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Performances of a Multi-Pulsed Hybrid Rocket Engine Operating with Highly Concentrated Hydrogen Peroxide

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Compared to solid propulsion, hybrid rocket engines allow multi-pulsed operation and thrust modulation at a lower cost than liquid propulsion. However, the hybrid technology suffers from low propulsive performances generally due to low combustion efficiency with regards to the other chemical propulsion systems. When hydrogen peroxide (H_2O_2) is used as oxidizer, static firing tests demonstrated that combustion efficiency can be increased by using a hollow cone nozzle, producing finest atomization of the liquid oxidizer with no central distribution, instead of a solid cone nozzle with coarse droplets. The use of a catalyzer, for decomposing the H_2O_2 upstream of the combustion chamber, is also slightly improving the combustion efficiency. The best performance is obtained when combining a catalyst and a swirl injector in order to inject a swirling oxidizing gaseous stream which improves mixing between the two propellants. The catalytic injection of oxidizer also allows removing the pyrotechnic igniter which provides more safety and is necessary for a multi-pulsed operation. A complementary solution to further increase the propulsive performances consists of using H_2O_2 at a higher concentration than the spatial grade (87.5%). Static firing test showed that specific impulse is improved by 8 s for 98% H_2O_2. Finally, multi-pulsed operation over more than one minute of cumulated firing time with 98% H_2O_2 was demonstrated with constant propulsive performances.

I. Introduction

THE development efforts realized for many years on solid and liquid chemical engines made their use almost exclusive for launcher applications. However, these two concepts have disadvantages (cost, complexity, safety, reliability, environmental impact, etc.) that strain budget, performances and environmental impact. They also hinder the development of new applications such as nano-launchers or orbital insertion of multiple payloads in various orbits. These new space missions require affordable, reliable and tailorable propulsion systems, at a reasonable price and the market exploitation must be sustainable from an environmental viewpoint. High performance (specific impulse), intrinsic safety (separation of the reactants, tolerance to grain cracks), throttleability, flexibility and system integration (easy-fit of the propulsion system to the spacecreaft) are attractive features offered by hybrid rocket engines, which generally combine a liquid oxidizer with a solid fuel.[1,2]

The most common configuration of a hybrid engine (Fig. 1) uses an oxidizer (stored as a liquid, injected through an atomizer and vaporized in the forward dome of the motor) which flows through a long fuel channel. The oxidizer burns with the pyrolysis gases (resulting from the solid fuel regression) through a diffusion flame located in 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 which is similar to the liquid propulsion technology. The

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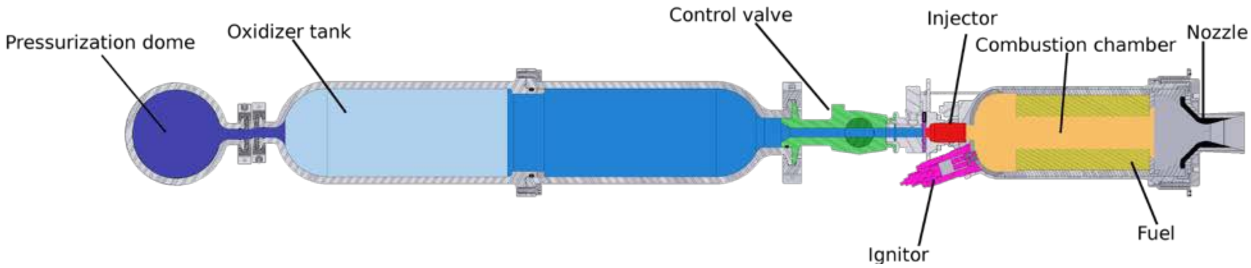

Figure 1. Hybrid chemical engine concept.

convective and radiative heat fluxes stemming from the diffusion flame provide the energy needed for the solid grain pyrolysis in order to sustain heterogeneous combustion.[3] Such a cycle makes hybrid combustion a self-sustained phenomenon which 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 as long as the oxidizer is injected and the solid fuel grain is regressing. The engine extinction can be obtained by closing the oxidizer valve.

Since the flame between oxidizer and 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. So, the fuel regression rate varies with mass flux and along the length of the combustion chamber following the relatively simple expression:[4-6]

$$\dot{r} = aG^n x^m \tag{1}$$

where $\dot{r}$ is the fuel regression rate, G the total propellant mass flux, x the distance down the combustion chamber and a, n, m are the regression rate constants, characteristic of the propellants and of the combustion chamber geometry. The solid fuel regression rate is one of the most important values in the hybrid conceptual design process used to determine the propulsive performances. Since G includes the mass flow from both the injected oxidizer and the pyrolyzed fuel, it increases continuously down the combustion chamber. Thus, for the common configuration (Fig. 1) and as opposed to solid rocket motor, the amount of ablated fuel and the local combustion chamber oxidizer to fuel mixture ratio (O/F) are varying down the motor length.

The hybrid propulsion technology has two main drawbacks compared to the other chemical engines: the oxidizer to fuel mixture ratio (O/F) is varying with the burning time, due to the variation of the fuel grain geometry, and the propulsive performances are low, generally due to low combustion efficiency. Both of these drawbacks appear for the common hybrid engine configuration in which the oxidizer is injected, through an atomizer, directly as liquid in the combustion chamber. Solutions exist to overcome both the varying O/F ratio[7-10] and the low combustion efficiency, solutions for the last drawback being developed later in the paper.

The combustion phenomenon is similar to that of a turbulent diffusion flame for which the flame zone is established within the boundary layer[11] (Fig. 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.

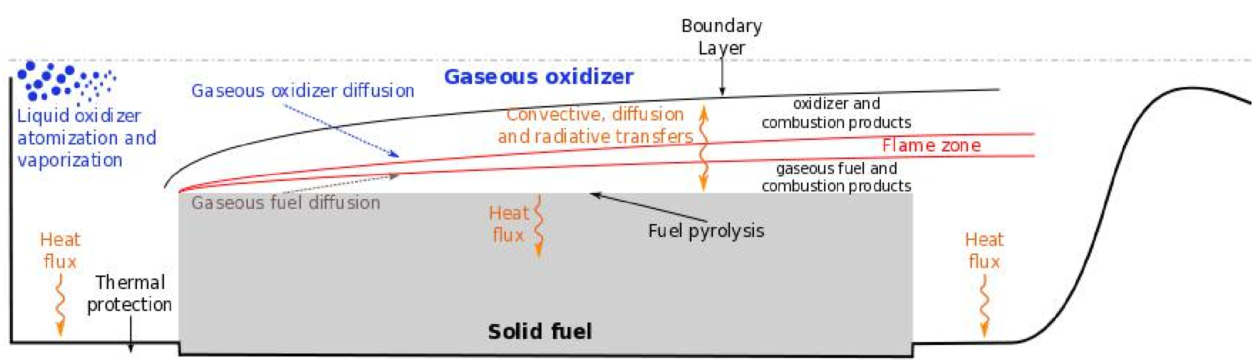

Figure 2. Hybrid propulsion physics.

