of the understanding of physico-chemical phenomena that control the combustion process and of the fluid dynamics inside the motor. The knowledge of the complex interactions among fluid dynamics, solid fuel pyrolysis, oxidizer atomization and vaporization (in case of liquid oxidizer), mixing and combustion in the gas phase, particle formation, and radiative characteristics of the gas and the flame can only be improved by combined experimental and numerical research activities. Similar considerations can be made in the ablation process of the nozzle thermal protection. The numerical study of the flow in the combustion chamber and in the nozzle of a hybrid rocket requires careful handling of the interaction between the reacting flow and the solid surface.
This project includes four research units: Politecnico di Milano (PoliMi), Università degli Studi di Napoli "Federico II" (UniNa), Università di Roma "La Sapienza" (UniRoma) and Politecnico di Torino (PoliTo).
The objective of the experimental activity performed by PoliMi is the development of innovative solid fuels for hybrid propulsion, characterized by higher regression rates in comparison to the more

traditional fuels currently used. The aim of the investigation is to achieve a regression rate increase up to 3–4 times higher than the current values. Such an increase has been achieved with liquefying fuels, but the unacceptable mechanical properties of the grain after ignition make it unsuitaible for use. Further investigation of innovative formulations is therefore needed. The aim of this research program is also to face the problem of paraffin-based liquefying fuels mechanical properties. Innovative fuel formulations development, and their experimental characterization provide a set of experimental data useful for development and validation of the numerical codes developed by PoliMi, UniRoma and PoliTo units.
UniNa research unit is involved in firing tests of fuel formulations characterized on lab-scale by PoliMi in a hybrid rocket motor. Tests are carried out with several chamber pressures and oxidizer mass fluxes, and different fuel compositions. Furthermore, the motor thrust, the characteristic exhaust velocity c^*, the corresponding efficiency, chamber pressure and the fuel consumption spatial distribution are measured in order to collect useful design parameters and to compare the data with the theoretical models concerning the injection and fuel composition influence on regression rate, combustion efficiency and stability. Data referred to the flow conditions at the nozzle inlet (p,T,O/F) are given to UniRoma in order to allow the nozzle environment characterization and the thermal protection behavior evaluation. The final output will be the motor characterization and the achievement of a deeper knowledge on the hybrid motors operation.
UniRoma's objective is to study the combustion chamber and nozzle of hybrid rockets by means of physico-mathematical models and numerical methods. This, in turn, requires the knowledge of the complex interactions between fluid dynamics, the process of solid fuel pyrolysis, the atomization and vaporization of the oxidizer (if liquid oxidizer is considered), mixing between oxidizer and fuel, combustion in the gas phase, particulate formation, and radiative characteristics of the gas and flame. In a classical hybrid propellant rocket, the liquid or gaseous oxygen injected into the ports of solid fuel grain (typical fuel is HTPB) reacts in the combustion chamber with the pyrolysis gas, which is produced on the surface and diffuses into the boundary layer, forming a turbulent diffusion flame. The solid fuel in a hybrid rocket regresses slowly making, in fact, necessary to use a large fuel surface exposed to hot gas to get the mass flow rate required by the motor design. Classical studies on hybrid propulsion are based on simplified models of the boundary layer to derive the heat flux to the surface of the solid fuel and, consequently, its regression rate. However, this simplified analysis cannot take into account many of the complex chemical and physical interactions among the various processes, making necessary to develop more advanced models based on computational fluid dynamics (CFD) to improve prediction and analysis capabilities for such propulsion systems.
The main objective of PoliTo research unit is to develop a multidisciplinary optimization (MDO) procedure for hybrid rocket motors which joins the optimization of propulsion system design parameters and trajectory. The intrinsic interaction between propulsion system performance and trajectory, and the peculiarity of the combustion process of hybrid propellants require this kind of approach. The multidisciplinary approach allows to consider different aspects of the propulsion system design (e.g. fuel grain geometry), in an integrated manner, reducing the times the different subsystem models have to be accessed during the optimization process. Another goal of the research unit is the development of a fast and accurate model capable of describing the engine components and determining system weights. This goal will be pursued with strong interaction with the numerical/experimental work carried out by the other research units.
The overall research project aims to increase the scientific knowledge of the combustion processes in hybrid rockets, using a strongly linked experimental-numerical approach. Methodology and obtained results will be applied to study a potential upgrade for the current generation launchers. The introduction of a hybrid upper stage could represent a possible application of this research project. The Italian aerospace industry is currently involved in the exploitation and the development of evolved versions of a new European launcher, named Vega. This new launch system is designed to place small/medium size satellites into low-Earth orbits, used for many scientific and Earth observation missions. Vega will complete the range of launch services offered by Europe,

complementing Ariane (which is optimized for large satellites and missions to geostationary transfer orbit and low Earth orbit) and Soyuz (tailored for medium satellites to low Earth orbit and small spacecraft to GTO). Vega has been designed as a single-body launcher with three solid propulsion stages and an additional liquid propulsion (UDMH/NTO) upper module used for apogee boost and orbit circularization. The potential evolution of the Vega launcher and its mission could represent the reference scenario for the current research project.

2. The state of the art

The recent literature includes significant review papers [1-5], scale effects analysis [6], fuel characterization [7-10], methods to increase the regression rate [11]. Among these, some research activities deal with the influence on performance of the oxidizer injection system [12-14]. Two main trends can be observed: the use of nano metal powders, as ingredients able to increase the solid fuel regression rate (on this topic research activities were performed in Russia and at The Pennsylvania State University [4,15,16]; aluminium and boron powders, micro and nano-sized, were investigated) the use of paraffin-based solid fuels, also investigating the so-called entrainment phenomenon. Karabeyoglu and Cantwell at Stanford University were the first to understand the potential of this approach [17-19].

The regression rate increase due to the introduction of aluminium particles, or of other energetic material, in the fuel grain dates back to 1965; the regression rate increases proportionally to the metallic particles mass fraction [20,21] compared to pure fuel. Some tests performed with micro particles of aluminium have confirmed a non significant increasing in regression rate and a relevant aluminium residue in exhausts; this means that particles are not involved in the combustion process. Nowadays, due to the progress in the field of nano-technology, the idea of dispersing aluminium nano-sized particles in the fuel seems a good one [22,23]. Using results from tests in small scale performed at PoliMi, UniNa can select promising additives and their size. A few tests can be sufficient to give information to be compared with theoretical models and to suggest a way to improve the regression rate.

Currently, in studying the behaviour of hybrid rocket motors, the prediction of the solid grain regression rate and of the nozzle thermal protection material erosion rate, are strongly relying on empirical correlations [24] similar to the ones adopted for the analysis of ablative thermal protection systems (TPS), based on bulk mass/energy transfer coefficients derived from semi-empirical correlations [25-18].

Such approaches need to be accurately calibrated relying on the availability of existing experimental data specific for each analyzed motor. Simplified models only provide qualitative understanding of the trends but more general models [29] are needed, capable of representing more accurately the physico-chemical phenomena involved. Similar considerations can be made in the case of an ablative surface such as the protection materials. The study of the gas flow evolution inside the motor is essential for the correct modeling of the flow/surface interactions and in particular for the estimation of the regression rate which determines the overall sizing, mass fluxes, and geometric configuration of the hybrid motor. Therefore, more comprehensive computational fluid dynamic (CFD) models based on a coupled fluid/structure analysis are necessary for improved design and prediction capability. CFD codes weakly coupled with the models that describe the thermal behavior of the solid material [30] represent the first set of studies found in the open literature. However, to correctly evaluate the efficiency of the material, it is fundamental to study the complex coupling between CFD and solid surface through a detailed description of the physico-chemical phenomena that occur at fluid–solid interface (as addressed in [31] for the ablative thermal protection materials). In addition for the modeling of the combustion processes, homogeneous chemical reactions simplified models are used [32,33]. The complete numerical simulation of the flowfield inside hybrid motors requires adequate models both for the surface thermochemistry and for the homogeneous combustion. The most important and detailed results are the ones obtained by the Purdue University [32].

