**J-Y. Lestrade, J. Messineo,
J. Hijlkema, P. Prévot, G. Casalis,
J. Anthoine**
(ONERA)

E-mail: Jean-Yves.Lestrade@onera.fr

DOI : 10.12762/2016.AL11-14

Hybrid Chemical Engines: Recent Advances from Sounding Rocket Propulsion and Vision for Spacecraft Propulsion

Hybrid propulsion generally combines a liquid oxidizer with a solid fuel. Compared to solid or bi-liquid engines, this type of propulsion offers advantages in terms of cost and flexibility. For several years, ONERA has been providing expertise in this area, both in experimental and numerical simulations. Various hybrid rocket motors have been designed and tested on ground, including under low ambient pressure conditions and in flight. The Propulsion Laboratory unit of the DMAE uses fully instrumented engine breadboards in order to optimize the oxidizer/fuel pair, as well as the injection system and the combustion efficiency. In addition, 0D and 1D codes for engine design are actively being developed. More recently, studies have been undertaken as part of the LEOP future satellite platforms, with the development of an innovative new combustion chamber.

Hybrid propulsion concept and principle

The development efforts carried out for many years on solid and liquid chemical engines made their use almost exclusively for launcher applications (missiles, commercial launchers, sounding rockets, etc.). However, these two concepts have budgetary, performance and environmental disadvantages (cost, complexity, safety, reliability, environmental impact, etc.). They also hinder the development of new applications, such as space tourism, nano-launchers, micro-gravity experiments or landers. Thanks to its benefits in terms of safety, cost, reliability, environmental impact and performances, hybrid propulsion is a promising candidate to overcome these constraints and hence facilitate the development of these new applications [3,25].

The most common configuration of a hybrid engine (Figure 1) comprises an oxidizer (stored as a liquid and vaporized in the forward dome of the motor or gasified by flowing through a catalyst bed), which flows through long fuel channels and burns with the pyrolysis gases resulting from the solid fuel regression to form a turbulent boundary layer. In this case, the solid fuel is cast in the combustion chamber as for the solid rocket motor technology, whereas the oxidizer is stored in a proper tank as in the liquid propulsion technology. The convective and radiative heat fluxes stemming from the diffusion flame provide the energy needed for the solid grain pyrolysis to sustain heterogeneous combustion [29]. Such a cycle produces, from hybrid combustion, a self-sustained phenomenon that occurs as a macroscopic diffusion flame. This implies a dependency of the fuel regression rate on the heat and mass transfer processes. This stabilized combustion continues for as long as the oxidizer is being injected and the solid fuel grain is regressing. The extinction of the engine can be achieved by closing the oxidizer valve.

A hybrid engine is an innovative propulsion system that has a well-known safety level thanks to its simple structure and the wide range of inert propellants that it can use. It can be throttled, restarted or turned off in case of an abort procedure. It has appreciably high specific impulse associated with reduced costs and a reduced environmental impact, since fuels mainly burn into steam and carbon dioxide [2,23]. These advantages are detailed in Box 1.

For the common configuration and as opposed to solid and liquid chemical engines, the amount of ablated fuel, and thus the combustion chamber oxidizer-to-fuel mixture (O/F) ratio, is directly linked to the heat received by the fuel grain from the reacting flow and thus varies along the length of the motor. Since the flame between the oxidizer and the pyrolysis gases is essentially conditioned by their diffusion towards each other, the regression rate is also mostly dependent upon the mass flow rate within the fuel channel. Thus, the fuel regression rate varies with the mass flux and along the length of the combustion chamber following the relatively simple expression [1,2,33]:

$$\dot{r} = aG^n x^m \quad (1)$$

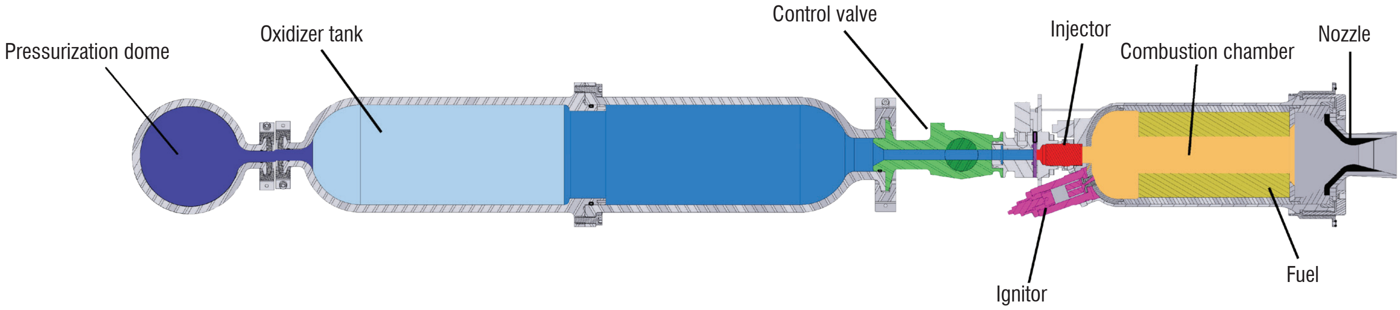


Figure 1 – Hybrid chemical engine concept

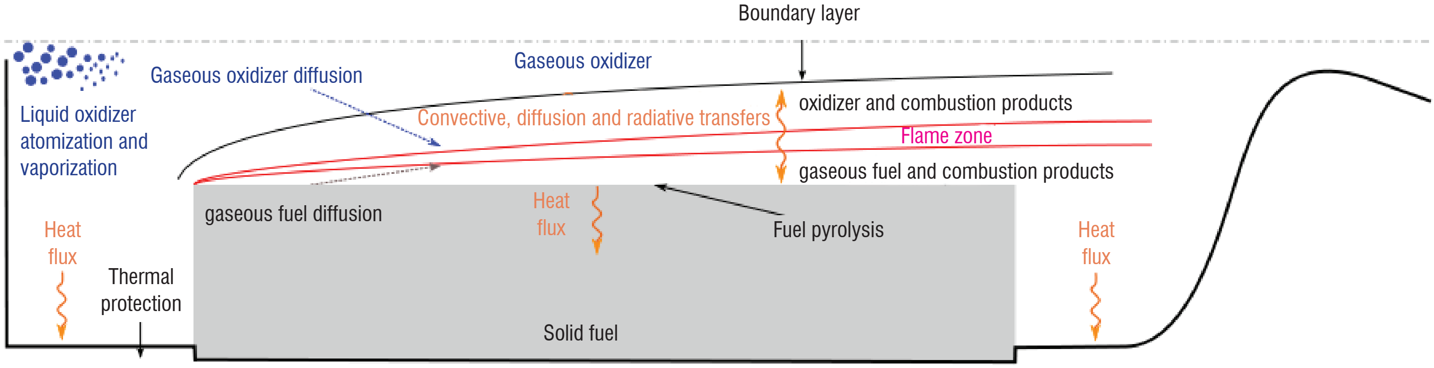


Figure 2 - Hybrid propulsion physics

Box 1 - Advantages of hybrid propulsion

Safety

Inert manufacturing / transportation / assembly / storage
Safe launch-pad operations
Reduced risk of catastrophic failure
Abort by engine shutdown

Reliability / Operating flexibility / Simplicity

Engine verification prior to lift-off
Tolerant to grain defects
Thrust modulation for aerodynamic load relief
or mission optimization
Start-stop capability
One liquid, simplified injector set-up

Costs

Reduced development costs
Reduced recurring costs
Safe materials

Environmental impact

On pad and in flight, very reduced

where $\dot{r}$ is the fuel regression rate, G is the total propellant mass flux, x is the distance down the combustion chamber and a, n and m are the regression rate constants, which are characteristic of the propellants and of the combustion chamber geometry. Since G includes the mass flow from both the injected oxidizer and the fuel that has vaporized from the surface of the fuel, it increases continuously down the combustion chamber. The solid fuel regression rate is one of the most important values in the hybrid conceptual design process used to determine the propulsive performance.

The combustion phenomenon is similar to that of a turbulent diffusion flame for which the flame zone is established within the boundary layer [32] (Figure 2) and results from the coupling of the:
- Kinetics of the condensed phase pyrolysis;
- Homogenous combustion mechanism in the gaseous phase;
- Convective and radiative heat transfers in the gaseous phase;
- Mass transfer of the chemical species.

This process can be treated by an idealized model that considers the flame zone as a point of discontinuity in the temperature gradient and composition. Actually, the flame zone is thicker and both the oxidizer and the pyrolysis gases from the fuel enter the flame zone by diffusing towards each other through the boundary layer. The combustion zone is established at the point where an approximate stoichiometric mixture ratio has been achieved.