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

Hybrid propulsion has been developed for decades but never received the development costs spent for solid and liquid propulsion. Therefore, the maturity is not yet achieved and there is still progress to be made before proposing that propulsive solution for a new application. Recent advances from sounding rocket propulsion performed at the French Aerospace Lab (ONERA) and the vision for spacecraft propulsion has been recently published in Ref. 12.

To date, research in developing hybrid propulsion systems strongly depends on lab-scale firing tests. These experiments do not allow to fully understand the complex physics within their combustion chambers. Hybrid chemical propulsion physics indeed 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). The knowledge of the complex interactions between these wide-ranging physical phenomena is however fundamental for hybrid motor design and performance prediction. Considering the severe operating conditions in term of pressure, temperature and oxidizing ambiance, numerical simulations are the only mean to obtain a good knowledge and understanding of the gaseous flow inside of the combustion chamber in order to optimize this kind of chemical engine. 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. Thus, one-dimensional codes are privileged during the preliminary design phases to carry out many calculations in a relatively short time.[6,13] The full Navier-Stokes simulations, providing much more details but being very costly in computation time, are, in turn, generally used for detailed understanding of physical phenomena.[14,15]

II. The HYCOM Lab-Scale Test Facility

To validate the models and the numerical simulations, experiments are still required on instrumented lab-scale engines such as the HYCOM facility (Fig. 3, left). Like most lab-scale hybrid rocket motors, the HYCOM facility is composed of five parts (Fig. 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 the fuel grain, etc.

The engine, which can operate until 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 get the propulsive performances. It also includes temperature and pressure measurements of the liquid oxidizer just upstream of the injector. In order to measure the fuel regression rate, the engine is also instrumented with ultrasonic 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.[16-18] It has the advantages of being non intrusive and easily implemented contrary to visualization and X-rays measurement techniques.

Several firing tests have been performed and a large database is available for different configurations: oxidizer/fuel couple, fuel length and initial port diameter, oxidizer mass flux, nozzle throat diameter, etc. The list of firing tests which are presented in the present paper is synthetized in Table 1.

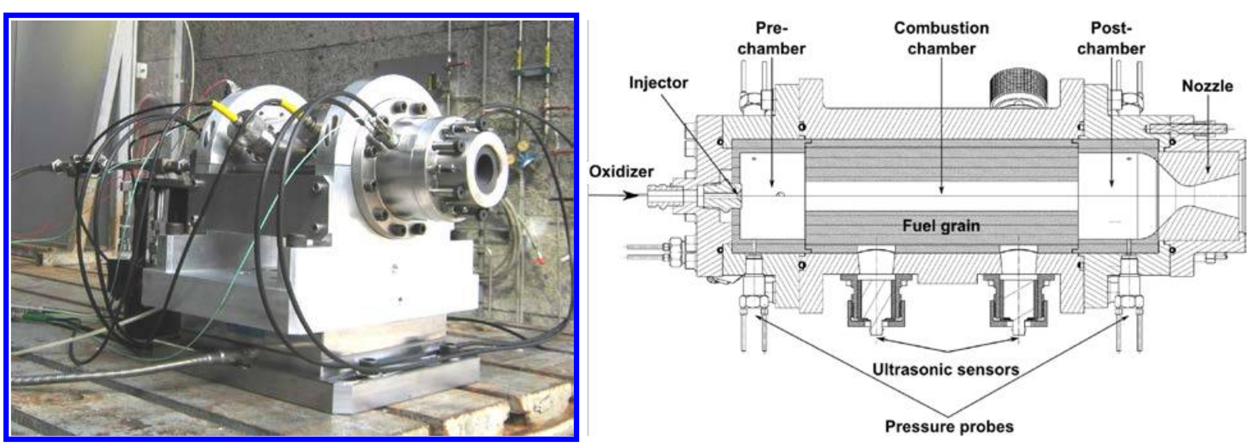

Figure 3. HYCOM lab-scale engine (left) and schematic view of this facility (right).

Table 1. List of hybrid engine firing tests presented in the following sections.

Test #	Duration	H_2O_2 grade	Oxidizer injector	Droplet size / Gaseous injector	Nozzle throat (mm)	Expansion ratio	Internal reference
1	Short	87.5 %	Atomizer	Coarse	7	50	HYTELSAT 04
2	Short	87.5 %	Atomizer	Very fine	7	6.3	HYTELSAT 11
3	Short	87.5 %	Catalyzer	Axial	7	6.3	HYCAT 12
4	Short	87.5 %	Catalyzer	Swirl	7	6.3	HYCAT 03
5	Short	98 %	Catalyzer	Swirl	7	6.3	HYCAT 13
6	Multi-pulsed	98 %	Catalyzer	Swirl	12	3	HYCAT 15

III. Hybrid Rocket Engine Tests with Liquid Oxidizer Injection

For all the tests with liquid oxidizer injection, hydrogen peroxide is combined with a high density polyethylene (HDPE) fuel grain. The fuel grain has a 20 mm diameter single-circular port and a 230 mm length, and the nozzle is conical. The oxidizer is injected directly as a liquid in the pre-chamber of the engine through an atomizer or injector (Fig. 3, right). In this configuration, the ignition requires the use of a pyrotechnic igniter. The first test is performed with a Delavan BN solid cone nozzle, producing uniform distribution of oxidizer droplets in a wide angle solid cone spray pattern. The short tests at atmospheric pressure are performed with a nozzle throat of 7 mm and a 6.3 nozzle expansion ratio in order to pressurize the engine. The nozzle is not necessarily adapted depending on the achieved combustion chamber pressure.

Figure 4 provides the temporal evolutions of the combustion chamber pressure, the oxidizer mass flow rate and the thrust. Even though the operating conditions are almost constant over the burning time, the pressure presents large peaks of several bars of pressure oscillations. This is probably due to the selected solid cone nozzle which provides coarse droplets. In order to reduce these oscillations, the atomizer is replaced by the Delavan WDA small hollow cone nozzle, producing finest atomization of the liquid oxidizer with no central distribution. This choice is supposed to reduce the vaporization and decomposition time of the oxidizer and to better mix the oxidizer with the vaporized fuel since the amount of oxidizer injected on the centerline of the fuel grain is minimized.