In this case the problem has been analyzed using a standard Navier–Stokes solver that includes the turbulent transport and chemical species equations, with a simplified kinetic model for the gaseous combustion process with 5 species and 2 global reactions involved [32,33]. The solid fuel pyrolysis is modeled with an Ahrrenius type expression. A detailed analysis through CFD can be useful to understand the combustion processes and hence design combustion chambers in hybrid motors. Other studies, instead, only deal with the flow over ablative thermal protection systems (solid rocket motors, reentry vehicles). In particular, surface mass and energy balances have been introduced at the fluid–solid interface which become a boundary condition for the Navier–Stokes solver. According to the range of surface temperature, the problem has been faced both in the hypothesis of heterogeneous surface chemical equilibrium [34] and in the more general non-equilibrium case [35]. These kinds of models have been presented for different types of materials that constitute the solid surface [36].
Hybrid rocket motors are attractive as they present some of the benefit of both liquid rocket engines and solid rocket motors. Hybrid rocket motors can be shut-off and restarted like liquid rocket engines, and can be throttled within a wide thrust range. Moreover, special safety steps needed for chemicals such as NTO, MMH and UDMH are eliminated and as a result operation costs are reduced. A large number of papers concerning numerical and experimental investigations can be found in the literature [37-43]. Potential applications of hybrid propulsion range from sounding rockets to space/satellite propulsion and launchers. A hybrid motor was used for a manned suborbital flight to a 100-km altitude and will be probably employed for commercial space flights [42]. An application of particular interest, due to the features of hybrid propellants, is the use of hybrid propellant motors to replace solid rocket motors and storable liquid rocket engines used in launchers upper stages. Due to the peculiar combustion process of hybrid motors, the determination of the strategy to obtain high specific impulse is more complex than in solid and liquid rockets. For these reasons, the design optimization should include trajectory analysis and, possibly, its optimization. In most numerical investigations, the optimization is performed to find the best grain and nozzle geometries as well as the best operating conditions, such as mean mixture ratio and chamber pressure, disregarding trajectory optimization. The PoliTo research unit has a wide experience in the optimization of spacecraft trajectories [44], [45] and ascent trajectories [46] and in the analysis of propulsion system performance. Procedures for the simultaneous and integrated optimization of propulsion system and trajectories [47] have recently been applied to hybrid propellant rockets. A preliminary study showed that the introduction of a hybrid propellant upper stage to replace solid motors can remarkably improve the launcher performance.

3. Technical-scientific content of the project

PoliMi is responsible for the development and characterization of innovative solid fuels, based on the use of metal nano powders and paraffin. The experimental activity concerns the study, development, manufacturing of solid fuels starting with the theoretical performance analysis in term of specific impulse, setting up the manufacturing procedure and finally performing the experimental characterization using a test bench specifically designed for this project. The new fuels characterization, performed by the PoliMi research unit, deals with regression rate measurements obtained with non intrusive optical techniques; surface fuel grain temperature measurements, obtained by microthermocouples; flame structure analysis, studied using a high speed video-camera. The obtained results form a data bank and each unit involved in numerical simulation will use these data to validate the numerical codes.
Selected fuels, developed and characterized at lab-scale, are then characterized at motor scale by UniNa. Fuel regression rate, motor thrust, characteristic exhaust velocity c^* and the corresponding efficiency, chamber pressure (downstream of the injector and upstream of the exhaust nozzle) and fuel consumption spatial distribution are measured in order to get experimental results to validate the numerical predictions. Data referred to the flow conditions at the nozzle inlet (p,T,O/F) will be given

to UniRoma in order to allow the nozzle environment characterization and the evaluation of different thermal protections behaviour.
UniRoma is dealing with the study of the combustion chamber and the nozzle of hybrid rocket engines involved in the coordinated research program, using physico-mathematical models and numerical methods, handling adequately the interaction between the chemically reacting flow and the solid surface. The approach is based on a Reynolds averaged Navier–Stokes solver for chemically reacting flows; an ablative wall boundary condition has been implemented, where flow and wall properties are calculated through mass and energy balances under the assumption of steady ablation. Once the approach is validated, an analysis is made of the scale effects on fuel regression rate and ablative thermal protection erosion. The combustion models can be validated thanks to the experimental data produced by UniNa and PoliMi units.
The goal of the PoliTo research unit is to develop a multidisciplinary method to optimize hybrid rocket motors. The optimal design of the propulsion system is strongly related to the mission to be performed. Furthermore, the peculiar combustion processes of hybrid propellants do not allow, unless using more complex feeding system, the control of thrust level and mixture ratio as a time. For these reasons, the combined study of engine design optimization, oxidizer and trajectory control is needed. The aforementioned problems are characterized by different peculiarities, and thus different optimization techniques must be adopted. In fact, the propulsive system optimization generally requires the determination of optimum values for a limited number of parameters, while the trajectory optimization requires the time-depending governing laws. The developed general optimization tools will be applied, in the final phase of the project, to a specific case of interest: an improved configuration of theVega launcher. The other units will provide data and simulation models for the optimization codes.

4. Results

4.1. Fuel formulations

The results presented in this paper mainly include the regression rate performance results obtained for several groups of fuel formulations. Pure HTPB-based formulations are used as a reference for the comparison; the other groups include paraffin-based solid fuels. The choice of paraffin as a matrix for solid fuels is aimed to obtain high regression rate values; nevertheless, it is well known that paraffin poor mechanical properties are a major drawback. For this reason, the paraffin matrix strengthening technique plays a major role in the research of innovative solid fuels.
Paraffin waxes GW (Gel Wax, $C_{12}H_{26}$) and SW (Solid Wax, $C_{24}H_{50}$) were selected to manufacture several formulations, through additive filling with nano-aluminum (Alex, with average particle size of 100 nm), or Magnesium hydride (MgH_2, particle size in the range 50–150 µm), or Lithium Aluminum hydride ($LiAlH_4$, particle size in the range 80–100 µm) powders. The manufactured and tested fuel formulations include a reference fuel, based on HTPB (H–), a group based on gel wax (GW–), a group based on solid wax (SW–), a fourth group based on GW, PUF and KER (GWPK–), and a fifth group based on SW filled with SEBS, mineral oil (MO) and/or KER.
Two strengthening strategies are used in the tests presented in this work. The first one is based on a poly-urethane foam (PUF) structure; this leads to a notable increase in the regression rate, but results in heterogeneous fuels, thus in non-isotropic mechanical properties. A second type of strengthening structure involves thermoplastic polymers (TPE) soluble in paraffin (SEBS-MA), with the aim to increase the paraffin elasticity without a significant decrease in the regression rate value and ensuring isotropic mechanical properties. SEBS is a thermoplastic polymer belonging to the family of styrene–buthadiene rubbers (SBR) with styrene content of about 30%. This thermoplastic polymer offers a good chemical compatibility with paraffinic materials. This characteristic ensures that a homogeneous formulation is obtained. The SW used in this work, supplied by an Italian Company, has a melting temperature of 333 K.The use of TPE reinforcing structure in paraffin-based fuels

results in relatively low manufacturing costs and homogeneous fuels. Moreover, the GW-PUF formulation melting point temperature and viscosity decrease through additive (kerosene) inclusion was investigated, with the aim to increase the fuel regression rate up to values similar to those typical of SW-PUF formulation.

Preliminary tests, performed at low oxidizer mass flux, show that SW-SEBS-based formulations containing solid wax (SW), Sigma Aldrich styrene-(ethylene-butadiene)-styrene polymer (SEBS), and 1% carbon black (CB) allow obtaining higher regression rate if compared to HTPB-based fuels. Several tests were performed in order to compare the regression rate in double slab configuration for those formulations which appear the most promising after a pre-burning rheological investigation [48-50]. Fuel nomenclature, composition, metal or metal hydride particles nominal size, density and regression rate percentage increase with respect to pure HTPB for these tests are presented in Table 1.

Table 1. Fuel formulations tested (double slab configuration). Oxygen mass flux: 150 kg/m^2 s. Operating pressure: 1.5 bar.