Hybrid chemical propulsion physics includes many complex phenomena: fluid dynamics coupled with combustion, turbulence, spray atomization and vaporization processes, soot formation and radiation, fuel surface pyrolysis and fuel liquid film (for liquefying fuels, see later). The knowledge of the complex interactions between these wide-ranging physical phenomena is fundamental for hybrid motor design and performance prediction.

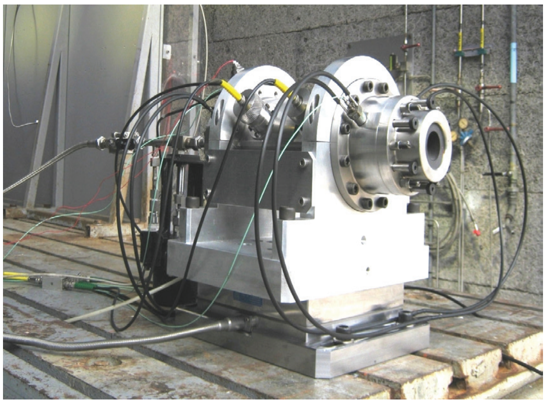
Modeling physical phenomena in the combustion chamber

The design of a hybrid engine generally requires lab-scale firing tests to characterize the regression rate as a function of the engine operating conditions, the propulsive performances, etc. In order to limit these costly experimental campaigns, reliable numerical simulations are required. These simulations range from one-dimensional representations of the gaseous flow to three-dimensional and time resolved simulations of the internal geometry of the combustion chamber, with the degree of modeling complexity depending on the desired application. Thus, one-dimensional codes are preferred during the preliminary design phases, in order to be able to carry out many calculations in a relatively short time. The full Navier-Stokes simulations, providing much more details but being very costly in computation time, are, in turn, generally used for optimization steps and for the detailed understanding of physical phenomena. These two approaches need to be combined with experimental tests in order to validate the models and/or provide input conditions for the numerical calculations.

Description of a typical lab-scale facility

In order to validate the models and the numerical simulations, experiments are still required on instrumented lab-scale engines such as the HYCOM facility (Figure 3, left). Like most lab-scale hybrid rocket motors, the HYCOM facility is composed of five parts (Figure 3, right): a forward end-plate including the injector, a pre-chamber including the igniter, a combustion chamber, a post-chamber and a nozzle. This facility was designed by making the different parts modular, to easily change the lengths of the pre- and post-chambers, the geometry and the type of fuel grain, etc. Some tests were performed with a catalytic injector, combining a catalyst bed and a gaseous swirl injector, instead of the classical atomizer [26]. The engine, which can operate at up to 7.5 MPa, is instrumented with a Coriolis mass flow meter for the oxidizer and four pressure probes (two in the pre-chamber and two in the post-chamber), and is connected to a thrust sensor to achieve the propulsive performances. In order to measure the fuel regression rate, the engine is also instrumented with ultrasound sensors (one located at the head-end of the fuel grain and two at the rear-end). This technique was initially developed for solid rocket applications and adapted for lab-scale hybrid engines [7,9,42]. It has the advantage of being non-intrusive and easily implemented, unlike visualization and X-rays measurement techniques.

Several firing tests have been performed and a large database is available for various configurations: oxidizer/fuel pair, fuel length and initial port diameter, oxidizer mass flux, nozzle throat diameter, etc. A typical result is presented in Figure 4. In this configuration, a catalyst injector was used [26].



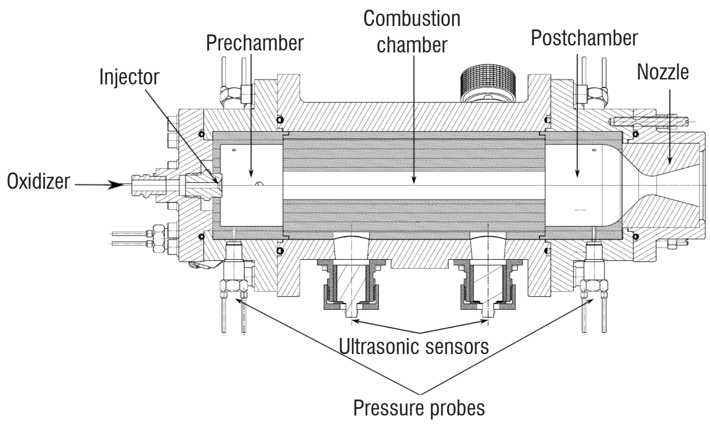


Figure 3 - HYCOM lab-scale engine (left) and schematic view of this facility (right)

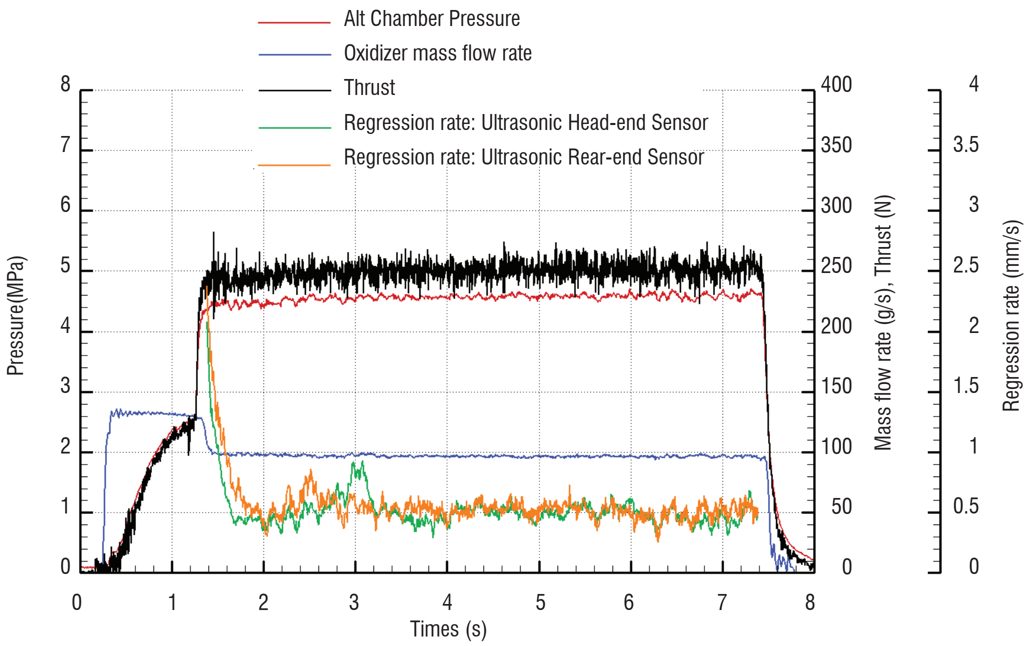


Figure 4 - Typical result obtained with the HYCOM facility

1D modeling

The objective of mono-dimensional codes is to provide a quick estimation of the propulsive performance of a hybrid engine as a function of the operating conditions, such as the oxidizer/fuel pair, the oxidizer mass flow rate, the dimensions and geometry of the fuel grain and the nozzle characteristics. Such codes also give an estimation of the average properties of the gaseous flow and of the fuel regression rate.

The main part of these numerical codes concerns the treatment of the flow inside the fuel channel. Since velocity gradients pilot heat and mass transfers, an aerothermochemical model based on an integral description of the flow in the combustion chamber seems more appropriate to describe the gaseous phase. The boundary layer integral method consists in integrating all of the transport equations along the normal direction of the fuel grain [27,33]. However, two configurations must be distinguished, according to the boundary layer development in a channel. For the first one, the boundary layer does not fully extend into the channel and an inviscid zone must be treated apart from the boundary layer zone while, for the second one, the boundary layer is fully developed and the confined fluid obeys a Poiseuille flow. It must be noted that this second configuration is only encountered for a high L/D ratio. With closure relations, the modeling of the gaseous flow provides the physical data (mass and thermal fluxes, etc.) at the gaseous/fuel interface to determine the fuel regression rate. This part has to be treated with two different approaches, according to the nature of the fuel grain.

For classical polymeric fuels, such as HTPB or polyethylene, the regression rate is modeled by an Arrhenius law (Equation (2)). This equation has to be combined with the energy balance at the solid/gas interface, which involves a balance between the convective, diffusive and radiative fluxes on the gas side and the diffusive and heat of pyrolysis fluxes on the solid fuel side. Finally, these two relations are closed, as described by Sankaran [43], due to the one-dimensional thermal conduction in the solid fuel, since its derivative form expresses the conduction heat loss in the fuel grain required in the energy balance at the solid/gas interface.

$$\dot{r} = A.\exp\left(\frac{-E_a}{RT_s}\right) \quad (2)$$

where A is the pre-exponential factor, E_a is the activation energy, R is the universal gas constant and T_s is the fuel surface temperature. The constants A and E_a only depend on the thermochemical properties of the fuel and are determined from fundamental experiments without reactive flow.