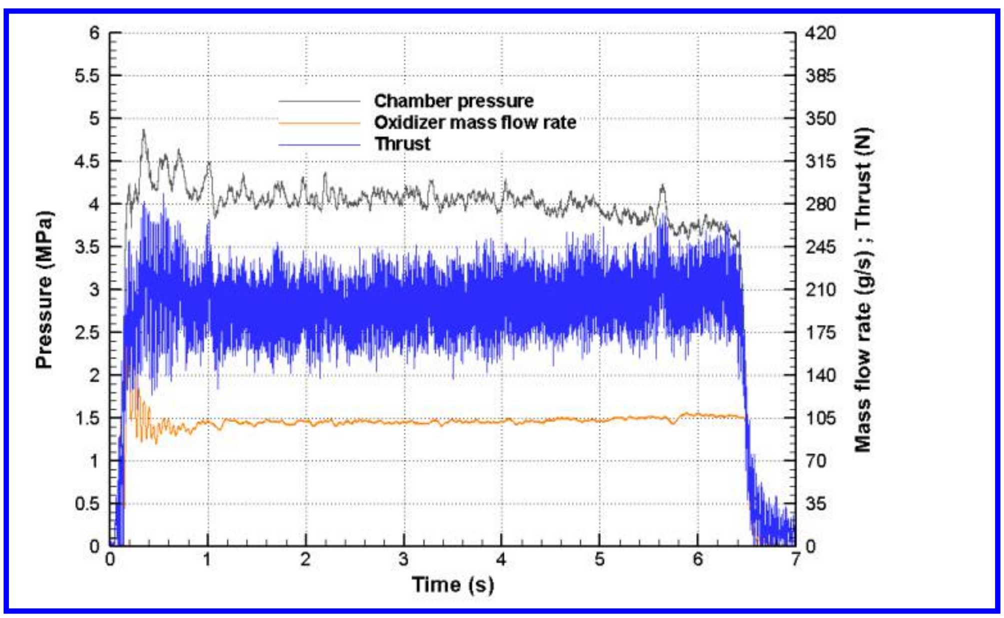

Figure 4. Firing test with liquid oxidizer injected through a coarse droplets atomizer (test 1).

Figure 5 provides the temporal evolutions of the combustion chamber pressure, the oxidizer mass flow rate and the thrust for the hybrid test with a very fine droplets atomizer. As expected, the combustion chamber pressure is much more stable indicating that the instability peaks observed in the test 1 are most probably due to the coarse droplets. Table 2 gives the comparison between the averaged data measured during both firing tests with the two types of atomizers. Since test 1 was performed with a clearly not adapted nozzle (nozzle expansion ratio of 50 for a atmospheric pressure condition), the two experiments cannot be compared in terms of thrust and specific impulse. When using fine droplets, the combustion efficiency, ratio between experimental and theoretical characteristic velocities, is increased by 6% to reach 91% at the optimal oxidizer to fuel mixing ratio, reached for both tests.

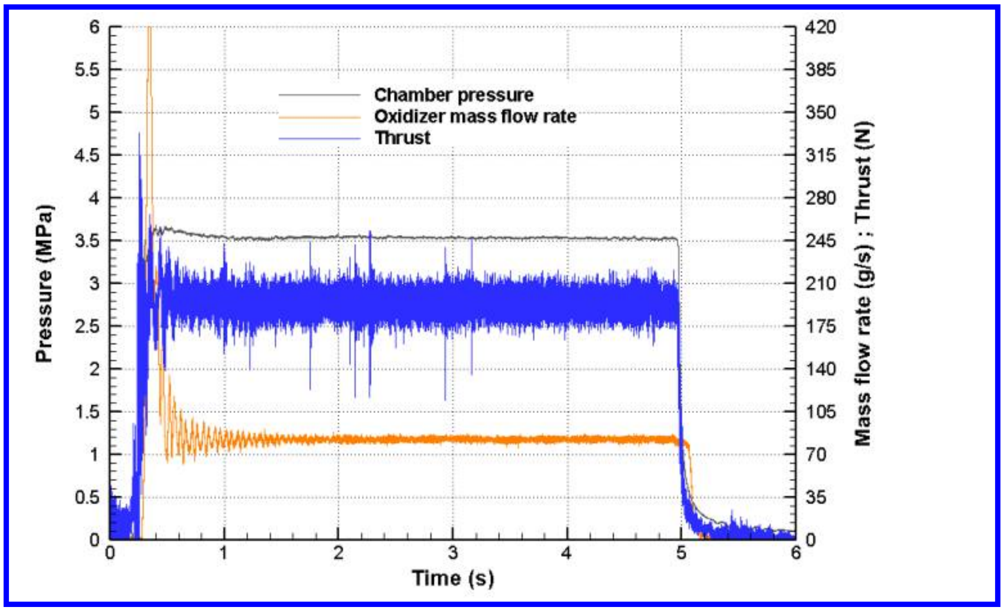

Figure 5. Firing test with liquid oxidizer injected through a very fine droplets atomizer (test 2).

Table 2. Averaged results of the hybrid tests performed with the axial and swirl gaseous injector.

	Coarse droplets (test 1)	Very fine droplets (test 2)
Firing test duration (s)	6.4	4.7
Oxidizer mass flow rate (g/s)	102.5	82.4
Fuel mass flow rate (g/s)*	11.8	11.6
Oxidizer to fuel ratio (-)	8.7	7.1
Chamber pressure (MPa)	4.0	3.54
Characteristic velocity (m/s)	1347	1449
Combustion efficiency (%)	85	91

*The averaged mass fuel rate was calculated based on the mass measurements before and after the firing test.

IV. Hybrid Rocket Engine Tests with Catalytic Injection of Oxidizer

Another way to increase the combustion efficiency is to use a catalytic injector instead of the classical atomizer.[19] The catalyzer decomposes the hydrogen peroxide into a hot gaseous oxidizer which is then injected through a gaseous injector directly within the combustion chamber without using a pre-chamber. The injection of a hot gaseous oxidizer is expected to improve the mixture with the pyrolysis gas coming from the fuel grain and to avoid using a part of the generated heat flux to vaporize and warm the liquid oxidizer. The catalytic injector combines a liquid injector plate, a decomposition chamber containing the catalyst particles and a gaseous injector (Fig. 6, left). The injector plate was designed in order to spread the liquid hydrogen all over the cross section of the decomposition chamber. This chamber consists of an Inconel cylinder closed by refractory steel meshed in order to maintain the catalyst particles inside the decomposition chamber.[20]

When a catalytic bed is coupled to a hybrid engine, the ignition of the engine doesn't require pyrotechnic device anymore, which provides more safety. The ignition is achieved thanks to the energy supplied from the hot oxidizer stream. Consequently, the catalytic bed has to have a very good efficiency associated to a short transient duration. This is also necessary for a multi-pulsed operation of the hybrid engine.