No.	**Fuel**	**Formulations ingredients**	**Metal powder size µm**	**ρ_{fuel} (g/cm^3)**	**Δr_f (%)**
1	HTPB	HTPB 100%	–	0.92	--
2	H-MGH5	HTPB 95% MGH 5%	50–150	0.94	57
3	GWP	GW 97% PUF 3%	–	0.88	61
4	GWP-LAH3	GW 94% PUF 3% LAH 3%	80–100	0.88	71
5	GWP-MGH5	GW 92% PUF 3% MGH 5%	50–150	0.87	102
6	GWP-Alex100-5	GW 92% PUF 3% Alex 5%	0.1	0.87	202
7	GWPK	GW 87.3% PUF 3% KER 9.7%	–	0.87	126
8	GWPK-MGH3	GW 84.6%PUF 3% KER 9.4% MGH 3%	50–150	0.88	144
9	GWPK-LAH10	PUF 3% GW 78.3% KER 8.7% LAH 10%	80–100	0.88	157
10	SWP	SW 97% PUF 3%	–	0.88	186
11	SWP-Alex100-5	SW 92% PUF 3% Alex 5%	0.1	0.87	208
12	SWP-MGH5	SW 92% PUF 3% MGH 5%	50–150	0.87	267
13	SWP-LAH3	SW 94% PUF 3% LAH 3%	80–100	0.88	219
14	SWP-LAH6	SW 91% PUF 3% LAH 6%	80–100	0.88	327
15	SWP-LAH10	SW 87% PUF 3% LAH 10%	80–100	0.88	378
16	SEBS-K	SW 50% SEBS 15% MO 25% KER 10%	–	0.90	142
17	SEBS-K-MGH5	SW 50% SEBS 15% MO 25% KER 10% MGH 5%	50–150	0.90	154

The experimental lab-scale test rig designed and developed at SPLab (Space Propulsion Laboratory) of PoliMi for the investigation of transient phenomena includes a combustion chamber equipped with an oxygen (oxidizer) and a nitrogen (used for a quick extinction after the oxygen shut off) inlet system. A piezo-electric pressure transducer (Kulite 35), a pyrotechnic ignition device, check and relief valves, and a Bronkhorst F113 Mass Flow Controller (MFC) complete the 2D slab burner device. The ignition is obtained using a small (1 g) charge of non-metalized propellant placed at the combustion chamber head-end. The propellant is ignited by means of a hot wire, electrically heated; the propellant ignition causes a flame resulting in the fuel slab combustion onset. The fuel slabs sizes are 50×13×17 mm; double slab configuration is used for the firing tests. The regression rate (r_f), averaged in space and time, is measured from the burned mass (Δm), the fuel density (ρ_f), the burning time (t_b) and the burning area (A_b), using the following equation:

$$r_f = \frac{\Delta m}{\rho_f t_b A_b} \tag{1}$$

4.2. Lab-scale experimental investigation

Firing tests allowed comparing the average regression rates of the different fuel formulations investigated. Pure gaseous oxygen was used as oxidizer, with a mass flux ranging from 100 to 350 kg/m^2 s. The pure HTPB regression rate is about 0.6 mm/s at the highest oxygen mass flux tested (350 kg/m^2 s); at the same mass flux GW and SW give a regression rate, under the considered operating conditions (p=1.5 bar), of about 1.1 and 2 mm/s, respectively. The regression rate vs. oxygen mass flux curves were obtained for all the tested fuel formulations, assuming a power law:

$$r_f = aG^n \tag{2}$$

Fuel accumulation and turbulence intensity increase along the fuel sample length, with a corresponding increase in heat transfer to the fuel surface and in regression rate. This trend, confirmed by several experimental regression rate results, is strongly mitigated by the sample size under the operating conditions of this work. Likewise, the regression rate usually decreases with time due to the port area increase, which determines a corresponding decrease in the port mass flux. Combustion times are very short in this investigation, thus allowing to neglect also this dependence. Sometimes regression rate correlations, proposed for hybrids, include the boundary layer blowing parameter, usually named B, which can be defined as the ratio between the core flow thermal energy (per unit mass) and the fuel gasification thermal energy (per unit mass), required at the fuel surface in order to sustain the solid to gas fuel transition. As noted by Marxman, B is raised to a small power, allowing to express the fuel regression rate dependence primarily on G, which in this work is considered as the oxidizer mass flux.

The average regression rate of the tested fuel formulations was measured at a reference condition, corresponding to 150 kg/m^2 s oxidizer mass flux and 1.5 bar operating pressure. Such mass flux was chosen in order to guarantee the onset of entrainment phenomena related to specific features of the experimental set-up used. Tests were performed in double slab configuration, with pure oxygen as oxidizer. The results of the ballistic characterization are shown in Fig. 1, where the different formulations are compared to the reference formulation (pure HTPB). The regression rate percentage increase with respect to pure HTPB is also reported in Table 1 at the same reference oxidizer mass flux.

The regression rate percentage increase with respect to the reference baseline formulation (pure HTPB, formulation no. 1 in Table 1) ranges from +57% for HTPB filled with 5% MgH_2 (no. 2)., up to+378% for SW filled with 10% $LiAlH_4$ (no. 15). Intermediate values are obtained for the GW-based fuels (from +61% (no. 3) up to +202% (no. 6)) and for GWK-based fuels (from +126% (no. 7) up to +144% (no. 8)). The highest regression rate enhancement, under the investigated operating conditions, is obtained with $LiAlH_4$ addition in SW binder (up to+378% compared to pure HTPB (formulation no. 15)). Results obtained suggest that kerosene addition is effective in enhancing GW-based fuels r_f, by decreasing their viscosity and thus increasing their tendency to entrainment [48, 49].

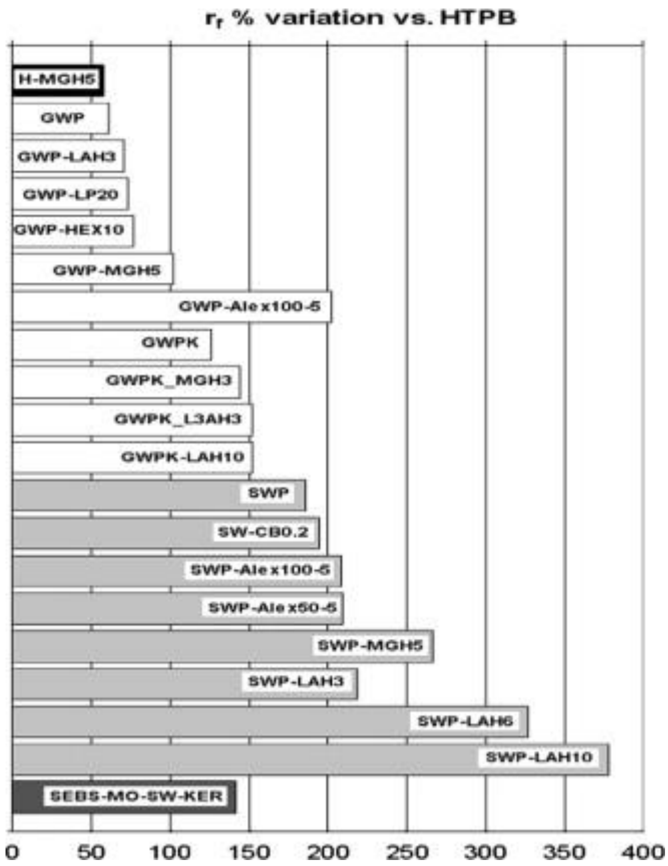

Fig. 1. Regression rate comparison for several different fuel formulations. Oxygen mass flux: 150 kg/m^2 s. Operating pressure: 1.5 bar.

The reasons for the observed behavior are due to the fuel higher or lower tendency to entrainment effect [51-53], which is determined by the fuel physical properties, in particular viscosity and surface tension. SEBS-based fuels display regression rate values which are very close (at low oxygen mass fluxes) to those typical of pure SW, or lower (at oxygen mass fluxes higher than 200 kg/m^2 s, approximately). This can be explained taking into account the different viscosity values of the fuel formulations: a higher viscosity results in a lower tendency to entrainment effect. A higher viscosity, therefore, is detrimental to entrainment effect and in turn to regression rate. The higher the oxygen flux, the higher the stresses on the surface are.

The regression rate increase is due to the generation of mass transfer by mechanical means, added to the mass transfer due to the fuel gasification. Following Karabeyoglu et al. [29,30], in the framework of the linear theory developed for the liquid entrainment modelling, the general empirical expression for the entrainment rate of liquid droplets is:

$$\dot{m}_{ent} \propto \frac{p_d^{\alpha} h^{\beta}}{\mu^{\gamma} \sigma^{\pi}}$$

where α, β, γ and π are empirical coefficients. This extra mass transfer mechanism can significantly increase the regression rate over that of traditional fuels, such as HTPB.

