DEVELOPMENT OF A TEST BENCH FOR HYBRID THRUSTERS

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Abstract. *The interest on the development of hybrid thrusters has grown in the last years motivated by their low cost, good reliability, low environmental impact and safety characteristics. This work presents the design and construction of a bench for testing hybrid thrusters up to 500 N. The main systems are shown, including propellant storage, pressurization, ducting, flow control, data acquisition, and diagnostics. A 400 N hybrid thruster using N_2O and paraffin as propellants was designed and tested in the bench. Preliminary test results are presented.*

Keywords*: test bench, paraffin, nitrous oxide, hybrid thruster*

1. INTRODUCTION

Chemical propulsion systems can be classified according to the physical states of the propellants used: solid, liquid, gas or hybrid. Hybrid propulsion systems use propellants in different states, most frequently use a solid fuel and a liquid oxidizer (Sutton, 2001).

The hybrid thruster technology is known for more than 60 years, however only in the 1960s its safety characteristics motivated a significant research. Nowadays the quest for low toxicity propellants and with low environmental impact ("green propellants") besides the needs of safe operation, storability, low cost missions and interest for launching small payloads or small satellites made the hybrid thrusters more attractive (Humble, 1995).

The safe operation of hybrid propulsion systems is related to the separation between fuel and oxidizer, differently of solid propellant systems in which there is mixing of fuel and oxidizer in the grain. Another important safety characteristic is the independence of the regression rate with respect to pressure, making the hybrid systems safer than solid systems specially when occur pressure peaks.

Hybrid thrusters use only a liquid propellant, requiring a single feed line and a relatively simple injection system, whereas liquid bipropellant thrusters require two feed lines and complex injection plate in order to make the liquid jets of propellants to collide and mix. The control of the flow rate of oxidant in the hybrid engines allows throttling and several restarts (Humble, 1995).

The interest for hybrid thrusters has grown in the last years in Brazil. Santos et al. (2004, 2005) and Contaifer et al. (2005) developed and launched small hybrid rockets using paraffin and nitrous oxide. Gouvêa (2007) tested hybrid thrusters burning hydrogen peroxide and paraffin. Costa et al. (2008) studied the performance of hybrid rockets using paraffin and hydrogen peroxide for launching nanosats into low Earth orbit.

The objective of this work is to describe the implementation and design of a new test bench to determine the performance parameters of low thrust hybrid motors, up to 500 N, for use in apogee rockets, rolling control or orbit transfer of satellites. It is also presented the design of a 400 N hybrid thruster using nitrous oxide or hydrogen peroxide as oxidizer and paraffin as fuel, for testing in the bench.

2. TEST BENCH SYSTEMS AND THRUSTER

A test bench is used to determine performance parameters of hybrid thrusters, such as thrust, specific impulse, characteristic velocity, thrust coefficient and burn time. To determine these parameters, a test bench allows the measurement of thrust, pressures, temperatures, elapsed times and flow rates. The performances of different propellants in different conditions can be evaluated and compared.

The present hybrid thruster test bench comprises a support, oxidizer tanks, pressurizer tanks, thrust balance, feed system, sensors, valves and data acquisition system.

The thruster has systems for injection, ignition, propellant grain burning (combustion chamber), nozzle and cooling system. The fuel is solid paraffin ($\sim C_{20}H_{42}$) with an approximate density of 900 kg/m^3 and the oxidizer is pressurized nitrous oxide or high concentration hydrogen peroxide.

The bench and thruster systems were designed by an interactive process. The thruster main elements were calculated by codes written in Matlab language, following the procedure shown in the next section. Some initial parameters are presented on Table 1. The next section shows the sizing of the paraffin grain and the design of the thruster nozzle.

Table 1. Some initial parameters for the design of a low thrust hybrid rocket.