Although classical fuels have been largely studied, they suffer from a low regression rate (around 1 mm/s), which is detrimental for applications requiring high thrust values such as sounding rockets. Several studies were conducted in order to increase this regression rate value (addition of metallic particles, improvement of convective transfers between the gaseous flow and the solid fuel, etc.), without success [6,13,24,35,41]. A significant improvement was obtained when liquefying fuels, such as paraffin-wax, were experimentally tested by Karabeyoglu [17], providing regression rates two to four times higher than the values encountered with conventional fuels. Contrary to classical polymeric fuels, liquefying fuels pass into gaseous phase by forming a thin and hydro-dynamically unstable liquid layer, which is convected, vaporized and atomized, leading also to an increase in the fuel regression rate (Figure 5). The physics inside the combustion chamber thus differs from that encountered with classic fuels and, consequently, a new model had to be developed for the liquefying fuel regression rates.

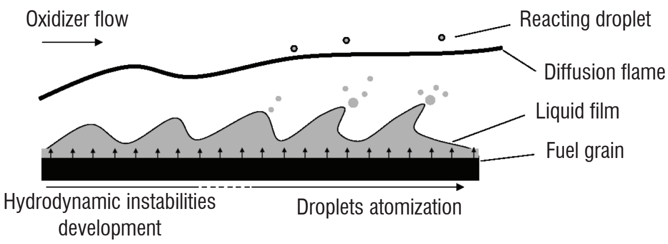


Figure 5 - Karabeyoglu's schematic diagram of the entrainment process for liquefying fuels [17]

As presented by Lestrade [27], such a model is based on a single equation model of the liquid film (Equation (3)). Combined with the momentum and energy equations in this layer and with the one-dimensional thermal conduction in the solid fuel (Equation (4)), this model provides the liquid film thickness, which determines the heat and mass transfers between the solid, liquid and gaseous phases. The mass transfer from the solid phase corresponds to the fuel regression rate.

$$\frac{d\rho_l \overline{u_l} h_l}{dx} = Q_{melt}\left(1-\frac{\rho_l}{\rho_s}\right) - Q_{vap} - Q_{atom} \quad (3)$$

where h_l is the liquid film thickness, $\overline{u_l}$ is the cross average liquid film velocity, ρ_s and ρ_l are the solid and liquid densities and Q_{melt}, Q_{vap} and Q_{atom} are the molten, vaporized and atomized mass fluxes.

$$T_s(y) = T_{amb} + (T_{melt} - T_{amb})\exp\left(\frac{Q_{melt} c_{p,s}}{\lambda_s} y\right) \quad (4)$$

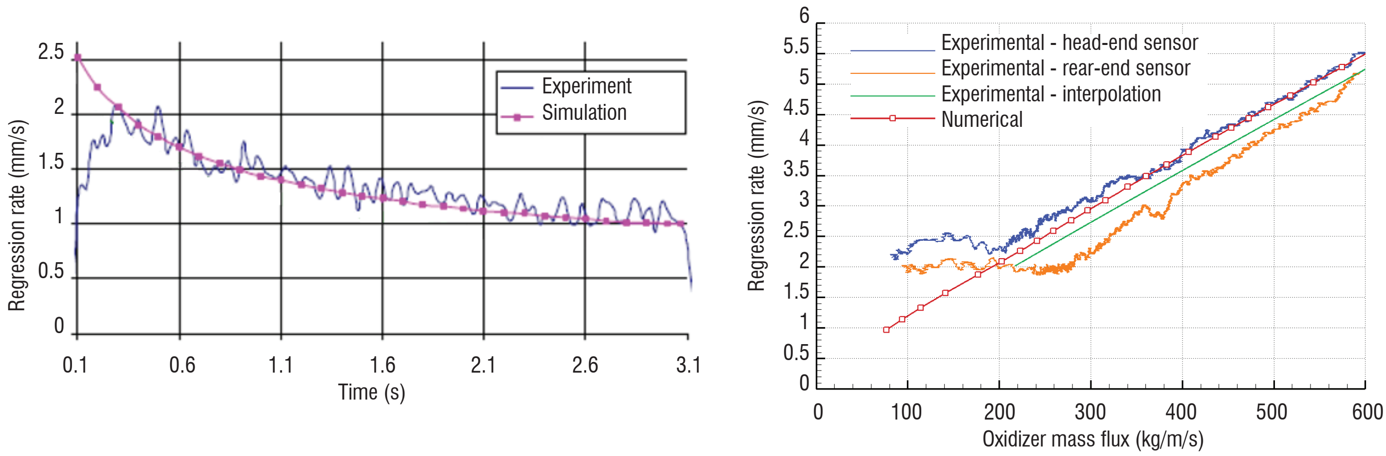


Figure 6 - Comparison between experimental and numerical regression rates for classical fuel (left) [3] and liquefying fuel (right) [27]

where T is the solid grain temperature, T_{amb} and T_{melt} are the ambient and the melting temperatures, $c_{p,s}$ is the solid heat capacity and λ_s is the solid heat conductivity.

As presented in Figure 6, the model developed to determine the regression rate for both classical and liquefying fuels is in agreement with experimental values obtained using HYCOM-like facilities. These two one-dimensional codes can thus be used to provide a good estimation of hybrid engine propulsive performances as function of their size and operating conditions. Nevertheless, the use of these numeral codes is limited to the quick dimensioning of hybrid engines and does not help to understand the internal physical phenomena occurring in the combustion chamber. 2D and/or 3D CFD simulations are required.

Computational Fluid Dynamics for hybrid engines

Numerical simulations are used to improve the comprehension of the internal flow and predict the behavior of the engines. Cheng et al [10] developed a 3D RANS model including, respectively, a Lagrangian approach for liquid droplet transport and an Eulerian approach for the gaseous stream. Surface pyrolysis based on an Arrhenius law has been used by Sankaran and Merkel [44] and by Serin and Gogus [45]. Sankaran [43] made an important contribution to hybrid flowfield calculations. His model is based on the time-dependent Navier-Stokes equations, including turbulence, gas-phase radiation, a finite-velocity model for the combustion, and a coupling between the gaseous and solid-fuel phases based on Relation (2).

Numerical simulations have recently been performed at ONERA [31] using the CFD finite volume code CEDRE [36] to numerically analyze an experimental firing test performed with decomposed hydrogen peroxide as oxidizer and high density polyethylene as fuel. These simulations are based on the Unsteady-Reynolds-Averaged Navier-Stokes equations with an additional transport equation for the species. The k-ωSST model [30] is used for turbulence, combining the k-ω model close to the walls and the k-ω model in the external flow. The gaseous phase reactions are defined by a two-step mechanism (Equation (5)). As done by the authors mentioned above, the chemical reaction rates are modeled by an Arrhenius law whose coefficients are given by Westbrook and Dryer [46]. Reactive phase flows are generally modeled by simplified mechanisms. Thermal radiation is not taken into account.

$$\begin{aligned} &C_2H_4 + 2O_2 \rightarrow 2CO + 2H_2O \\ &CO + 0.5O_2 \leftrightarrow CO_2 \end{aligned} \tag{5}$$

The computation geometry, considered to be 2D axisymmetric, is fixed at the initial instant. The flow is completely single-phase and the walls are assumed to be adiabatic. The fuel boundary condition is treated as an injecting wall (no slip condition). The fuel mass flow rate is constant and defined based on the measurements from the baseline firing test [31], since the regression rate model (Equation (2)) is under development in CEDRE. When such a law becomes available, the model will provide a local and temporal estimation of the fuel mass flow rate, including its possible fluctuations. The oxidizer injection mass flow rate and temperature are imposed from the baseline test measurements.

The comparison between numerical and experimental results is based on the internal pressure. Numerical sensors have been placed at the same locations as the pressure probes in the experimental facility, which enables the experimental pressure to be compared to the simulated pressure and their Fast Fourier Transform. The 2D U-RANS computation gives a mean pressure of 4.32 MPa compared to the experimental pressure of 4.15 MPa. This difference may be explained by various factors: firstly, the real injection geometry is approximated (holes were replaced by axisymmetric rings). Secondly, the boundary condition for the fuel mass flow rate is uniform along the grain and the thermal losses are not taken into account in the simulation.

According to the ratio between the turbulent and the classical viscosities, which changes from 20 to about 500, and to the Reynolds number based on the fuel port diameter, which is around 150 000, the flow can be considered as fully turbulent. The diffusion flame is consequently located within the turbulent boundary layer (Figure 7). The oxidizer is concentrated along the engine axis and the gaseous flow is accelerated due to the total mass flow rate increase along the fuel grain.