4.3. Motor-scale experimental investigation

The motor, in the class of 0.2 kN thrust, has an axisymmetric combustion chamber. The oxidizer is oxygen or nitrous oxide; in the last case it is supplied by a reservoir of 10 cylinders from which it is delivered in gaseous phase with an upstream pressure approximately about 40 bar. The cylinder rack is connected to the motor feeding line with an electronically controlled pressure regulator. Oxidizer mass flow rate is evaluated through gas temperature and pressure measurements across a sub-critical Venturi tube. Temperature and pressure are measured at a section upstream of the Venturi throat; a differential pressure transducer is used to measure the pressure drop between the section upstream of the throat and the throat itself. Oxidizer is axially injected into the combustion port through a showerhead injector with 13 holes with 1.1 mm diameter. A sketch of the motor is shown in Fig. 2. The inner case diameter is 69.2 mm and the length 350 mm.

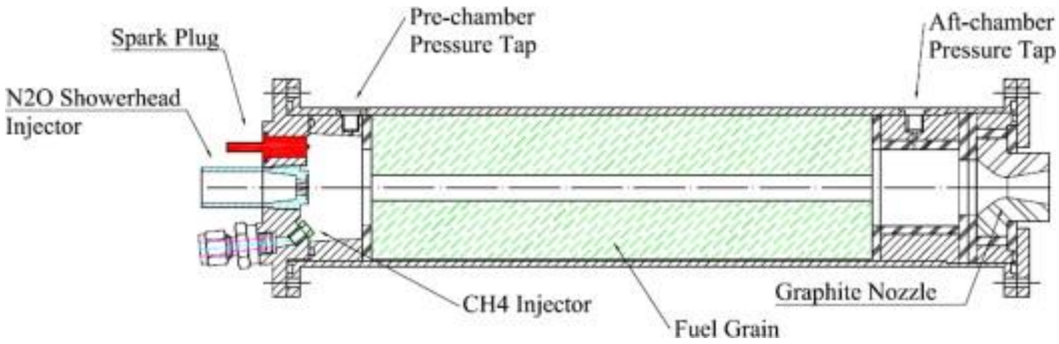

Fig. 2. Sketch of the motor designed and operated by the Research Unit of Naples University.

The pre-chamber makes the recirculation region caused by the oxidizer injection move towards the fore end of the grain, in order to increase the overall regression rate. The aft-chamber, covered with thermal insulations, promotes additional gas mixing, thus improving the combustion efficiency. Graphite converging-diverging nozzles with either 9.6 or 12 mm throat diameter have been employed. The motor is suspended from the test bench by 4 load cells; this arrangement allows computing the motor thrust as the sum of the loads measured by each cell. Chamber pressure is measured by two capacitive transducers, Setra model C206, set up in the pre-chamber and in the aft-chamber. A spark plug (powered by a Honeywell solid-state igniter spark generator) is arranged in the pre-chamber, where methane gas is injected for 3 s together with the oxidizer to ignite the motor. This system has been set up in order to ensure repeatable conditions at the motor ignition as well as to guarantee motor re-ignition.

Using a software developed in LabView, the firing test is completely automated. The time-space-averaged regression rate is calculated from the ratio between the average fuel mass flow rate (determined by dividing the fuel grain mass loss by the burning time) and the fuel density multiplied by the combustion area, according to the equation:

$$r_f = \dot{m}/(\rho_f \pi L D) \tag{3}$$

where L is the grain length and ρ_f is the fuel density; $\dot{m}$ is the average fuel mass flow rate determined by dividing the fuel grain mass loss by the burning time, t_b. D is the port diameter averaged over the entire burning; it is determined starting from the initial port diameter D_0 and the final average port diameter D_f, estimated by means of the fuel mass burned as follows:

$$D_f = [D_0{}^2 + 4\Delta m/(\rho_f \pi L)]^{½} \tag{4}$$

The burning time, which involves the identification of the initial surface regression and the web burnout instants, has been determined based on a well assessed procedure; the inflection point on the primary rise portion of the pre-chamber pressure trace and the one on the end decrease portion are assumed, respectively, as these two characteristic times.

The time-space-averaged mass flux (whether it is the oxidizer or the total one) is defined based on the average port diameter:

$$D = 0.5(D_0 + D_f) :$$

$$G = 4(\dot{m}/\pi D^2) \tag{5}$$

The combustion efficiency, ηC^*, is defined as the ratio between the experimentally measured characteristic exhaust velocity, c^* to the theoretical one computed with the CEA chemical equilibrium code at the effective mean pressure measured in the aft-mixing chamber (p_{aft}).

Among the fuels previously discussed, the characterization at motor scale was focused on the fuels of the fifth group, based on SW filled with SEBS. Results are presented for the fuel grain formulation which consists of 84% SW, 15% SEBS and 1% CB. This formulation was found to be the most promising in a campaign performed on radial samples; results are not reported in Table 1, which summarizes the regression rate results obtained for slab samples.

The average parameters measured in the burning tests are reported in Table 2.

Table 2. Firing tests results for SW (84%)+SEBS (15%)+CB (1%).

Test	GOX	O/F	G_{tot}	$\dot{m}_{ox}$	p_c	r_f	c^*	ηc^*	p_{feed}
	kg/m² s		kg/m² s	kg/s	bar	mm/s	m/s		bar
SW_01	32.8	1.5	55.2	0.028	4.12	0.87	1003.2	0.86	11.5
SW_02	54.4	1.5	90.4	0.047	5.41	1.48	785.7	0.66	18.0
SW_03	91.8	1.9	139.5	0.073	12.35	1.89	806.2	0.65	28.0
SW_05	81.4	1.3	145.4	0.066	15.84	2.57	946.1	0.81	27.5

Tests carried out with paraffin-based fuels show pressure oscillations, as can be observed in Fig. 3. This pressure behaviour is quite similar in tests SW_02 and SW_03 and it is common to several experimental data found in the literature with the same kind of propellants. Large pressure spikes have been observed in the early phase of combustion; this behaviour is probably due to some chunks of fuel produced at the grain inlet. All grains, indeed, have been subjected to a loss of fragments. Paraffin fuel chunks are generated as a consequence of grain poor mechanical properties; low stiffness can be due to the low casting temperature that inhibits the breakdown of the maleic anhydride molecule, which is a component of SEBS. This molecule at high temperature is brought to break its characteristic ring structure creating strong chemical bonds with the consequent improvement of the polymer mechanical properties. This phenomenon has been observed when casting a previous batch of grains, for which higher temperature has been achieved as well as larger stiffness.

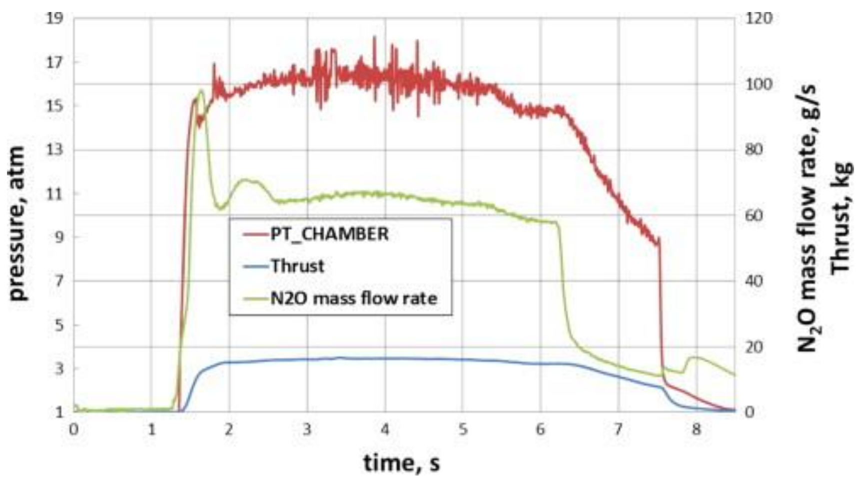

Fig. 3. Pressure, nitrous oxide mass flow rate and motor thrust vs. time measured for test SW_05 reported in Table 2.

Fuel fragments production can influence also the regression rate measurement, which appear larger than expected, as shown in Fig. 4, when compared with data obtained with gaseous oxygen and swirling nitrous oxide. The highest dotted curve for SW-GOX is taken from [18]; results obtained in the present work are represented by the other SW-GOX curve (dash dot dash). The lowest curve for SW-N_2O (dash dot dot dash) is taken from [61]. Results of the present work are reported as blue rhombs. Test SW_02 (oxidizer flux around 60 kg/m^2 s) was affected by nozzle failure, which occurred in the middle of the firing; therefore the regression rate measurement has been estimated only on the first portion of the test, prior to failure. The measurement uncertainty has been consequently evaluated and reported with the relative error bars. Whereas the point around 80 kg/m^2 s (relevant to test SW_05) was affected by a major grain failure and appears definitely out of range.