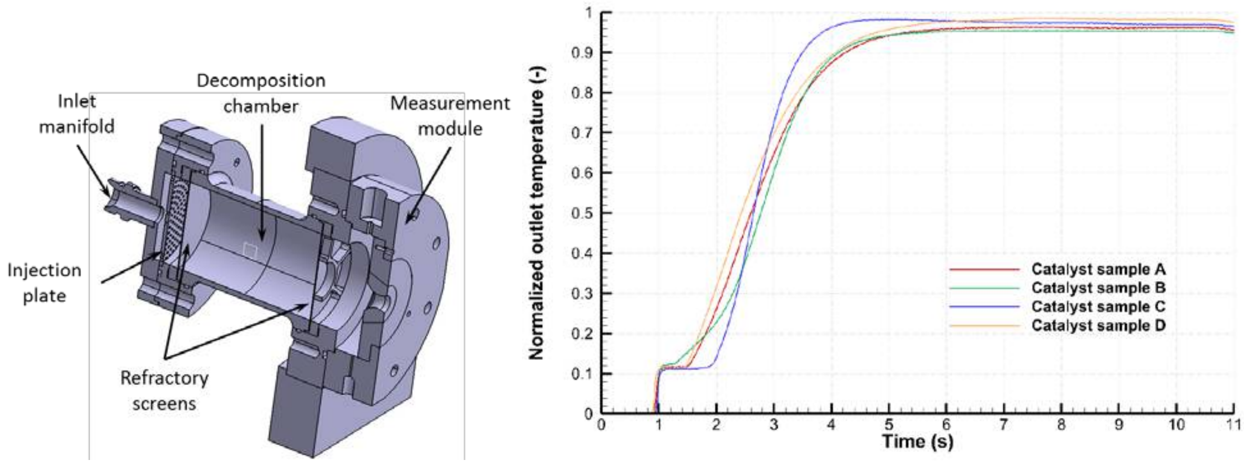

Figure 6. Catalytic injector (left) and comparison between the decomposition temperatures of the four catalyst samples (right).

The measurement module (Fig. 6, left) is only used to characterize the performances of the decomposition chamber and to select the best catalyst particles, though a monopropellant test campaign.[20] The decomposition temperature is measured at the outlet of the catalytic bed thanks to three thermocouples. The measurement of the decomposition chamber pressure enables to obtain the characteristic curve (oxidizer mass flow rate as a function of the pressure differential) of this chamber in order to precisely control the operating conditions of the monopropellant tests and of the hybrid firing tests. However, to determine this curve, tests under pressure are needed which require the use of a nozzle at the outlet of the decomposition chamber. The instrumentation of this facility also includes a Coriolis oxidizer mass flow measurement and temperature and pressure measurements of the liquid oxidizer upstream the manifold.

Four catalysts were selected after a preliminary screening of their catalytic activity with low concentration hydrogen peroxide. Catalysts A, B and C are Pt based catalysts supported on the flight proven Al_2O_3 granules used for the Heraeus hydrazine decomposition catalysts H-KC12GA. They were prepared with different alumina particle sizes, A being supported on the largest one (14-10 mesh) and C on the smallest one (30-25 mesh). Catalyst D is also a Pt based catalyst but supported on an alternative Al_2O_3 material having the same particle size as the one used for catalyst A. Details on the monopropellant test campaign are reported in Ref. 20. Figure 6 (right) provides the comparison between the decomposition temperatures of the catalyst samples. The lower the catalyst particles are, the lower is the transient phase duration and the higher is the efficiency based of the normalized temperature. Catalyst sample D provides better results than catalyst sample A for both the transient phase duration and the efficiency.

For the hybrid firing tests with H_20_2 catalytic decomposition, the geometry of the fuel grain is slightly changed compared to the liquid injection test cases. The fuel grain has a 25 mm diameter single-circular port and a 240 mm length, and the nozzle is again conical. The synoptic diagram of the measurement chain is indicated in Fig. 7.

The gaseous injector is either axial or swirl. Figures 8 and 9 provide the temporal evolutions of the combustion chamber pressure, the oxidizer mass flow rate, the thrust and the oxidizer decomposition temperature for the hybrid tests with the axial and swirl injectors. A monopropellant phase precedes the hybrid mode, the duration of each phase being of the same order of magnitude for both injector. When combustion occurs, the diffusion flame increases the chamber temperature and then the characteristic velocity with the effect to provide a higher combustion chamber pressure. This pressure increase reduces the oxidizer mass flow rate compared to its amplitude during the monopropellant phase. The use of an axial injector is associated to larger pressure oscillations, their amplitude being smaller than with the coarse droplets atomizer but not as small as than with the atomizer providing very fine droplets. These pressure fluctuations are reduced when using a swirl injector which can be explained by an enchanced mixing between the hot gaseous oxidizer and the pyrolyzed fuel. The temperature recorded at the outlet of the decomposition temperature is very similar for both tests and is reaching a value of about 875 K just before the extinction of the engine.

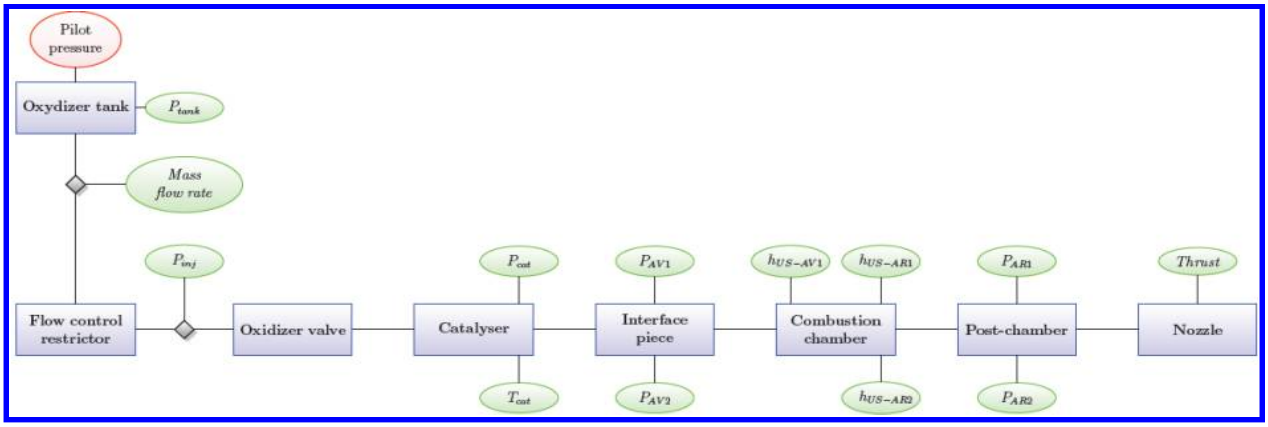

Figure 7. Synoptic diagram of the measurement chain of the HYCOM hybrid engine with catalyzer.