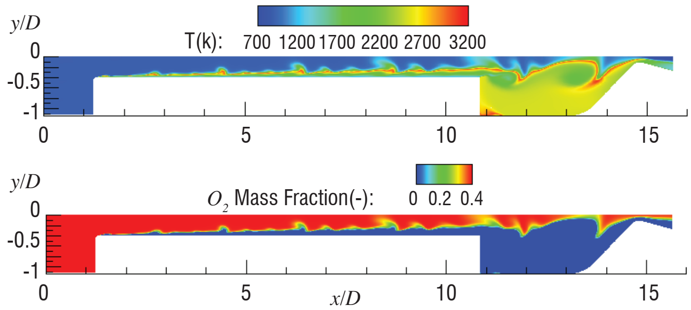


Figure 7 - Temperature and oxidizer mass fraction fields

The correspondence between the fuel mass fraction and vorticity fields shows that large-scale vortices containing unburnt fuel are formed at the end of the fuel grain (Figure 8). These coherent structures seem to play a major role in the mixing in the post-chamber, since they transport unburnt fuel from the end of the grain to the nozzle.

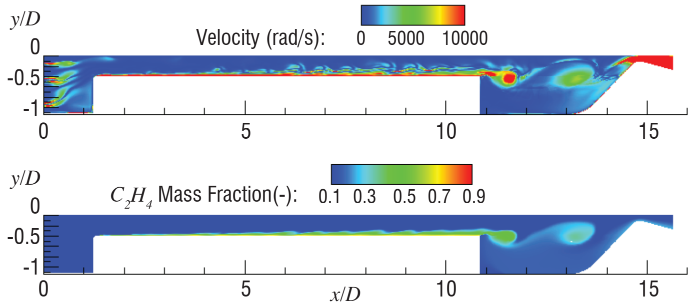


Figure 8 - Vorticity and fuel mass fraction fields

Crossing the nozzle throat, these structures lead to pressure fluctuations in the combustion chamber, which can be visualized by calculating the FFT on the pressure signals (Figure 9). A large peak at 470 Hz is observed for the firing test and at 530 Hz for the simulation. The numerical amplitude of the pressure instability is close to 2.5% of the mean pressure in comparison to the 1.9% during the experimental firing test. The results are close, even though the computation presents pressure instabilities with slightly higher frequencies and magnitudes than the firing test. A more detailed analysis of this instability is presented in the next part.

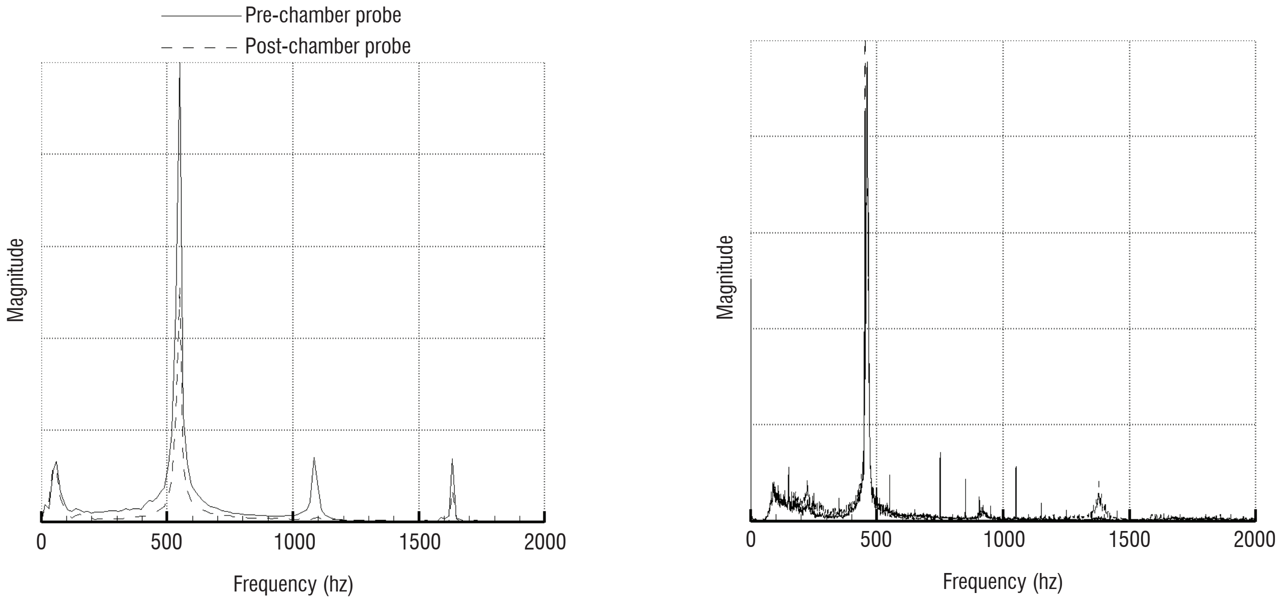


Figure 9 – Numerical (left) and experimental (right) FFT on the pressure signal

Instabilities

Several kinds of instabilities are possible in hybrid engines and may be provoked by acoustics, hydrodynamics or combustion. Low frequency instabilities have been studied by Karabeyoglu et al [16,18] and combine thermal lag in the solid grain, combustion and gas dynamics. Carmicino [8] studied the interaction between different instabilities and revealed the importance of vortex shedding for hybrid rocket motors. Recent studies showed the formation of small-scale vortices in the near wall region leading to small variations in the pressure amplitude [19,20]. Acoustic instabilities are generally related to classical longitudinal modes or to the Helmholtz modes [11].

In this study (Figure 9), the main instability is hydrodynamic and provoked by vortex shedding at the end of the fuel grain, as described by Carmicino [8]:

$$f = \eta c_{th}^{*} \psi^{2} S_{D} \frac{D_t^2}{D^3} \tag{6}$$

where η is the combustion efficiency, c_{th}^{*} is the theoretical characteristic velocity, ψ is a function of the isentropic coefficient, S_p is the Strouhal number based on the end-port diameter D, and D_t is the nozzle throat diameter. The Strouhal number varies in the range of 0.25 to 0.5 for free and ducted non-reacting jets, which could lead to the frequency range of 440-880 Hz. The relation provides a good order of magnitude, but the estimated frequency is strongly dependent on the Strouhal number, a priori unknown.

Another approach, based on the Rossiter vortex shedding theory in a cavity [40], is used to provide an approximation of the main frequency peak. The vortex impingement on a surface generates an acoustic pulse traveling back upstream and generating a new vortex at the shear layer initiation point. For small Mach numbers, it can be expressed as [37]:

$$f = \frac{U}{L}\left(\frac{m-\alpha}{M+1/k}\right) \tag{7}$$

where L and m are respectively the distance and the number of vortices between the shear-layer initiation and the impingement point, U and M are the freestream velocity and Mach number, k is the ratio of the vortex transport velocity to the freestream velocity and α is a dimensionless empirical constant. The empirical model leads to the range of 380-510 Hz for the vortex shedding frequency, which is in agreement with the observed peak at 470 Hz.

Finally, a low frequency shift ranging from 470 Hz to 455 Hz was observed during the firing test. This shift was explained based on the Rossiter's approach considering the variations of the end-port diameter as inducing a variation in the main stream velocity [31].

Application to sounding rocket propulsion

Hybrid rocket engines are not competitive for the first stages of large launchers. However, they could become competitive for the third stages of such launchers, or for new space missions, such as sounding rockets for Cubesat launches, mars landers, space debris deorbiting systems, spacecraft propulsion, missile systems, etc. [28]. The application to sounding rockets is discussed in this section, while a vision for spacecraft propulsion is presented in the last section.

LEX sounding rocket

The earliest work on hybrid rockets was conducted in 1933 by Tikhonravov and Korolev. With the GIRD-9 sounding rocket based on LOx and jellified gasoline, they reached an altitude of 1.5 km. In the late 1930s, both Germany and the USA started also developing hybrid engines and a rocket was taken to an altitude of about 9 km [14] in June 1951.