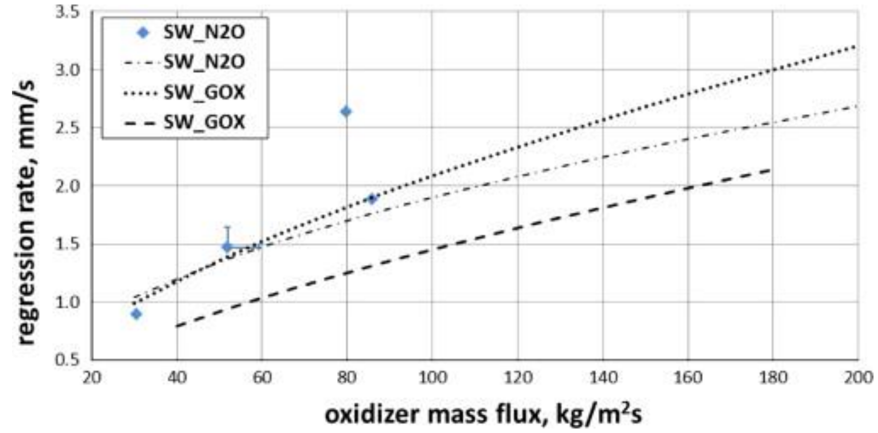

Fig. 4. Regression rate vs. oxidizer mass flux. Comparison among different data set.

Regression rates gathered from all the tested fuel formulations (basically pure HTPB, aluminum-loaded HTPB and paraffin-wax) are compared in Fig. 5 vs. the oxidizer mass flux (either gaseous oxygen or nitrous oxide). The curve for SW-N_2O (dash) is taken from [61] for a comparison with all other data reported in Fig. 5 obtained in this work.

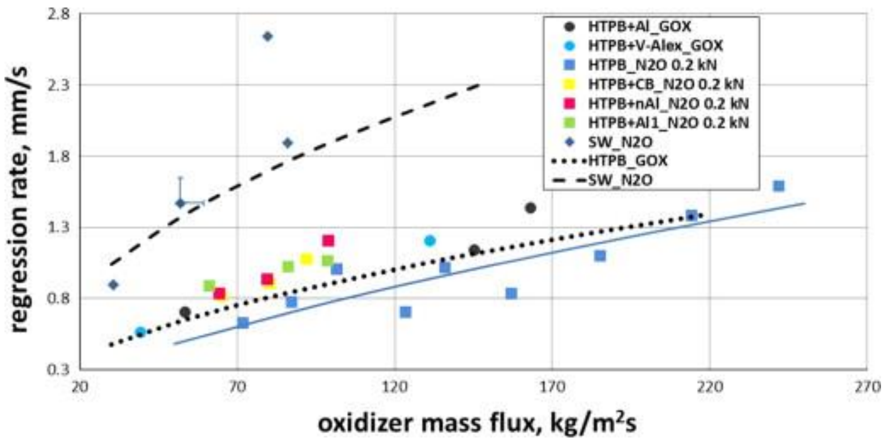

Fig. 5. Regression rate vs. oxidizer mass flux. Trend of SW-based fuels compared to HTPB-based, pure or aluminized, with GOX or N_2O oxidizers.

Aluminium powder addition to HTPB has a positive effect on regression rate, both with oxygen and nitrous oxide. Aluminized HTPB burnt with nitrous oxide, in the range of fluxes and pressures tested, show a regression rate increase which is not appreciably sensitive to either aluminium particles mass fraction or size.

Moreover, it seems that aluminium improves the regression of HTPB more effectively if burnt with nitrous oxide than with oxygen. A possible reason can be related to the flame adiabatic temperature increase due to the aluminium addition, which is higher with nitrous oxide than with oxygen in the range of mixture ratios investigated.

Efficiency of paraffin grains is considerably lower than the one of HTPB-based fuels. The maximum value achieved over the four tests of Table 2 is 86%. Lower efficiency is expected when burning paraffin grains due to the higher fuel flow rate which does not burn in the combustion chamber, but in this case these values probably appear even lower because of the anomalous behaviour of the solid grains (fragments loss). During the firing test a non negligible amount of liquid paraffin is generated, entrained in the main flow and transported back to the injection region (due to the broad recirculation head of the combustion port), where it is deposited on the injection flange, as post-firing inspections demonstrate. This issue has been observed also with oxygen. In addition, unburned fuel was found on the thermal protections covering the aft-mixing chamber. Of course, fuel slivers are not consumed during combustion, and do not contribute to raise the combustion chamber pressure, thus lowering the efficiency.
Regression rates measured with paraffin-wax and nitrous oxide are the largest over all the available data. This result, even if almost aligned with literature, is affected by fuel fragments loss, as previously mentioned.

5. Graphite nozzle erosion caused by thermochemical ablation

Nozzle ablative cooling is adopted for solid rocket motors (SRM) as well as hybrid rocket motors (HRM). Therefore, nozzle and material technologies widely used during design and development of SRMs can be successfully applied to study hybrid motors. However, in so doing, it should be considered that operating conditions of thermal protective materials (TPS) in hybrid motors differ from those in SRM. A main distinctive feature of HRM operating conditions is the greater concentration of oxygen-containing combustion products than SRM, which can significantly affect the material behavior. For this reason, the throat erosion rate in a hybrid is generally significantly greater than the one in a solid propellant system and is a strong function of chamber pressure and mixture ratio. For accurate performance predictions, it is necessary to include the nozzle erosion in the design process as the continuous enlargement of the nozzle throat during the firing directly affects the engine thrust curve and specific impulse.
To evaluate the erosion behavior of nozzle protection materials, a proper description and modeling of the interaction between the combustion gases arising from hybrid fuels and the protective material is required. Currently, in studying the behavior of hybrid rocket motors, the predictions of the nozzle thermal protection material erosion rate are strongly relying on empirical correlations. Such simplified models are clearly limited to providing qualitative understanding of the trends but are inadequate to providing the kind of quantitative data needed for engine design and optimization. Indeed, the extension of such models to new engines which can be different in scale, geometry, etc. is hardly possible without the availability of experimental data for each engine. The development of more general numerical models capable of representing more accurately the physico-chemical interactions between the reacting flow and the protective material is hence required for HRM. The authors have developed and validated an approach able to treat in detail the interaction between the chemically reacting hot gases and the carbon-based ablative thermal protection material [54]. The approach relies on a validated full Navier–Stokes flow solver coupled with a thermochemical ablation model which takes into account finite-rate heterogeneous chemical reactions at the nozzle surface, rate of diffusion of the species through the boundary layer, ablation species injection in the boundary layer, heat conduction inside the nozzle material, and variable multispecies thermophysical properties. The parametric analysis performed in this study allows to assess the impact of various parameters that affect the nozzle erosion rate, such as O/F ratio, chamber pressure, and combustion efficiency. The rocket-nozzle material considered in the present study is graphite, which is one of the most widely used nozzle material. The heterogeneous gas-surface chemical reactions are described by a semi-global heterogeneous reaction mechanism for graphite oxidation consisting of five reactions, listed in Table 3, Table 4.

$$\dot{m}_i = \Sigma^4{}_{j=1} A_j T_w \exp(-E_j/RT_w) p_i^{nj} \tag{6}$$

where j=H_2O, CO_2, OH, O; A_j is the pre-exponential factor, T_w the surface temperature, E_j the activation energy, R the gas constant.

Table 3. Heterogeneous reaction rate constants for graphite with H_2O, CO_2, OH, O, according to: Eq. (6)

Surface reaction	j	A_j	E_j (kJ/mol)	b_j	n_j
$C_3+H_2O \rightarrow CO+H_2$	1	4.8×10^5	287.9	0.0	0.5
$C_3+CO_2 \rightarrow 2CO$	2	9.0×10^3	285.1	0.0	0.5
$C_3+OH \rightarrow CO+H$	3	3.61×10^2	0.0	−0.5	1.0
$C_3+O \rightarrow CO$	4	6.65×10^2	0.0	−0.5	1.0

Table 4. Heterogeneous reaction rate constants for graphite with O_2, according to Eq. (7) $Y=(1+k_g/(k_7\, p_{O2}))^{-1}$.