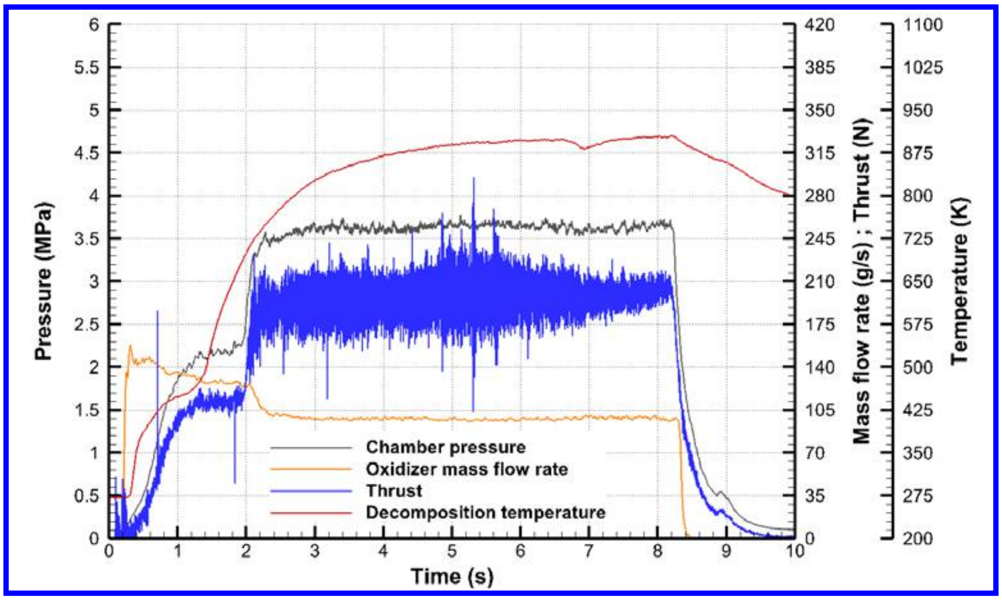

Figure 8. Firing test with catalyzer and the axial gaseous injector (test 3)

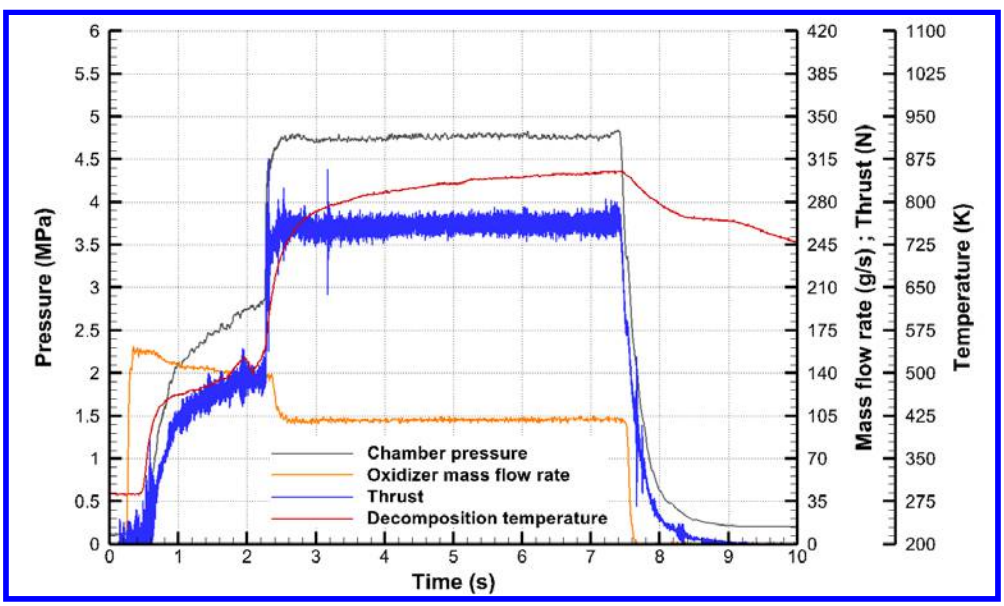

Figure 9. Firing test with catalyzer and the swirl gaseous injector (test 4)

The results are then averaged over the hybrid mode duration in order to deduce the oxidizer to fuel ratio, the propulsive performances and the efficiencies. As presented in Table 3, the averaged experimental oxidizer to fuel ratio is equal to 12.3 for the axial gaseous injector, value away from the optimal one (7.3) for which the specific impulse is maximal. For such mixing ratio, it is easier to have good combustion efficiencies since there is not enough fuel to complete the combustion process. However, the combustion efficiency reaches only 91% while the engine efficiency, ratio between experimental and theoretical specific impulses, reaches 87%. The combustion and engine efficiencies are clearly improved when using the swirl gaseous injector while the oxidizer to fuel ratio is now even closer to the optimal value.

These firing tests enabled to highlight that the use of a decomposition chamber with an axial injector is not enough to increase hybrid engine efficiencies since the performances are comparable to those of the firing test performed with a liquid injection of the oxidizer if an atomizer generating very fine droplets is used. However, when a decomposition chamber is combined to a gaseous swirl injector, the combustion efficiency jumps up to 98%. This value is closer to the one obtained for solid and liquid rocket engine.

Table 3. Averaged results of the hybrid tests performed with the axial and swirl gaseous injectors.

	Atomizer with very fine droplets (test 2)	Catalyzer with axial gaseous injector (test 3)	Catalyzer with swirl gaseous injector (test 4)
Monopropellant phase duration (s)	N/A	1.8	1.6
Hybrid mode duration (s)	4.7	6.3	5.1
Oxidizer mass flow rate (g/s)	82.4	97.8	101.7
Fuel mass flow rate (g/s)*	11.6	8.0	16.1
Oxidizer to fuel ratio (-)	7.1	12.3	6.3
Chamber pressure (MPa)	3.54	3.64	4.72
Thrust (N)	194.0	200.7	260.8
Characteristic velocity (m/s)	1449	1324	1542
Specific impulse (s)	210	193.4	225.7
Combustion efficiency (%)	91	91	98
Nozzle efficiency (%)	97	95	93
Engine efficiency (%)	88	87	91

*The averaged mass fuel rate was calculated based on the mass measurements before and after the firing test.