Intensive research on hybrid propulsion started at ONERA as from 1956, at the initiative of Barrère and Moutet [4,5,12]. The program, called LEX for Lithergol Experimental, was aimed at developing an autonomous rocket with reduced technological developments and a low-cost production. Six years later, ONERA investigated all aspects of hybrid propulsion and developed the MT.27 hybrid engine using a hypergolic propellant based on nitric acid and an amine fuel. The combustion time of the engine was between 30 and 35 s for an operating pressure of 75 bar and a thrust of 10 kN, providing a total impulse of 100 kN.s. The MT.27 hybrid engine was integrated into the LEX sounding rocket. The first flight (LEX.01) took place in April 1964 and was the first flight of a hybrid rocket in Europe. It was limited to 12 s of operation by reducing the oxidizer mass, in order to avoid reaching a populated area. The launch was aimed at comparing flight data to ground firing tests and the performances were found to be identical. This first test was followed by three successful flights in 1965 and four other flights in 1967, both with the LEX.02. Figure 10 shows the drawing and main features of the LEX02 rocket and a launch from the Isle of Levant. The last flight exceeded an altitude of 100 km, with a rocket weighing 80.5 kg and a meteorological payload of 9.5 kg, which was a record for that time. An instrumented parachute, dropped at the apogee, allowed wind measurements during the 31 min of the descent. The flight tests of 1967, which concluded the LEX program, proved that a hybrid engine with modulated thrust could be cheap and that the performance of this hybrid engine combined with its simplicity make it the best candidate for sounding rockets.

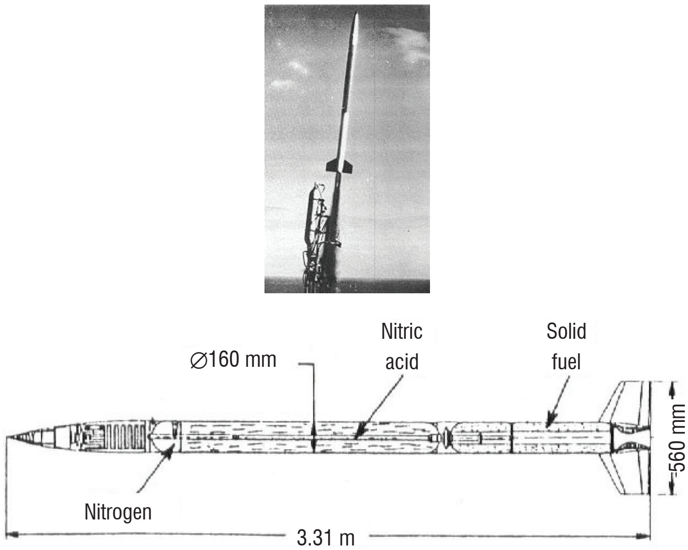


Figure 10 - LEX02 sounding rocket section (bottom) and flight test (top)

PERSEUS project

The PERSEUS project (Projet Etudiant de Recherche Spatiale Européen Universitaire et Scientifique / University and scientific European space research student project) is an initiative of the Launch Directorate of the CNES, the French Space Agency, to promote the emergence of innovative technical solutions for space projects, accomplished by students [34]. The main objective of the project is to make a set of ground and flight demonstrators as a precursor to a nano-satellite (from 10 to 50 kg) launch system.

ONERA's propulsion laboratory has been actively participating in this adventure from the beginning, particularly through the development of a small hybrid engine meant to propel experimental rockets developed by students and flown during the annual C'Space meeting [35]. This FUSEX engine in its original form consists of:

- a 3.5 liter reservoir;
- a pyrotechnical guillotine valve;
- a titanium combustion chamber;
- a graphite convergent and nozzle.

The self-pressurization capacities of N_2O allow an extremely simple concept without pumps or supplementary pressurization systems.

Thrust modulation

For its fifth flight on a rocket named Amidala and developed by S^3 (the student rocketry club involved), the FUSEX engine has been equipped with a two-stage valve upstream from the guillotine valve. This allowed for an in-flight variation of the oxidizer mass flow and hence for a thrust modulation during flight [15]. During the take-off phase, the valve is completely open and the oxidizer mass flow is unobstructed. At a pre-defined instant, a linear motor retracts and releases a notch. The spring loaded valve flips and partially obstructs the flow. Both the instant of modulation and the reduction level are variable, but defined before flight. A possible, future, modification would be to replace the spring and notch mechanism with a stepper motor, to allow for continuous modulation during the entire flight. Several cold ground tests have validated the concept.

Encouraged by a successful first test run with half of the oxidizer charge (1.4 kg of N_2O), we repeated the run, this time with a full oxidizer tank (2498 g) and we delayed the valve timing to 2.5 s with a reduction of 92% of the passage surface in order to obtain a thrust reduction of around 50%. Figure 11 (left) gives the results of this run and has confirmed that the design was working and resulted in a 59% thrust reduction, a total impulse of 7545 N.s and a specific impulse of 227 s. We decided to keep the same configuration for the certification firing (needed to obtain flight authorization from the organization of the C'Space event). Figure 11 (right) yields the results of this final test run and once again confirms the flawless and reproducible functioning of our Thrust Modulation Module (TMM). This last run resulted in a 57% thrust reduction, a total impulse of 7609 N.s and a specific impulse of 220 s. The variation in impulse and thrust values is due to the sensibility of our self-pressurized oxidizer

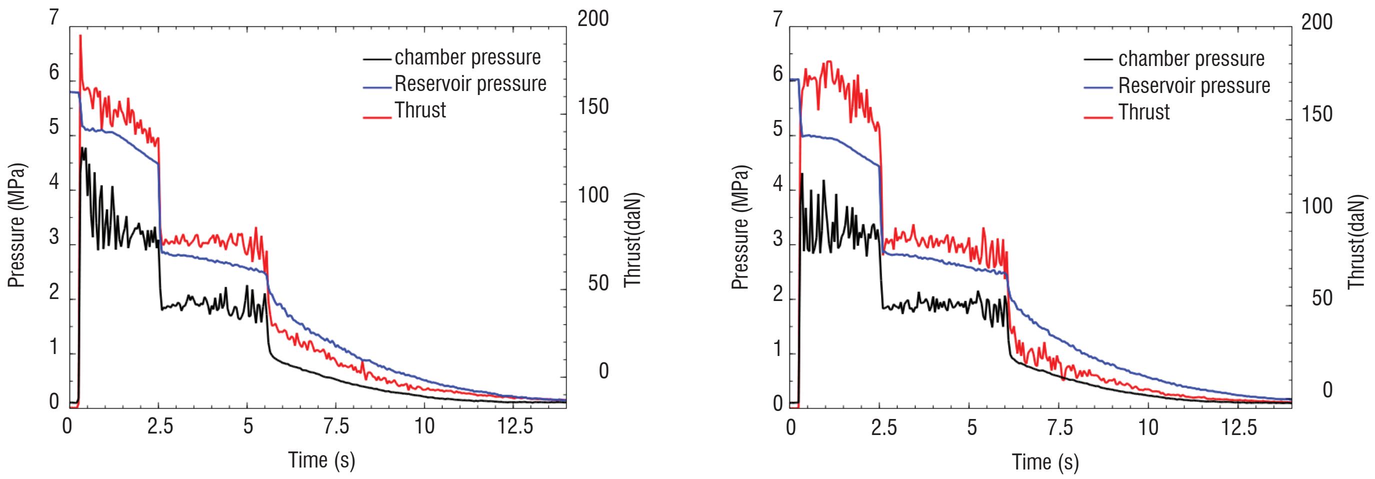


Figure 11 – Test run (left) and qualification run (right)

system to the external temperature. This is a drawback that proved to be rather severe during the launch.

On August 21st, 2011, students, rocket and engine were taken to the C'Space site near Biscarosse. There, the final assembly and tests were completed before the launch date, set for August 24th. One of the C'Space regulations states that the recovery system timing has to be fixed to a certain period of time after the rupture of the umbilical. The period is to be determined with a performance simulation, using a spreadsheet simulation provided by Planètes Sciences called Trajecto. This works reasonably well for solid propulsion, where engine performances are more or less reproducible. However, for our self-pressurized hybrid engine this is a problem, since the engine performance depends significantly on the external conditions, especially the ambient temperature at the time of the launch. This played a major role in the events that followed.

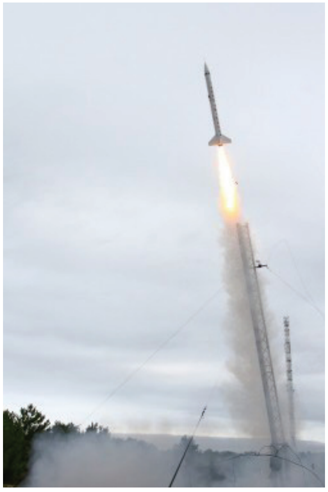


Figure 12 - Lift off (courtesy of Julian Franc (Planètes Sciences))

On the 24th , we were ready to go but already at dawn we noticed that the weather god was not on our side; it rained and instead of the heat that we had suffered from during the preparation phase we measured a meager 17°C. So much for our predictions! We only managed around 50 bars of reservoir pressure, which is at least 10 bars less than for the qualification test (Figure 11). The water also hindered the packing of the parachute, adding insult to injury. Despite the weather, we managed to get Amidala ready on the launch pad at 10:15. Several seconds later, with a roar that is typical for hybrid engines, Amidala freed itself from the launchpad and hurled into the overcast skies. About 2.5 seconds after lift-off a sudden change in pitch indicated that the TMM had kicked in. About 20 seconds later, Amidala pierced the clouds and disappeared from sight. We all waited for the distinct thud of the parachute opening but instead we only heard an increasing whining followed by a bang as Amidala hit the ground. It was clear that the recovery system had failed and that our rocket had gone ballistic. Due to the lower tank pressure, performances were diminished and the rocked reached apogee too soon. Therefore, when the deployment timer triggered the release mechanism, the air speed was too high and the ejection mechanism not strong enough to overcome the dynamic pressure; the destruction was quasi complete. Apart from the on-board camera, which miraculously survived the crash, all of the payload was wrecked leaving us with only the down-linked data (limited to reservoir pressure and temperature) and the Pitot pressures stored in the nosecone but valid only up until separation.