Surface reaction	J	A_j	E_j (kJ/mol)	b_j
$C_3+½\, O_2 \rightarrow CO$	5	2.4×10^3	125.6	0.0
	6	2.13×10^1	−17.2	0.0
	7	5.35×10^{-1}	63.6	0.0
	8	1.81×10^7	406.0	0.0

The hot exhaust gas flowing in the nozzle consists of the combustion products of various fuel and oxidizer combinations. A total of three fuels (hydroxyl-terminated polybutadiene, HTPB; polyethylene, PE, and paraffin wax, SW) and a total of four oxidizers (90% hydrogen peroxide, oxygen, nitrous oxide, and nitrogen tetroxide) have been considered. Nozzle inflow conditions have been obtained by a chemical equilibrium code at a pressure of 10 bar and for an equivalence ratio of 1.0. Ten gaseous species have been considered for each simulation as they constitute, for each case, more than 99.9% of the total combustion gas mass.(7)

Fig. 6 shows the exhaust gas composition, in terms of mass fractions, for all the propellant combinations. Clearly, there is a strong effect of the oxidizer choice on the exhaust gas composition which reflects into a significant variation of the throat erosion rate level. This is summarized in Fig. 7, where the throat erosion contribution from each of the oxidizing species is reported as computed on a reference nozzle geometry.

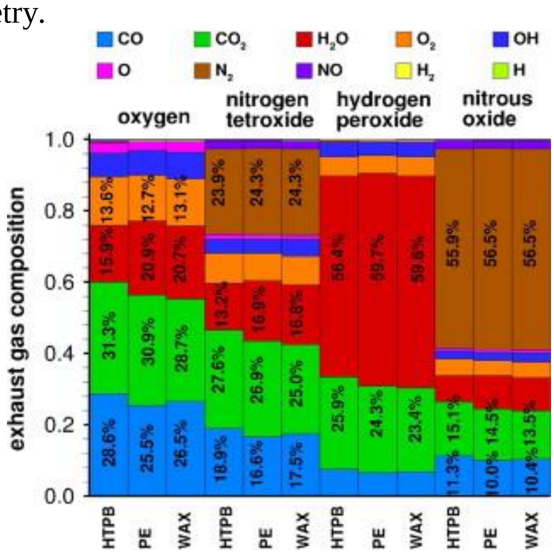

Fig. 6. Exhaust gas composition for different hybrid propellant combination at stoichiometric mixture ratio and chamber pressure of 10 bar.

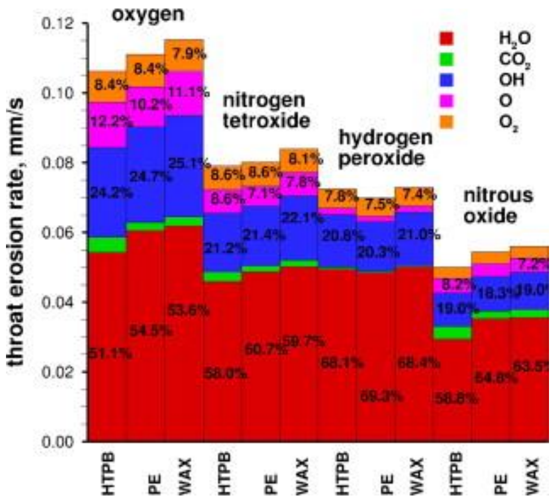

Fig. 7. Throat erosion rate for different hybrid propellant combination at stoichiometric mixture ratio, and chamber pressure of 10 bar.

Analysis of the erosion behavior has shown that oxidizing reactions with atomic and molecular oxygen, typically neglected in SRM nozzles, are important for HRM and can contribute up to 20% to total erosion for stoichiometric conditions. Water vapor is found to be the most important oxidizing species, contributing to more than 50% of the total erosion rate for every propellant combination. Typically, erosion is found to be diffusion-limited for radicals O and OH, which are characterized by faster reaction rates with graphite, and kinetic-limited for O_2 and CO_2, characterized by slower kinetics. Erosion contribution from H_2O can be either diffusion-limited or kinetic-limited depending on the propellant and operating conditions. Erosion rates are found to be 1.5 to 3 times that of a comparable solid rocket motor, depending on the choice of the oxidizer. The type of oxidizer, in fact, can significantly influence the erosion rate, with high-oxygen content oxidizers such as oxygen and nitrogen tetroxide showing the highest values. An extensive parametric analysis including the effect of different combinations of fuels and oxidizers, different operating conditions (chamber pressure, O/F ratio, gaseous/liquid injection) and other parameters which can affect the material behavior (combustion efficiency, wall radiation, and flow field chemistry) is reported in [55].

Here, the results obtained for the GOX-SW, which is the propellant pair selected for the present project, are briefly discussed, focusing the attention on the effect of the mixture ratio on the nozzle erosion rate. A peculiar characteristic of hybrid rocket motors is, in fact, an intrinsic shifting of the O/F ratio during both steady-state operation and throttling. Typically, shifts in mixture ratio during burning can involve both fuel-rich and oxidizer-rich conditions and their effect on the throat erosion rate need to be evaluated in order to accurately predict the motor performance. Two equivalence ratios Φ of 1.5 and 0.5 have been selected for fuel-rich and oxidizer-rich conditions, respectively. Typically, the flame temperature shows a maximum for the stoichiometric condition, and tends to equally decrease for fuel- or oxidizer-rich mixtures. Radical species such as O and OH, which are shown to react at high rates with graphite, usually peak close to the stoichiometric condition, due to the higher flame temperature which favors their production. The dominant oxidizing species H_2O, instead, tends to increase for mildly fuel-rich conditions due to the increasing hydrogen content in the mixture. Differently, O_2 concentration is rising continuously as the mixture ratio becomes Φ increasingly oxidizer-rich.

Fig. 8, Fig. 9 show the erosion rate and the wall temperature distributions at varying equivalence ratios assuming a reference nozzle geometry characterized by a throat diameter of 5.08 cm. Interestingly, the erosion rate is mildly affected by the mixture ratio for oxidizer-rich conditions while it is strongly reduced for fuel-rich conditions. This is due to the fact that, while flame temperature is reduced when off-stoichiometric mixture are considered, molecular oxygen presence in the combustion gases is greatly increased for oxidizer-rich mixtures thus leading to a heat release by the graphite oxidizing reactions which in turns increases the wall temperature and the overall reaction rate. Therefore, hybrid motor operation at oxygen-rich mixture ratios and high pressures typically results in very high throat erosion rates while operation at fuel-rich mixture ratios and low chamber pressure generates very low throat erosion rates.

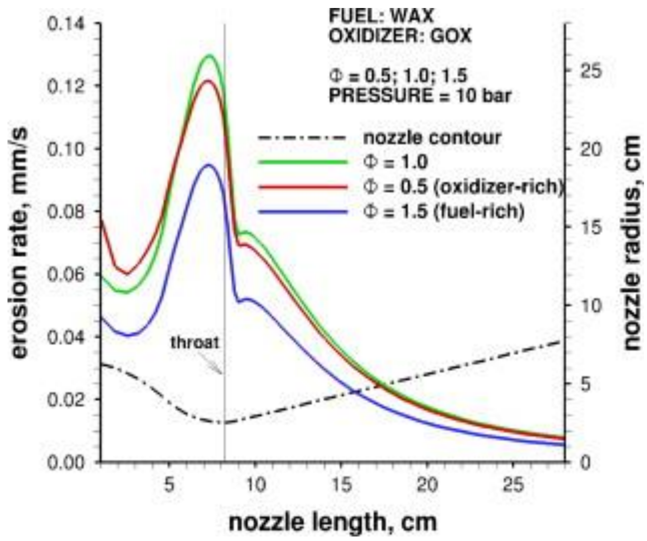

Fig. 8. Computed erosion rate for SW/GOX propellant combination at three different equivalence ratio in case of the reference nozzle geometry and 10 bar chamber pressure.

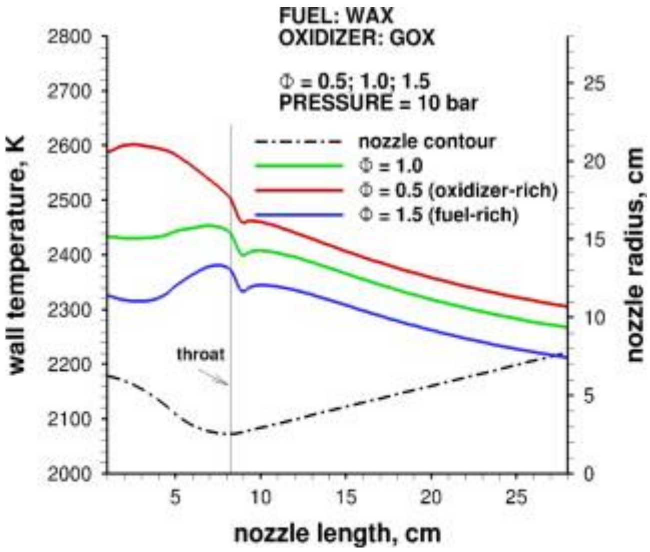

Fig. 9. Computed wall temperature for SW/GOX propellant combination at three different equivalence ratio in case of the reference nozzle geometry and 10 bar chamber pressure.