Maximum theoretical thrust (N)	400
Maximum burn time (s)	40
Chamber pressure (bar)	40
Pressure in the nitrous oxide tank (bar)	80
Pressure in the peroxide tank (bar)	60
Chamber internal diameter (mm)	127
Expansion ratio	4

2.1. Grain Sizing and Thruster Nozzle Design

The total mass flow rate of propellants, $\dot{m}_p$, and the mass flow rate of oxidizer, $\dot{m}_o$, are given, respectively, by

$$\dot{m}_p = F/\left(g_o Isp\right) \tag{1}$$

$$\dot{m}_o = \dot{m}_p/\left(1+1/OC\right) \tag{2}$$

where F is the thrust, g_o is the standard gravity acceleration, Isp is the specific impulse and OC is the mass ratio of oxidizer and fuel.

The initial internal diameter of the grain, D_{ig}, is given by

$$D_{ig} = \left(4\dot{m}_p/\pi G_{io}\right)^{1/2} \tag{3}$$

where G_{io} is the initial mass flux of oxidizer, assumed equal to 200 kg/m^{2}s for both oxidizers.

The external diameter of the paraffin grain, D_{eg}, is calculated by

$$D_{eg} = \left[\left(a/1000\right)\left(4n+2\right)\left(4\dot{m}_p/\pi\right)^n t_b + D_{ig}^{2n+1}\right]^{1/(2n+1)} \tag{4}$$

where t_b is the burn time and a and n are fuel regression constants of the fuel burning with a given oxidizer.

The mass, m_c, and volume, V_c, of fuel are given, respectively, by

$$m_c = \dot{m}_p t_b/OC \tag{5}$$

$$V_c = m_c/\rho_c \tag{6}$$

The grain length, L_g, is given by

$$L_g = 4V_c/\left[\pi\left(D_{eg}^2 - D_{ig}^2\right)\right] \tag{7}$$

The throat area, A_t, and exit area, A_s, of the nozzle are given, respectively, by

$$A_t = F/C_F P_c \tag{8}$$

$$A_s = \varepsilon A_t \tag{9}$$

where C_F is the thrust coefficient, P_c is the chamber pressure and ε is the nozzle expansion ratio.

The specific impulse and the thrust coefficient were calculated with the help of the NASA CEA 2004 code. It was determined the ideal OF ratio for each propellant combination at different chamber pressures. The ideal OF ratio is the ratio that allows to obtain the higher specific impulses. Figures 1 and 2 show plots of the results.

Table 2 shows the propellant regression constants, the specific impulses and the OF ratios used, considering P_c = 40 bar and a nozzle expansion ratio equal to 4. The nozzle flow was considered frozen, i.e., it maintains the same composition along the nozzle.

Table 3 shows the dimensions of the grain and nozzle calculated for each propellant combination.

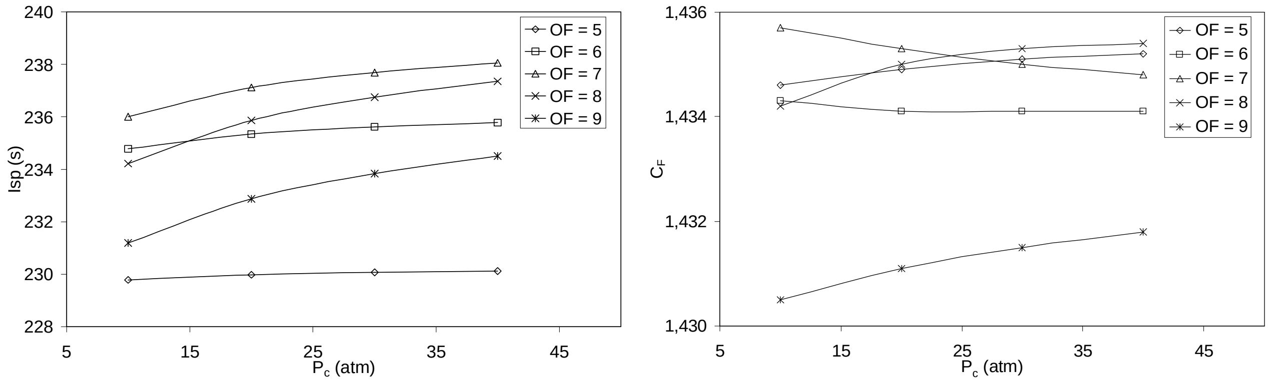


Figure 1. Specific impulse and thrust coefficient for different chamber pressures using N_2O as oxidizer.