Given the limited data, the reconstruction of the flight was difficult and approximate at best. Coupling data from the high-speed camera at lift-off, the telemetry and the Pitot measurements resulted in the reconstructed thrust curves given in Figure 13. We see that there is a rather large difference in the maximum thrust found through the various methods. This is certainly due to the big differences between the conditions of the flight and those of the ground-based experiments that served to convert the flight data. However, it is very clear that the objective, that is, an in-flight modulation of the thrust, was met. This makes Amidala's flight a European first.

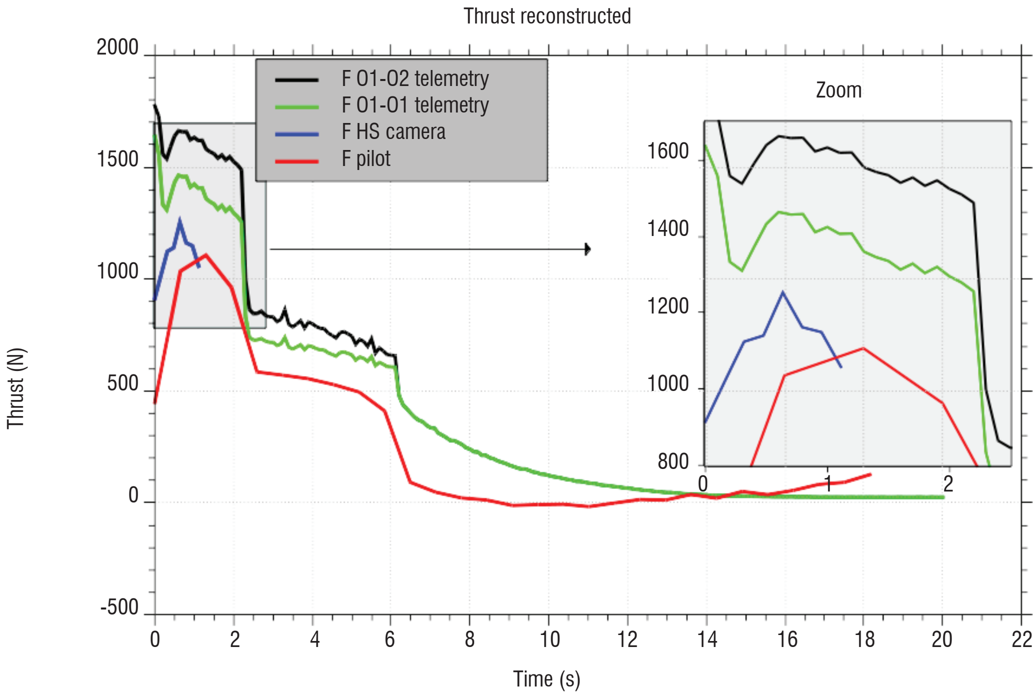


Figure 13 - Reconstruction of the thrust evolution

Vision for spacecraft propulsion

Although some hybrid rockets were tested in flight [15,22,34,35,38,39], thrust levels achieved with new fuels (liquefying and additive-doped classical fuels) still fall short of those provided by solid propellants and, consequently, hybrid propulsion cannot currently be competitive compared to solid propulsion. Nevertheless, thanks to its low regression rates and due to theoretical propulsive performances comparable to those of liquid engines, hybrid propulsion could be competitive with regard to liquid technology for missions requiring long firing duration, moderate thrust levels, flexibility (thrust modulation) and good propulsive performances (specific impulse and characteristic velocity).

Within the framework of the NEOSAT program, a new European platform for telecommunication satellites is under development, based on full electrical propulsion. Combining a chemical engine with the electric propulsion could be a very efficient alternative or complement (as an intermediate carrier) by reducing the GTO to GEO transfer duration from several months (full electrical) to few days (electrical / chemical). Compared to the current bi-liquid engine based on monomethylhydrazine (MMH) and nitrogen tetroxide (N_2O_4), hybrid propulsion offers positive expected impact at system level in terms of dry mass, performance and recurring cost. Indeed, with a theoretical specific impulse of the order of 300 s in a vacuum [28], hybrid propulsion is slightly less efficient than the current bi-liquid engines (320 s specific impulse in a vacuum and a few daN thrust). However, its design is simpler, and thus expected to be cheaper and lighter, and the density specific impulse is higher (fuels in solid phase generally have higher densities than those in liquid phase) reducing the overall system volume. Last but not least, hybrid propulsion is REACh (Registration, Evaluation and Authorization of Chemicals) compatible, thereby facilitating its implementation, and has throttling and restart capabilities to perform the satellite orbital injection.

However, the aforementioned theoretical specific impulse of 300 s was never experimentally proven from vacuum firing tests, mainly because of low combustion efficiency of this technology. Secondly, a sustained long duration burn (of the order of hours) with medium thrust is far from the specifications of a rocket propulsive phase and would require a dedicated design. These new mission requirements have a direct impact on the solid fuel regression rate in order to limit the size of the solid fuel combustion chamber. Regression rates of less than 0.1 mm/s would be needed, but these values are one order of magnitude below the regression rates obtained with polymeric fuels. The risk of combustion instabilities at such small regression rates is not negligible and neither is the risk of self-extinction of the engine. Another disadvantage of the conventional hybrid engine configuration (Figure 1) is the variation of the oxidizer-to-fuel ratio (O/F) during operation, since the fuel burn area steadily increases as the burn progresses, resulting in variations in the specific impulse. This is especially limits the application of hybrid propulsion for the satellite apogee engine when considering the conventional engine configuration. This is not the case for solid rocket engines, in which the stoichiometric ratio is obtained by the correct mixing of the oxidizer and fuel directly cast in a solid powder. For bi-liquid engines, the stoichiometric ratio is guaranteed by using controlled injection valves and turbo-pumps.

In order to demonstrate the vacuum specific impulse, ONERA developed and qualified a new research facility for high-altitude testing. Due to time and budget constraints (only four months to design and put into service the vacuum test facility), an existing low-density wind tunnel was recycled into an 18 m^3 "vacuum" test facility (Figure 14, top) and adaptations were made to make it compatible with hybrid propulsion testing. A vacuum pump was used to provide an initial pressure of 1 mbar in the vacuum chamber. However, since the vacuum pump did not run during firing tests, the pressure inside the vacuum facility constantly increased during a test due to the exhaust gas, restricting the test duration. Wall passages through flanges of the vacuum chamber were set up to enable the cabling of the instrumentation connected to the engine, in order to operate the engine and to feed the engine with the oxidizer. The HYCOM hybrid engine (Figure 3) was modified with a new nozzle adapted to low ambient pressure. Two pressure taps were added on either side of the nozzle exit section, one inside the nozzle and one outside (Figure 14, low). Since the ambient pressure constantly increased during firing tests, these two measurements enabled us to know when the nozzle was adapted during the test.