After this general overview on the erosion behavior of graphite in HRM, attention has been focused on the motor tested by UniNa, which is characterized by a graphite nozzle with a throat diameter of 9.6 mm. Fig. 10 shows the nozzle total erosion rate and the erosion contributions from the five oxidizing species. As shown, water vapor and the hydroxyl radical OH are the dominant oxidizing species, contributing to almost 70% of the total erosion rate for the present test case. The remaining 30% is due to erosion contributions from atomic and molecular oxygen, which are found to be important oxidizing species in hybrid motors. Finally, the erosion contribution from CO_2 is minimal, despite its abundance in the exhaust gas, due to its slow kinetics. Referring to the data that could be measured at the end of the experimental campaign it is underlined that the expected value of erosion rate at throat is less than 0.1 mm/s.

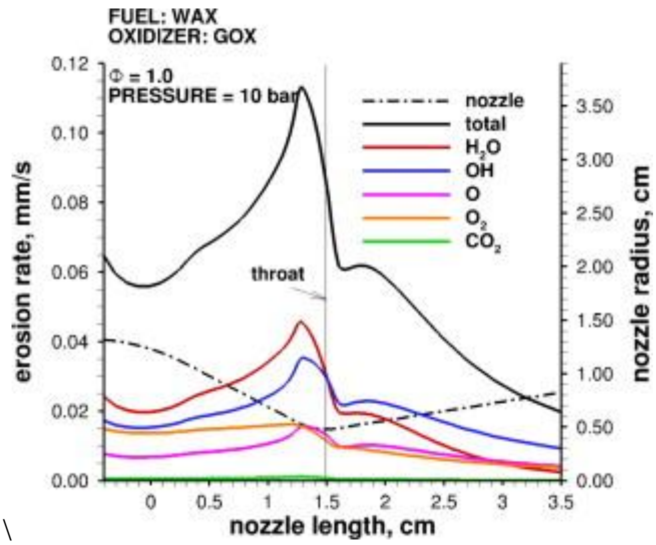

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Fig. 10. Computed erosion rate for SW/GOX propellant combination at stoichiometric equivalence ratio in case of UniNa motor nozzle geometry and 10 bar chamber pressure.

6. Multidisciplinary optimization of hybrid rocket motor

For any given application and mission, the design parameters and the corresponding operation parameters of the hybrid rocket motor (HRM) may have different optimal values, which depend on assumptions such as propellant combination, grain geometry, and feed system. As an example, Fig. 11 shows the thrust history of an optimized HRM with a multiport grain. The propellant combination is HP/PE. The required thrust level depends on the number of grain ports and on the feed systems (blowdown, pressurized–partially regulated and turbopump) [56].

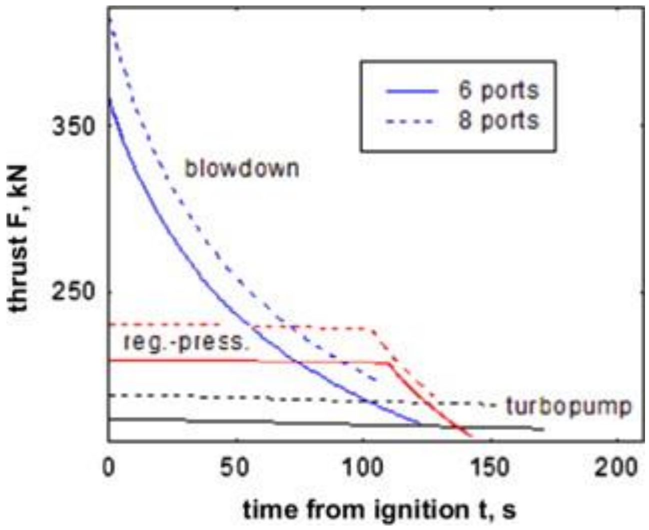

Fig. 11. Thrust history of an optimized hybrid rocket motor.

In order to find the expected range of values of motor design and operation parameters (e.g. thrust level, chamber pressure, mixture ratio, oxidizer mass flow), and to compare performance of different propellant combinations, grain geometries or feed systems, a MDO code is developed by PoliTo.
The optimization procedure aims at finding the motor design parameters and the corresponding trajectory that maximize the required mission performance index, while taking into account all required constraints (e.g. maximum heat flux). The trajectory optimization may be characterized by continuous controls (namely, the thrust direction), which would either require a discretization by means of a large number of parameters or the use of indirect methods. This second approach is more accurate and fast and PoliTo developed a very efficient indirect procedure [44].

On the other hand, the relations, which determine the motor behavior, cannot be written explicitly and indirect methods cannot be used. An adequate ballistic model can be used where the number of motor design parameters is low (e.g. 5 when using oxidizers such as LOX, 3 when using self-pressurizing oxidizers such as N2O), so that their optimization is easily carried out by an evolutionary or a direct method. Details of such a model are not given here for the sake of conciseness and can be found in [37]. According to this model, the design of the HRM is defined by the propellant combination, the initial thrust level *Fi*, the initial mixture ratio α_i, the nozzle expansion ratio ε_i, the initial value of pressure $(pt)_i$, the initial value of chamber pressure $(pc)_i$, the initial ullage volume $(Vg)_i$, and ratio J of the throat area to the initial port area. The chamber pressure is constrained in order to avoid coupling between the hybrid motor and the oxidizer feed system. Moreover the initial port area to throat area ratio J should be as large as possible but not exceed 0.5 to avoid excessive pressure losses and nonuniform grain regression. The remaining design parameters have to be optimized.

In previous works an in-house direct method was used to evaluate motor design parameters for different applications. A nested direct/indirect procedure was used, where the indirect method optimizes the trajectory for each choice of the motor parameters given by the direct procedure. One should note that the direct method is a local optimization method that requires an initial tentative solution, which influences the result of the optimization procedure i.e. the direct method can get stuck on a local optimum.

In order to have a more flexible optimization code, an evolutionary method may be adopted. Evolutionary methods do not depend on an initial tentative solution and are far more flexible as only a reasonable range of variation for each design parameter is required. Also, an evolutionary algorithm is suited to deal with non differentiable discrete variables (e.g. the one describing the propellant combination), and to manage inequality constraints. Then, a new procedure has been developed [57]: the indirect method has been retained, whereas the engine design parameters are instead optimized by means of an in-house evolutionary procedure [58].

Evolutionary optimization uses bio-inspired algorithms in which each individual represents a solution, its genes represent the solution parameters and the objective function determines the individual fitness to the landscape where individuals evolve. The optimization procedure starts with the production, in a random way, of an initial population of individuals, whose evolution through the generations is dictated by the rules of the algorithm and produces the improvement of the objective function. Each method is characterized by different rules used to determine the evolution and the selection of the individuals, and a proper number of parameters must be set for each method. Usually, a proper tuning of these parameters is needed and an optimization of the method itself is required. This tuning is here avoided using a cooperative approach. In fact the in-house procedure employs a genetic algorithm (GA), differential evolution (DE), and particle swarm optimization (PSO) in parallel. According to the island model, each optimizer acts on a separate population and work differently on the same problem. The best individuals found by each algorithm can migrate to the others at prescribed intervals, giving the cooperative algorithm better performance, in terms of efficiency, compared to the basic algorithms.

In order to test the new evolutionary/indirect nested procedure, a simple problem has been considered first: the optimization of a sounding rocket. The initial mass of the sounding rocket is 500 kg, while 100 kg are left for payload and other masses not directly related to the propulsion system. The cost function to be optimized is the time spent above 100 km (microgravity time). The motor grain is cylindrical with a single circular port; a blow down feed system is used. The length to diameter ratio of the rocket has an upper bound of 12 and the propellant tank may have a diameter larger than the fuel grain.