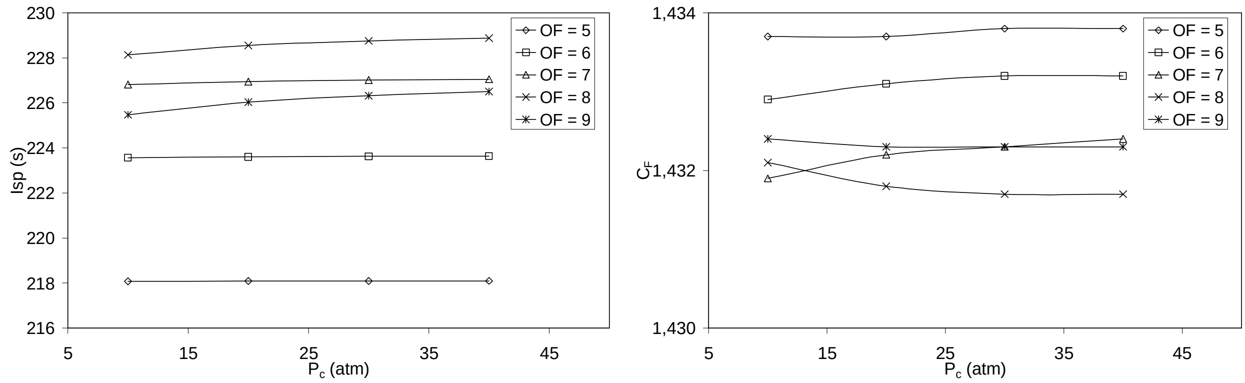


Figure 2. Specific impulse and thrust coefficient for different chamber pressures using H_2O_2 (85%) as oxidizer.

Table 2. Parameters of the propellants and thruster.

Parameter	H_2O_2 (85%)	N_2O
a (mm/s)(m2s/kg)^n	0.0344	0.1704
n (-)	0.9593	0.5
OF (-)	8.5	7.0
Isp (s)	228	236
$\dot{m}_p$ (kg/s)	0.1785	0.1728
$\dot{m}_o$ (kg/s)	0.1597	0.1512

Table 3. Sizes of paraffin grain and nozzle for each oxidizer.

Variable	H_2O_2 (85%)	N_2O
D_{ig} (mm)	33.7	33.1
D_{eg} (mm)	118.7	117.8
L_g (mm)	91.7	109.2
D_t (mm)	9.4	9.4
D_s (mm)	18.8	18.8

2.2. Thruster Design

Figure 3 shows the hybrid thruster for testing, depicting the injection system, ignition system, combustion chamber and nozzle. The combustion chamber is divided in three sections: the pre-chamber, the paraffin grain section and the post-chamber. The thruster main components are made of stainless steel 316 or 304 which have good mechanical strength and compatibility with the combustion products and the oxidizers (nitrous oxide and hydrogen peroxide). Two nozzles were designed: one without cooling, for short tests up to 10 s, and the other one with a jacket for water cooling, thus allowing long duration tests, up to about 40 s. In this case the water flow rate is controlled and measured by a rotameter, and the water temperature are measured by thermocouples at the inlet and exit of the cooling jacket. Pressure transducers and thermocouples measure, respectively, pressures and oxidizer temperatures before and after injection. The oxidizer flow rate is measured and controlled by a Coriolis flow meter and controller.