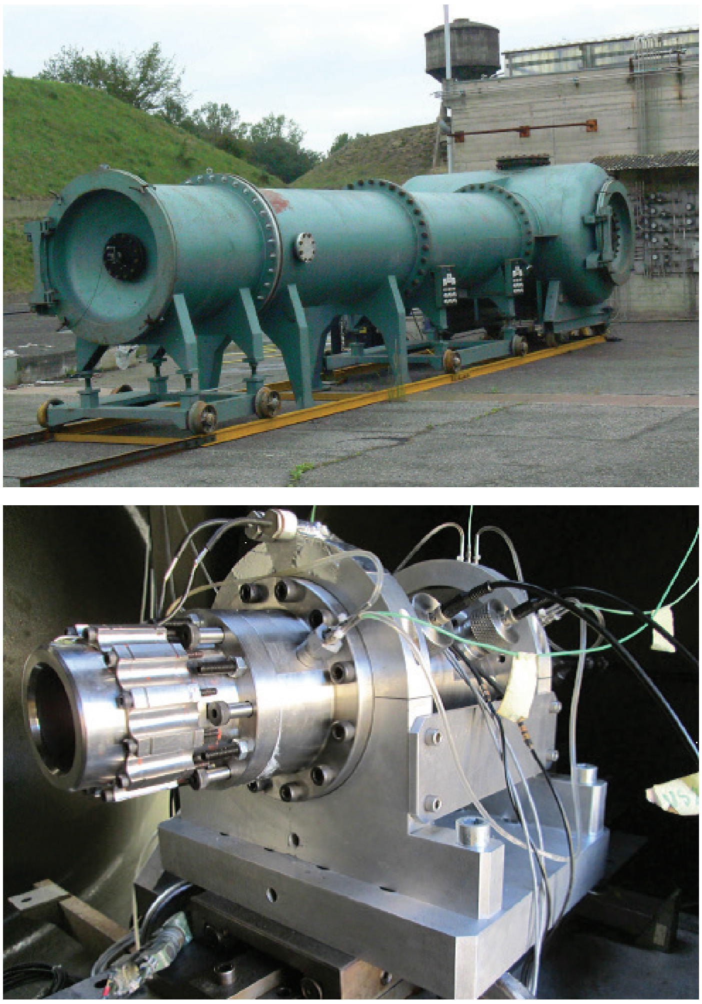


Figure 14 - Research facility for high-altitude testing

The expected performance of hybrid propulsion under satellite environmental conditions was assessed experimentally. The parameters of the hybrid engine have been selected to operate close to the stoichiometric oxidizer-to-fuel ratio. The first test campaign in the new test facility, based on a nozzle expansion ratio of 50, resulted in a combustion efficiency of the order of 85% and a specific impulse of around 230 s. The deficit in the specific impulse performance is mainly due to the low value of the combustion efficiency. According to the state-of-the-art in hybrid propulsion, how the oxidizer is injected strongly influences the engine performance, which can be explained by the role of the injector in the swirling and turbulence properties of the injected flow and in the atomization of the oxidizer.

The type of injector is the main parameter to increase the combustion efficiency. The first campaign involved a swirl injector providing coarse droplets. When using a swirl injector generating very fine droplets, the combustion efficiency was enhanced by 6%. Another way to increase the combustion efficiency is the catalytic injection of the oxidizer. Indeed, the injection of a hot gaseous oxidizer is expected to improve the mixture with the pyrolysis gas coming from the fuel grain and avoid using a part of the generated heat flux to vaporize and warm the liquid oxidizer. The catalytic decomposition of H_2O_2 also enables the combustion to be initiated without a pyrotechnic igniter.

Within the framework of a collaborative project between ONERA, Nammo and Saab, supported by the NEOSAT program, ONERA has recently plugged the Saab catalyzer and the Nammo swirl injector into the ONERA hybrid engine, whose instrumentation enables the instantaneous evolutions of the propulsive performances to be deduced [26]. Three firing tests for an oxidizer-to-fuel ratio close to the optimum one were performed: the two first tests at atmospheric pressure and the third one under low pressure conditions in the high-altitude test facility. The three firing tests have provided a mono-propellant phase of 1 s followed by a hybrid mode of more than 6 s (Figure 15). Thanks to the pressurization system of the oxidizer tank, the oxidizer mass flow rate was quite constant and very close to 100 g/s during the hybrid mode, which provided a chamber pressure of 4.5 MPa. The thrust for the tests at atmospheric pressure was equal to 250 N, while it achieved 300 N for the test under low ambient pressure conditions.

The experimental results from the three firing tests were analyzed following two methods. The first one, based on averaged quantities, showed a very good repeatability of the three firing tests in terms of chamber pressure, oxidizer mass flow rate, mixing ratio and combustion efficiency, which achieved 98%. The averaged specific impulses were in agreement, for the two tests at atmospheric pressure, with a value close to 230 s. This value was increased for the test under low pressure conditions and reached 272 s with a nozzle expansion ratio of 50, which is consistent with the extrapolation made from the results of the atmospheric pressure tests. Compared to the first test campaign, the combustion efficiency has been improved by over 15% and a gain of over 40 s was measured in the specific impulse. The second analysis method considered the instantaneous data based on the ultrasonic measurement technique, which enabled the evolution of the fuel regression rate during the firing tests to be deduced. It is then possible to plot the instantaneous evolution of the experimental specific impulse as a function of the oxidizer-to-fuel ratio, which presents an evolution similar to the theoretical one (Figure 16). The dispersion of the experimental data around the averaged value is due to both the combustion instabilities inherent to all types of chemical engines and the uncertainty in the ultrasonic sensor measurements.

Based on the experimental results of the test performed under low pressure conditions, an extrapolation to the satellite conditions can be achieved by assuming the same efficiencies (98% and 91.4% for the combustion and engine efficiencies), the same chamber pressure and the same oxidizer-to-fuel ratio. According to the isentropic laws, the extrapolated performances with a nozzle expansion ratio of 400 should be 302 s in a vacuum for hydrogen peroxide concentrated at 87.5%. Moreover, according to the hydrogen peroxide supplier, it would be possible to consider H_2O_2 at 98% for a commercial engine. In this case and assuming a catalyzer adapted to this hydrogen peroxide concentration, the vacuum specific impulse should be 320 s for a nozzle expansion ratio of 400. It should be noted that the two previous extrapolated values are computed based on a nozzle efficiency of 93%, which can still be improved. Using a standard nozzle efficiency of 98%, the expected specific impulse for a hybrid engine operating with a nozzle expansion ratio of 400 and with hydrogen peroxide concentration of 98% would be 337 s. This value is competitive if compared to the current bi-liquid MMH/N_2O_4 engine.

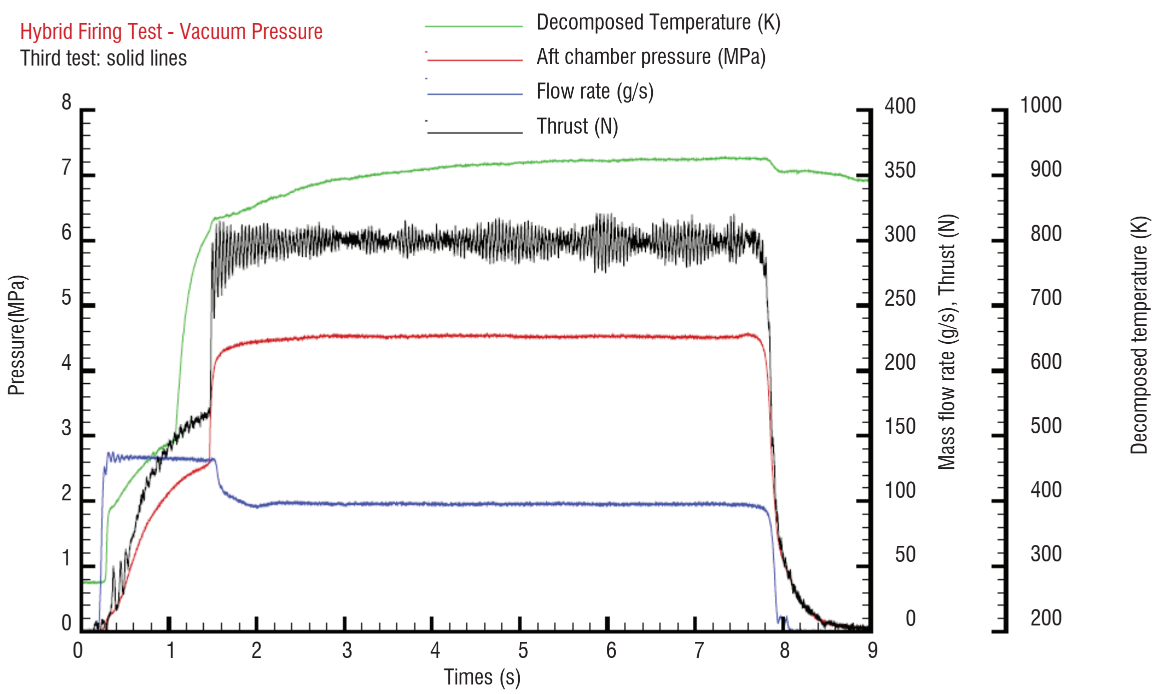


Figure 15 - Experimental results of the firing test performed under low pressure conditions