V. Hybrid Rocket Engine Tests with 98% Hydrogen Peroxide

A complementary solution to improve the hybrid engine performances consists of using more energetic propellants. Several studies were conducted to evaluate the benefit of adding metallic particles in the fuel grain.[21-23] Regarding the oxidizer, when firing tests are performed with hydrogen peroxide, the concentration is generally 87.5% which is the current spatial grade. Nevertheless, according to theoretical computation performed with thermochemical equilibrium code such as RPA (Rocket Propulsion Analysis) or CEA (Chemical Equilibrium with Applications), specific impulse is improved by 12 s when hydrogen peroxide at higher concentration than spatial grade (98%) reacts with the fuel grain. In order to combine very high concentrated hydrogen peroxide with a catalytic bed, the catalyst has to be able to withstand the decomposition temperature of 98% H_2O_2, which is not the case of catalysers made of silver. The catalyst used for the previous tests and based on Pt supported on Al_2O_3 material is compatible with the expected highest decomposition temperature.

The hybrid firing test #5 is identical to test #4, except for the grade of H_2O_2. Figure 10 provides the temporal evolutions of the combustion chamber pressure, the oxidizer mass flow rate, the thrust and the oxidizer decomposition temperature. The decomposition temperature at the end of the firing test is increased to 1060 K, which corresponds to a difference of 180 K compared to the firing test with 87.5% H_2O_2. This temperature difference is slightly lower than the difference between the theoretical decomposition temperatures of both grades, which is equal to 257 K. The combustion chamber pressure is not constant and starts to decrease after 2.5 s of hybrid mode operation. This is explained by a nozzle erosion which occurs due to the much higher flame temperature.

Due to the nozzle erosion, the data are averaged over the two first seconds of the hybrid mode and are reported in Table 4. Thanks to the increase of the decomposition temperature, the ignition of the engine is faster and the monopropellant phase is now reduced to about 1 s. The specific impulse, deduced from the measured thrust and mass flow rates, is increased by 8 s which is lower than the theoretical increase of 12 s calculated using the thermochemical equilibrium code. The smaller benefit is linked to a similar characteristic velocity between the two experiments. In fact, the combustion efficiency is reduced which could be explained by a performance reduction of the catalyst material. It should be noted that nine firing tests involving the same catalyst material were performed between tests 4 and 5, as reported in the internal reference column of Table 1. This can explain the reduction of the combustion efficiency, as well as the smaller increase of decomposition temperature compared to the theoretical expectation. It could be interesting to repeat both tests with a new bench of catalyst material. Nevertheless, the specific impulse demonstrated experimentally is equal to 234 s. On top of an expected increase from using new catalyst material, this value could still be improved by correctly adapting the nozzle expansion ratio, which was not the case for the present experiments as shown by the low nozzle efficiency.

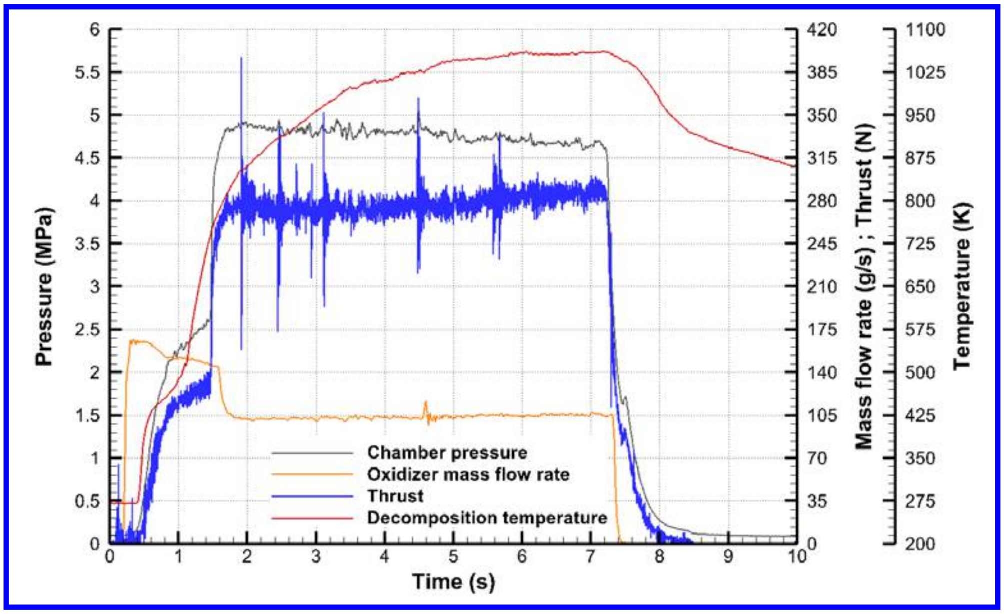

Figure 10. Firing test with catalyzer, swirl gaseous injector and H_2O_2 concentrated at 98% (test 5)

Table 4. Averaged results of the hybrid tests performed for two different concentrations of H_2O_2.

	87.5% H_2O_2 (test 4)	98% H_2O_2 (test 5)
Monopropellant phase duration (s)	1.6	1.1
Hybrid mode duration (s)	5.1	5.9
Oxidizer mass flow rate (g/s)	101.7	102.4
Fuel mass flow rate (g/s)*	16.1	16.6
Oxidizer to fuel ratio (-)	6.3	6.2
Chamber pressure (MPa)	4.72	4.82
Thrust (N)	260.8	273
Characteristic velocity (m/s)	1542	1558
Specific impulse (s)	225.7	233.8
Combustion efficiency (%)	98	93.6
Nozzle efficiency (%)	93	94.4
Engine efficiency (%)	91	88.4

*The averaged mass fuel rate was calculated based on the mass measurements before and after the firing test.

VI. Multi-Pulsed Hybrid Rocket Engine Test

Finally, the last static firing test was performed to demsontrate the capability of several extinctions and re-ignitions of the hybrid engine when using a catalyser and the highest grade of H_2O_2. The test is similar to test #5 in terms of fuel grain geometry, catalyser and is also using a swirl gaseous injector. Since the nozzle throat is made of graphite, the nozzle of test 5 (throat equal to 7 mm) cannot sustain long duration firings. The throat is then changed to 12 mm and the expansion ratio to 3 since the targeted combustion chamber is consequently reduced to about 1 MPa. Again, it should be noted that the nozzle is not necessarily adapted depending on the achieved combustion chamber pressure.