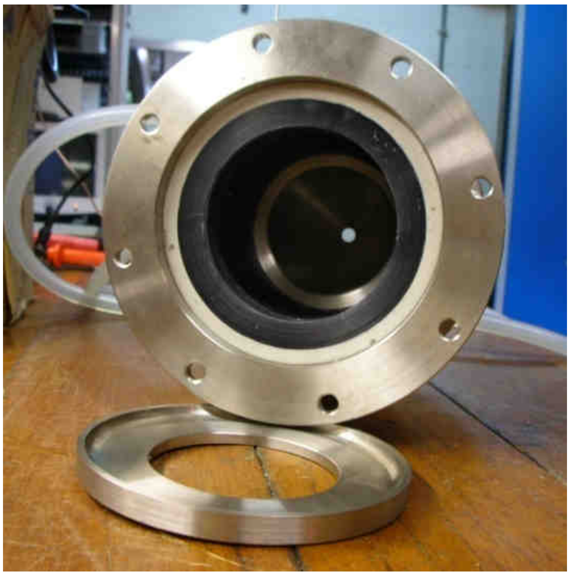


Figure 3a. Hybrid thruster chamber with grain.

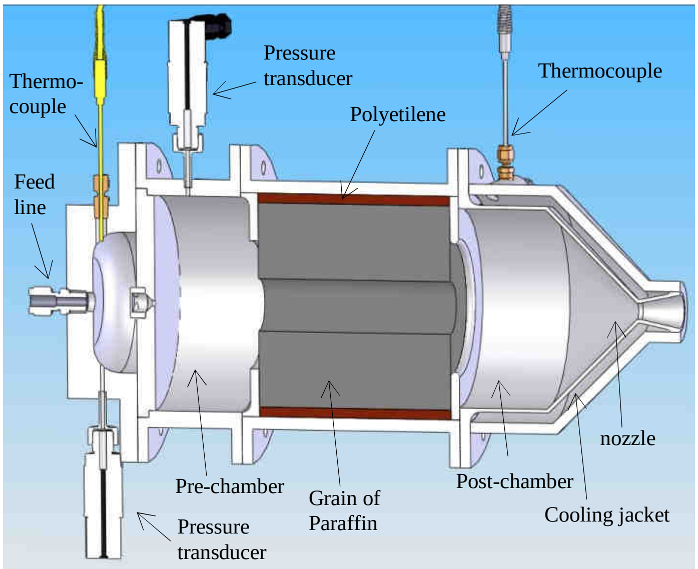


Figure 3b. Cut view of the 400 N hybrid thruster.

A simplex pressure swirl injector was adopted to make possible the atomization in the short pre-chamber. Figure 4 shows a scheme of a pressure swirl injector which was designed following the procedure presented by Lefebvre (1989). The injector was designed for use of nitrous oxide at 80 bar and 293 K, and the generalized diagram of viscosity was used to calculate, approximately, the nitrous oxide viscosity. The main equations for the calculation of the injector geometry are shown next, with the design parameters shown on Table 4.

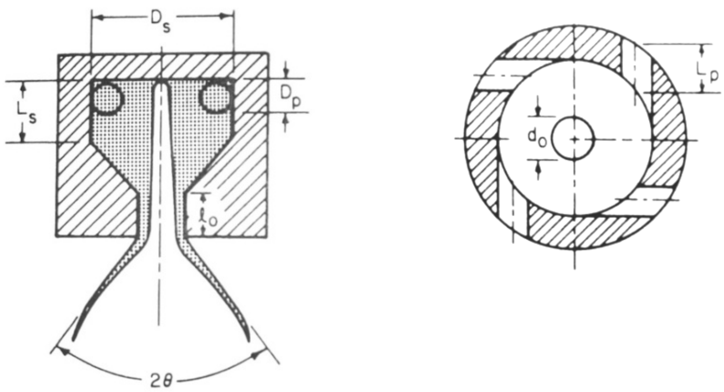


Figure 4. Scheme of a simplex pressure swirl injector. Adapted from Lefebre (1989).

Table 4. Parameter and data used for the injector design.