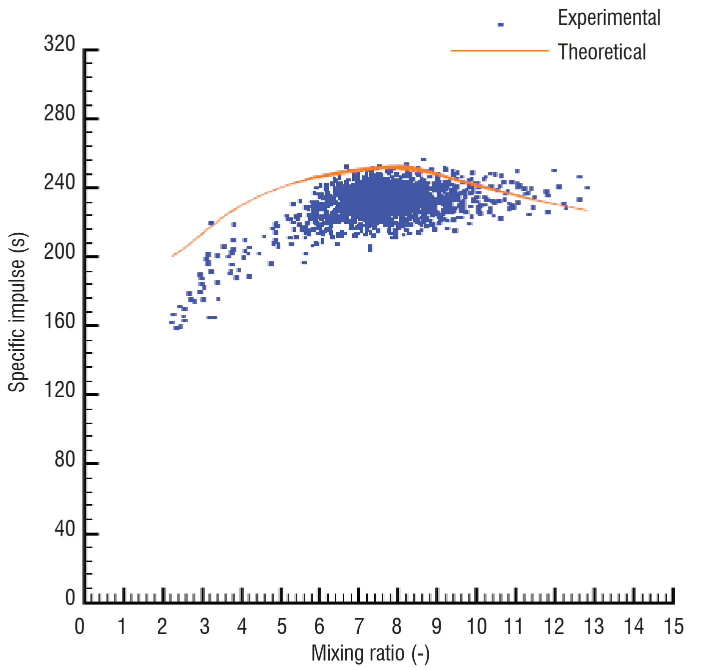


Figure 16 - Evolution of the specific impulse as a function of the mixing ratio for the first atmospheric firing test

The next step consists in developing a hybrid engine compatible with the satellite mission requirements, i.e., a long duration burn with medium and constant thrust. Long duration meaning more than one hour, this represents a technology breakthrough compared to the rocket engine applications. The main impact concerns the combustion chamber configuration and the nozzle erosion. These challenges are being investigated within the framework of the H2020 European project HYPROGEO, coordinated by Airbus DS. The project is aimed at improving the technological maturity of key components of the hybrid chemical propulsion technology, in order to allow independent access to new space transportation missions and to obtain significant cost reductions. The targeted application is a kick-stage module that can be plugged into a full electrical satellite platform in order to reduce the transfer duration from the GTO to the GEO orbits. This module also combines hybrid chemical engines and electrical thrusters to optimize its efficiency in term of propulsive performances, mass, cost, etc. In this context, the objective of WP2, led by ONERA, is to develop a new innovative combustion chamber architecture compatible with the mission needs. Indeed, due to the radial regression of the fuel, the classical configuration of a hybrid engine (Figure 1) is not reconcilable with these specifications, since the propulsive performances evolve over time and the diameter of the fuel grain would not be consistent with the accommodation constraints. At the beginning of the HYPROGEO project, highly innovative combustion chamber designs were created, but only one should remain at the end of the project. Consequently, the first task in the WP2 was to perform a trade-off between these concepts to select the most promising architecture of the combustion chamber, which will be further investigated, manufactured and tested later in the project. The selected architecture is based on the piston concept, in which the fuel grain burns like a cigarette and its regression is balanced with the displacement of the fuel grain thanks to a system to be defined (i.e., pressurized gas at the top of fuel associated with a stop in the combustion chamber, etc.). The gaseous oxidizer, coming from the catalyzer, is injected from the wall to the center of the combustion chamber. Further R&D activities

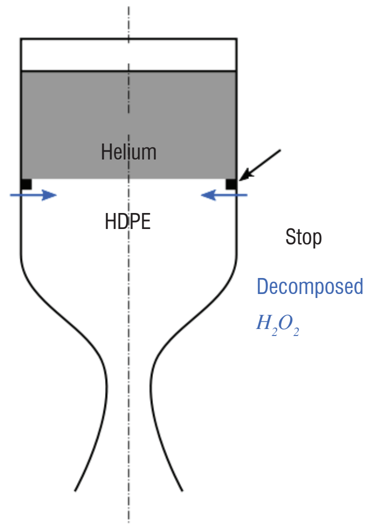


Figure 17 - Principle scheme of the combustion chamber for the satellite apogee engine

are now planned, including CFD analysis to help the design of the hybrid engine demonstrator. The firing tests are planned for the beginning of 2017.

Concluding remarks

New transportation vehicles for space exploration missions or micro-gravity experiments would require simplified, low-cost and thrust-modulated operations combined with a high level of performance, reliability and availability. These capabilities could be achieved through hybrid propulsion, which combines a solid fuel with a liquid oxidizer. However, the hybrid propulsion concept has never been the subject of thorough studies, mainly because heavy investments were made in liquid and solid space propulsion making them today nearly unavoidable for large space launchers and sounding rockets. The demand for a new small launch system could justify maturing this technology today. However, the scaling of hybrid rocket technology would first need to be demonstrated, in order to develop this technology for these kinds of applications.

ONERA has developed a know-how of more than 40 years in the hybrid propulsion concept, based on theoretical, experimental and numerical approaches. The first European hybrid rocket (LEX) was launched by ONERA in 1964, followed by 7 other successful flights in 1965 and 1967. Twenty years ago, ONERA again started research activities on hybrid propulsion based on polymeric fuels and more recently on liquefying fuels, which provide higher regression rates compatible with launcher applications. A large database of static firing tests is available for various operating conditions and fuel compositions. Some successful flight tests also took place within the PERSEUS program. Appropriate numerical models are also available for predicting the fuel regression rate and CFD is now widely used to better understand the physical mechanisms and instabilities inside the combustion chamber. Finally, firing tests were performed to demonstrate the validity of the extrapolation of the propulsive performances from atmospheric pressure to vacuum pressure conditions.

Future activities are based on the development of an innovative combustion chamber architecture compatible with satellite mission requirements, within the framework of the HYPROGEO project. Moreover, this development does not require a scale factor study, since the geometry of such a combustion chamber and its dimensions are comparable with current lab-scale facilities. From 2016 to 2020, ONERA will also support and finance its own project on hybrid propulsion, called HYSAC. It will be aimed at strengthening ONERA skills in terms of numerical modeling of the combustion chamber, oxidizer injection, fuel formulation, definition of thermal protection materials and optimization of propulsive performance of the hybrid propulsion technology, with the goal of designing an innovative engine for a launcher upper-stage

Acknowledgements

The authors would like to thank the French Space Agency (CNES), the French Armament Procurement Directorate (DGA), the H2020 program of the European Union, the Midi-Pyrénées Region, Airbus DS and Thales Alenia Space for funding research studies and equipment that contributed to this research work. The authors would also like to thank Michel Prévost, Yves Maisonneuve, Gérard Lavergne and Nicolas Pelletier.

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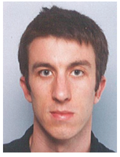
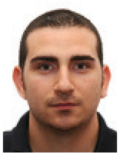


Jean-Yves Lestrade is a research scientist at ONERA. He received his Engineering diploma from ENSIETA in 2009, a Master of Sciences Degree from ISAE-ENSMA in 2009 and a PhD in Energetic and Transfers from ISAE in 2012. His research activities are mainly related to hybrid propulsion. He is also the WP2 leader of the H2020 HYPROGEO project and is the coordinator of the HYSAC project



Jérôme Messineo has been a PhD student at ONERA's Propulsion Laboratory since October 2013. His research field is mainly focused on numerical simulations and instabilities of hybrid rocket engines. He received his Engineering Diploma from ISAE-ENSMA and a Master of Sciences Degree from UPS at ISAE-SUPAERO in 2013.

Jouke Hijlkema is a senior scientist at ONERA. He performs research in the domain of rocket propulsion and manages projects with government agencies and industrial customers. He received a masters degree in aeronautical engineering from Hogeschool Haarlem and a masters degree in mathematics from the university of Nijmegen, both in the Netherlands. He obtained a PhD from ISAE in Toulouse in the field of applied mathematics.

Pierre Prévot, is a senior scientist at ONERA. He has worked for many years in studies and tests in the fields of solid, liquid and hybrid propulsion at the Propulsion Laboratory. He received his engineering diploma from ESSTIN in Nancy in the field of fluid mechanics and energetics.

Grégoire Casalis is the scientific deputy director of DMAE. His scientific topics are related to flow stability from generic configurations (laminar boundary layer) to complex ones (internal solid rocket motor flow).

Jérôme Anthoine is the Head of the Propulsion Laboratory Unit of the ONERA Aerodynamics and Energetics Modeling Department (DMAE). He received a Mechanical Engineering Degree from the Faculté Polytechnique de Mons (Belgium) in 1995, the von Karman Institute Diploma in 1996 and a Ph.D. in Space Propulsion from the Université Libre de Bruxelles in 2000. Then, he joined the Faculty of the Von Karman Institute to develop the aeroacoustics and space propulsion activities. Since 2009, he has been a senior scientist in Space Propulsion at ONERA and is currently managing research activities on pressure oscillations in solid rocket motors and on hybrid chemical propulsion.