Figure 11 provides the temporal evolutions of the combustion chamber pressure, oxdizer mass flow rate, thrust and decomposition temperature at the outlet of the catalyser for a firing test performed with a cumulative firing duration of more than 30 seconds. Starting from the second ignition, the monopropellant phase disappears and the ignition delay of the engine is then reduced significantly. This is explained by the very high temperature of the catalyser reached at the end of the previous burning pulse involving a sudden increase of this temperature when the engine is re-ignited. The curves of the different pulses are very well superposed, as indicated by Fig. 12.

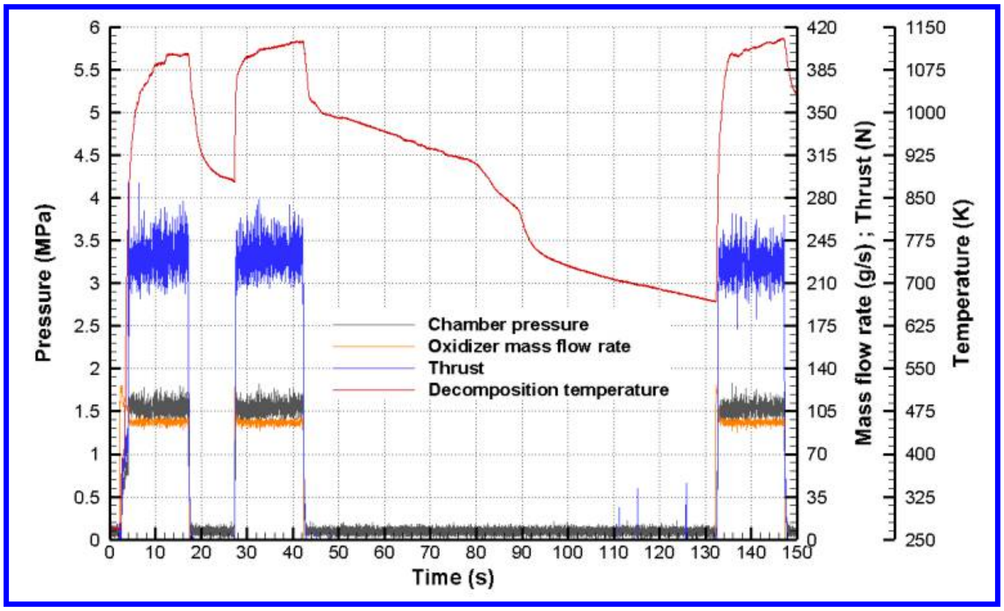

Figure 11. Firing test with multi-pulsed operation and H_2O_2 concentrated at 98% (test 6)

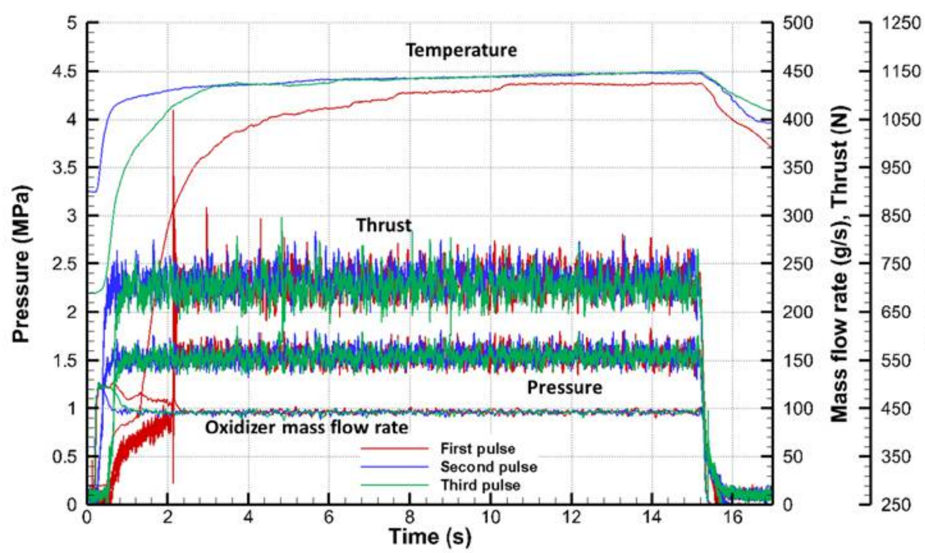

Figure 12. Superposition of the data from the different pulses for the multi-pulsed firing test (test 6)

The data can then be averaged over 10 s over each pulse and the performances reached from each pulse are compared in Table 5. The pulses are very well reproducible. Finally, the combustion efficiency of the short and long tests with 98% H_2O_2 can be compared in Table 6 since the two experiments are similar except the nozzle dimensions. For the multi-pulsed firing test, the data are averaged over all the three pulses. The data of both tests are very well matching showing a very good reproducibility between firing experiments. Since the nozzles are different and not necessarily adapted for both experiments, the thrust and specific impulse cannot be compared.

Table 5. Averaged results from the different pulses for the multi-pulsed firing test (test 6).

	Pulse 1	Pulse 2	Pulse 3
Oxidizer mass flow rate (g/s)	96.2	95.8	95.8
Chamber pressure (MPa)	1.52	1.55	1.54
Thrust (N)	231	234	226

Table 6. Averaged results of the short and long duration tests performed with highly concentrated H_2O_2.

	Short duration (test 5)	Long duration (test 6)
Monopropellant phase duration (s)	1.1	N/A
Hybrid mode duration (s)	5.9	35
Oxidizer mass flow rate (g/s)	102.4	95.9
Fuel mass flow rate (g/s)*	16.6	15,9
Oxidizer to fuel ratio (-)	6.2	6.0
Chamber pressure (MPa)	4.82	1.54
Characteristic velocity (m/s)	1558	1557
Combustion efficiency (%)	93.6	93.6

VII. Conclusions

Hybrid chemical rocket engine suffers from low propulsive performances generally due to low combustion efficiency with regards to the other chemical propulsion systems. Static firing tests of an hybrid engine were then performed to demonstrate the possibility of improving these performances by modifying the injection of the liquid oxidizer. When hydrogen peroxide (H_2O_2) is used as oxidizer, static firing tests demonstrated that combustion efficiency can be increased from 85 to 91% by using a hollow cone nozzle, producing finest atomization of the liquid oxidizer with no central distribution, instead of a solid cone nozzle with coarse droplets. When combining a catalyzer, for decomposing the H_2O_2 upstream of the combustion chamber, with a swirl gaseous injector in order to inject a swirling oxidizing gaseous stream which improves mixing between the two propellants, the combustion efficieny reaches 98%. The catalytic injection of oxidizer also allows removing the pyrotechnic igniter which provides more safety and is necessary for a multi-pulsed operation. A complementary solution to further increase the propulsive performances consists of using H_2O_2 at a higher concentration than the spatial grade (87.5%). Static firing test showed that specific impulse is improved by 8 s for 98% H_2O_2 and this value could be probably larger since experiments were performed with used catalyst materials which had already reduced performances. Finally, multi-pulsed operation over more than one minute of cumulated firing time with 98% H_2O_2 was demonstrated with constant propulsive performances.