Parameter	Symbol	Value	Recommended (Lefebvre, 1989)
Exit diameter	d_o	0.0023 m	
N_2O density at 80 bar and 293 K	ρ_L	832 kg/m^3	
N_2O viscosity at 80 bar and 293 K	μ_L	0.000081 kg/m/s	
Injector pressure difference	ΔP_L	40 bar	
c_1	l_o/d_o	0.87	between 0.1 and 0.9
c_2	L_s/D_s	0.7	between 0.31 and 1.26
c_3	$A_p/(D_s d_o)$	1.11	between 0.19 and 1.21
c_4	D_s/d_o	3.35	between 1.41 and 8.13

The mass flow rate of oxidizer through the injector, $\dot{m}_i$, is given by

$$\dot{m}_i = C_d A_o (2\rho_L \Delta P_L)^{0.5} \tag{10}$$

where C_d is the discharge coefficient, A_o is exit area, ρ_L is the density of the liquid oxidizer and ΔP_L is the pressure difference in the injector. The design aims to obtain an injector geometry that yields a discharge coefficient satisfying Eq. (10), since, usually, $\dot{m}_i$, A_o, ρ_L and ΔP_L are known or specified. The discharge coefficient is given by an empirical equation obtained by Jones (Lefebvre, 1989) for a simplex pressure swirl injector:

$$\begin{aligned} C_d &= 0.45\,\mathrm{Re}^{-0.02} c_1^{-0.03} c_2^{0.05} c_3^{0.52} c_4^{0.23} \\ \mathrm{Re} &= d_o \rho_L U / \mu_L \,;\; c_1 = l_o / d_o \,;\; c_2 = L_s / D_s \,;\; c_3 = A_p / D_s d_o \,;\; c_4 = D_s / d_o \end{aligned} \tag{11}$$

Eq. (11) shows that small variations on Re do not affect significantly C_d, whereas variations in c_3 and c_4 can affect strongly C_d. According to Lefebvre, 1989, the total ideal flow velocity at the injector exit, disregarding friction losses, is given by

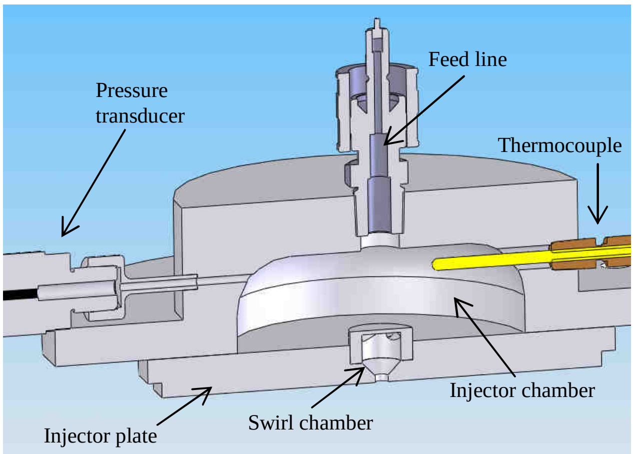
$$U_{ideal} = \left(2\Delta P_L / \rho_L\right)^{0.5} \tag{12}$$

The velocity coefficient, K_v, is the ratio between the actual velocity, U, and the ideal velocity, U_{ideal}. Consequently, the actual velocity at the injector exit is

$$U = K_V \left(2\Delta P_L / \rho_L\right)^{0.5} \tag{13}$$

$$K_V = 0.00367\left(A_p / D_s d_o\right)^{0.29} \left(\Delta P_L \rho_L / \mu_L\right)^{0.2} \tag{14}$$

Figure 5 shows the injection plate with a pressure swirl injector for a liquid nitrous oxide mass flow rate of 150 g/s, to yield a thrust of about 400 N. The pressure transducers and thermocouples are also shown in Fig. 5.