Different propellant combinations can be considered by the code. In the proposed example four propellant combinations, namely HP/PE, HP/HTPB, LOX/HTPB and LOX/SW, are compared. The global population contains 80 individuals while the three subpopulation have 40 (GA) or 20 (DE and PSO) individuals. It has been found that just 10 generations (4 min on a PC with 3.4 GHz CPU) are

required to have quite precise estimation of the optimal performance. The approximation of the optimal design parameters is slightly less accurate and can be improved by increasing the number of generations.
Fig. 12 shows the evolution of subpopulation composition during optimization. The LOX/SW individuals gain a larger population fraction generation after generation and seem to be the most promising solution as this propellant combination presents high regression rate and performance. Fig. 13, Fig. 14 show the evolution of grain port radius and mixture ratio during operation of the optimized motors, respectively. The faster regression of the paraffin grain reduces the required grain length and the propellant tank diameter is similar to the grain outer diameter. Also the mixture ratio shifting is small.

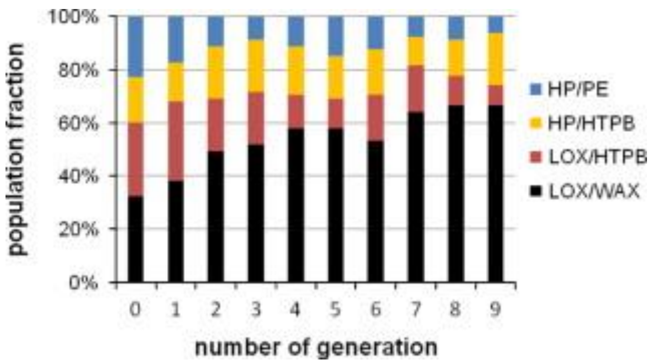

Fig. 12. Evolution of population composition in terms of propellants combination.

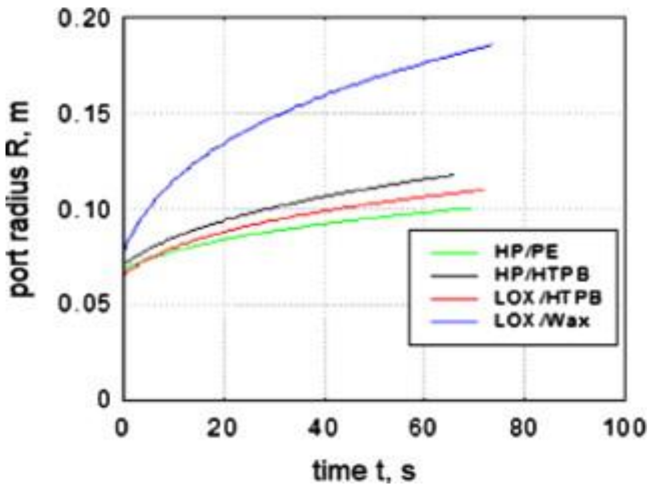

Fig. 13. Port radius history for different propellant combination.

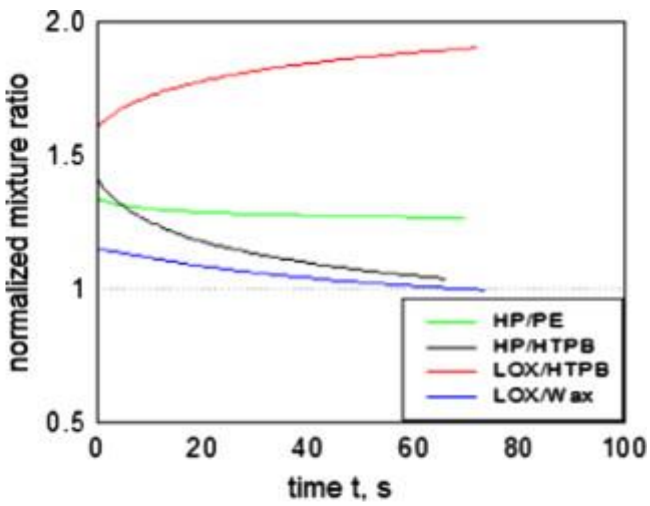

Fig. 14. Mixture ratio (normalized with mixture ratio that maximizes c^*) history for different propellant combination.

The data presented in Fig. 13, Fig. 14 are obtained without considering the influence of the throat erosion. In order to estimate the nozzle weight, the code includes the evaluation of the needed ablative material thickness according to correlations presented in [59]. These correlations are not useful to evaluate the regression rate. Moreover, they refer to solid rocket motor rockets, where the combustion gases are not oxidizer rich such as in hybrid rockets.
The throat erosion rate is then evaluated according to the data provided by UniRoma (see Section 5 and [60]) and the influence of the variation of chamber pressure and throat radius is taken into account. Fig. 15 shows the effect of throat variation on the rocket trajectory, while effects on microgravity time (t_{mg}) and initial thrust level (F_i) are compared in Table 5.

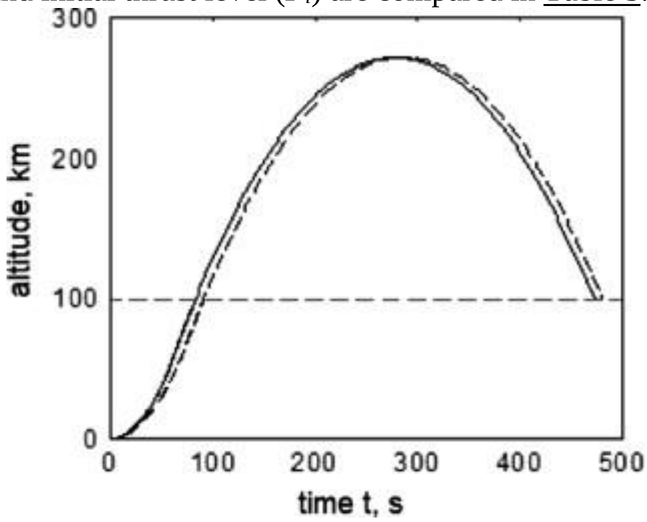

Fig. 15. LOX/SW rocket trajectories neglecting (dashed line) or considering (solid line) throat erosion.

Table 5. Effects of throat erosion on the required initial thrust and microgravity time for different propellant combinations.

Propellants	No erosion		Erosion	
	F_i (kN)	t_{mg} (s)	F_i (kN)	t_{mg} (s)
HP/PE	26.0	299	19.5	287
HP/HTPB	25.8	323	26.1	318
LOX/HTPB	23.7	229	22.6	221
LOX/SW	27.4	390	31.0	389

The throat erosion influences the motor design. The effects on performance are weak, especially for the best-performing LOX/SW combination.

7. Conclusions

This paper presents the overall strategy, the organization, and the first experimental and numerical results obtained in the framework of the Italian PRIN project, devoted to the improvement of hybrid rocket engine technology. The program supports the joined effort of four Italian Universities to contribute to the development of a deeper knowledge of this propulsion technology.
Results show the advancement performed in fuels compositions. Selected formulations, characterized at lab-scale, have been characterized also in motor-scale. A motor installation allowed starting an experimental coordinated program about paraffin-based solid fuels.
The erosion rate and the wall temperature distributions at varying equivalence ratios were investigated for graphite, showing that hybrid motor operation at oxygen-rich mixture ratios and high pressures typically result in very high throat erosion rates, while operation at fuel-rich mixture ratios and low chamber pressure generate very low throat erosion rates. After this general overview on the erosion

behavior of graphite in HRM, attention has been focused on a graphite nozzle with a throat diameter of 9.6 mm, of specific interest in the framework of this project. The results allow to estimate the nozzle total erosion rate and the erosion contributions from the five oxidizing species considered.
Moreover, a new procedure has been developed in order to have a more flexible and fast procedure to perform preliminary optimization of hybrid rocket motors. An accurate indirect trajectory optimizer, previously coupled with a direct method, is now nested with a cooperative evolutionary procedure. The evolutionary approach does not need a tentative solution, and is able to deal with discrete variables (e.g. propellant combination) and constraints of different kinds (e.g. heat flux limits). The maximization of the time spent by a sounding rocket during a ballistic phase above 100 km has been used as a test case. Even if this is not a complex application, many conflicting objectives require compromise values of the motor design parameters. The optimizer is able to select the best propellant combination. LOX/SW is the most performing combination, as the strong regression rate allows a more regular shape of the grain and mixture ratios which are very near the values determining maximum c^*. The procedure proved to be fast and reliable and will be used for more complex applications, such as upper stages.

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