References

[1]Anthoine, J., Maisonneuve, Y. and Prevost, M., "The hybrid propulsion to serve space exploration and micro-gravity experiments," *61st International Astronautical Congress*, Prague, Czech Republic, 2010, IAC-10-C4.6.9.

[2]Lengelle, G. and Maisonneuve, Y., "Hybrid propulsion: past, present and future perspectives," *6th International Symposium on Propulsion for Space Transportation of the 21st Century*, Versailles, France, 2002.

[3]Marxman, G. and Gilbert, M., "Turbulent Boundary Layer Combustion in the Hybrid Rocket," *9th International Symposium on Combustion*, Academic Press, 1963, pp. 371-372.

[4]Altman, D. and Holzman, A., "Overview and history of hybrid rocket propulsion," Fundamentals of Hybrid Rocket Combustion and Propulsion, edited by M. J. Chiaverini and K. K. Kuo, *Progress in Astronautics and Aeronautics*, Vol. 218, Academic Press, New York, 2007, pp. 1-36.

[5]Altman, D. and Humble, R., "Hybrid rocket propulsion systems," *Space propulsion analysis and design*, edited by R. Humble, G. Henry and W. Larson, McGraw-Hill, New York, 1995, pp. 365-441.

[6]Pelletier, N. and Maisonneuve, Y. "A numerical code for hybrid space propulsion design and test," *3rd International Conference on Green Propellants for Space Propulsion*, Poitiers, France, 2006.

[7]Rice, E.E., Gramer, G.J., St Clair, G.P. and Chiaverini, M.J., "Mars ISRU CO/O2 rocket engine development and testing," *7th International Workshop on Microgravity Combustion and Chemically Reacting System*, 2001.

[8]Chiaverini, M.J., "Review of solid-fuel regression behaviour in classical and non-classical hybrid rocket motors," Fundamentals of Hybrid Rocket Combustion and Propulsion, edited by M. J. Chiaverini and K. K. Kuo, *Progress in Astronautics and Aeronautics*, Vol. 218, Academic Press, New York, 2007, pp. 37-125.

[9]Lestrade, J-Y., Messine, J., Anthoine, J., Musker, A. and Barato, F., "Development and Test of an Innovative Hybrid Rocket Combustion Chamber," *7th European Conference for Aerospace Sciences (EUCASS)*, Milano, Italy, 2017.

[10]Odic, K., Denis, G., Ducerf, F., Martin, F., Lestrade, J-Y., Verberne, O., Ryan, J., Christ, P. and de Crombrugghe, G., "HYPROGEO hybrid propulsion: project objective and coordination," *6th International Symposium on Propulsion for Space Transportation (Space Propulsion)*, Roma, Italy, May 2016.

[11]Muzzy, R.J., "Schlieren and shadowgraph studies of hybrid boundary layer combustion," *AIAA Journal*, Vol. 1, 1963, p. 2159.

[12]Lestrade, J.Y., Messineo, J., Hijlkema, J., Prévot, P., Casalis, G. and Anthoine, J., "Hybrid Chemical Engines: Recent Advances from Sounding Rocket Propulsion and Vision for Spacecraft Propulsion," *AerospaceLab Journal*, Issue 11, Paper 14, June 2016.

[13]Lestrade, J.Y., Anthoine, J. and Lavergne, G., "Liquefying fuel regression rate modeling in hybrid propulsion," Aerospace Science and Technology, Vol.42, 2015, pp. 80-87.

[14]Messineo, J., Lestrade, J-Y. and Anthoine, J., "Numerical simulation of a H_2O_2/PE hybrid rocket motor," *6th European Conference for Aerospace Sciences (EUCASS)*. Krakow, Poland, 2015.

[15]Messineo, J., Lestrade, J.Y., Hijlkema, J. and Anthoine, J., "Vortex Shedding Influence on Hybrid Rocket Pressure Oscillations and Combustion Efficiency," *Journal of Propulsion and Power*, Vol. 32, N. 6, 2016, pp. 1386-1394.

[16]Carmicino, C. and Russo-Sorge, A., "Performance comparison between two different injector configurations in a hybrid rocket," *Aerospace Science and Technology*, Vol.11, 2007, pp. 61-67.

[17]Cauty, F., "Non-intrusive measurement techniques applied o the hybrid motor solid fuel degradation," *2rd International Conference on Green Propellants for Space Application*, Chia Laguna, Italy, 2004.

[18]Russo Sorge, A., Esposito, A., Quaranta, G. and Torella, G., "Regression rate measurements in a hybrid rocket," *36th AIAA/ASME/SAE/ASEE Joint Propulsion Conference*, 2000, AIAA Paper 2000-3438.

[19]Lestrade, J.Y., Anthoine, J., Verberne, O., Boiron, A.J., Khimeche, G. and Figus, C., "Experimental Demonstration of the Vacuum Specific Impulse of a Hybrid Rocket Engine," *Journal of Spacecraft and Rockets*, Vol. 54, No. 1, 2017, pp. 101-108.

[20]Lestrade, J.Y., Prévot, P., Messineo, J., Anthoine, J., Casu, S. and Geiger, B., "Development of a Catalyst for High Concentrated Hydrogen peroxide," *6th International Symposium on Propulsion for Space Transportation (Space Propulsion)*, Roma, Italy, May 2016.

[21]Calabro, M., De Luca, L.T., Galfetti, L., Raina, H. and Perut, C., "Advanced hybrid solid fuels," *58th International Astronautical Congress*, Hyderabad, India, 2007, IAC-07-C4.2.09.

[22]Evans, B., Favorito, N., Risha, G., Boyer, E., Wehrman, R. and Kuo, K.K., "Characterization of Nano-Sized Energetic Particle Enhancement of Solid-Fuel Burning Rates in an X-Ray Transparent Hybrid Rocket Engine," *40th AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit*, Fort Lauderdale, Florida, 2004, AIAA Paper 2004-3821.

[23]Maggi, F., Gariani, G., Galfetti L. and DeLuca, L.T., "Theoretical analysis of hydrides in solid and hybrid rocket propulsion," *International Journal of Hydrogen Energy*, Vol. 37, No. 2, 2012, pp. 1760-1769.