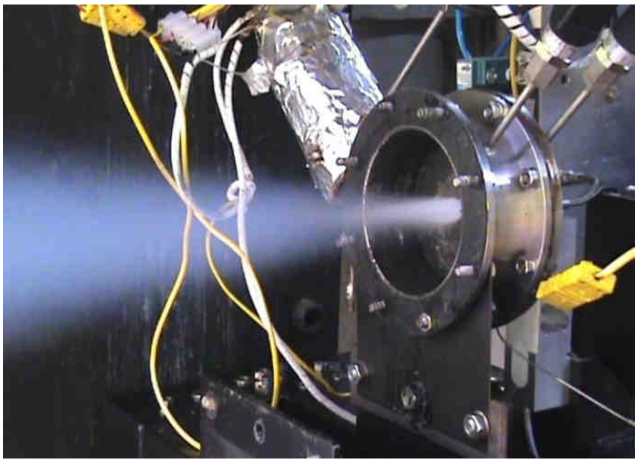
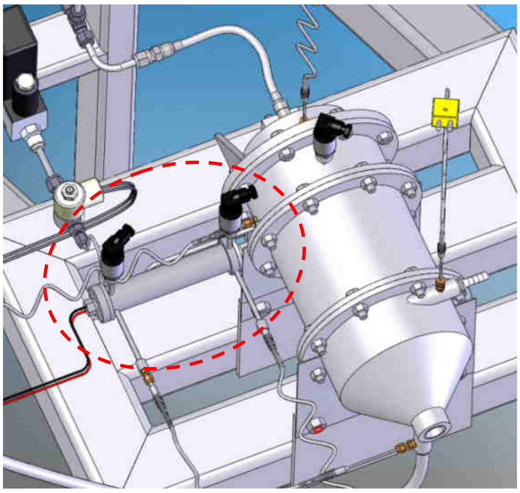


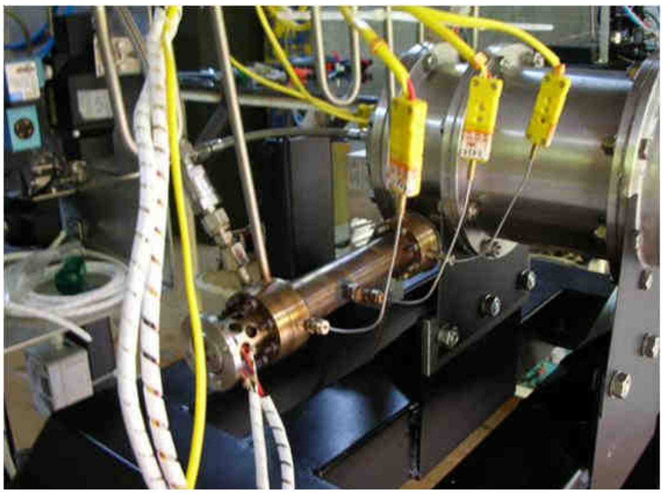
Figure 5. Design of hybrid thruster injector system and photo of an injection test.

Figure 6 shows the thruster ignition system which includes an electrothermal catalytic chamber, designed to decompose a small amount of the oxidizer and to release heated oxygen at the grain surface. The ignition system has a separate gaseous oxidizer feeding line.

The design of the electrothermal catalytic ignition system is described in detail by Albuquerque et al. (2008a,b). Using the previous equations and input data, it was obtained C_d = 0.33, C_F = 1.43 for N_2O and 85% H_2O_2, and C^*= 1613 m/s for N_2O and C^* = 1564 m/s for 85% H_2O_2.



a) View of the ignition system



b) Photo of the ignition system.

Figure 6. Thruster ignition system.

2.3. Test Bench Design

The test bench includes support, thrust balance, oxidizer tanks, pressurizer tanks, feed lines, pressure and temperature transducers, Coriolis mass flow rate controller and meter for the oxidizer, valves and data acquisition system. Figures 7, 8a and 8b show, respectively, a scheme, a 3D view and a photo of the test bench with the thruster assembled.

Figure 9a shows the propellant tank, the tank weighing system and the control valves. Figure 9b shows the thrust balance. The oxidizer is stored at 80 bar in a 3.7 l tank, pressurized by gaseous nitrogen stored at 200 bar. The oxidizer tank is weighed by a 500 N load cell.

A relief valve is set at 100 bar in the oxidizer tank. A vent tube is used during the oxidizer filling, to check if the liquid level is adequate. After the liquid attains the correct level it leaks out through the vent at the top of the nitrous oxide tank. The tank ullage allows liquid expansion due to temperature and pressure variations and facilitates the pressurization.

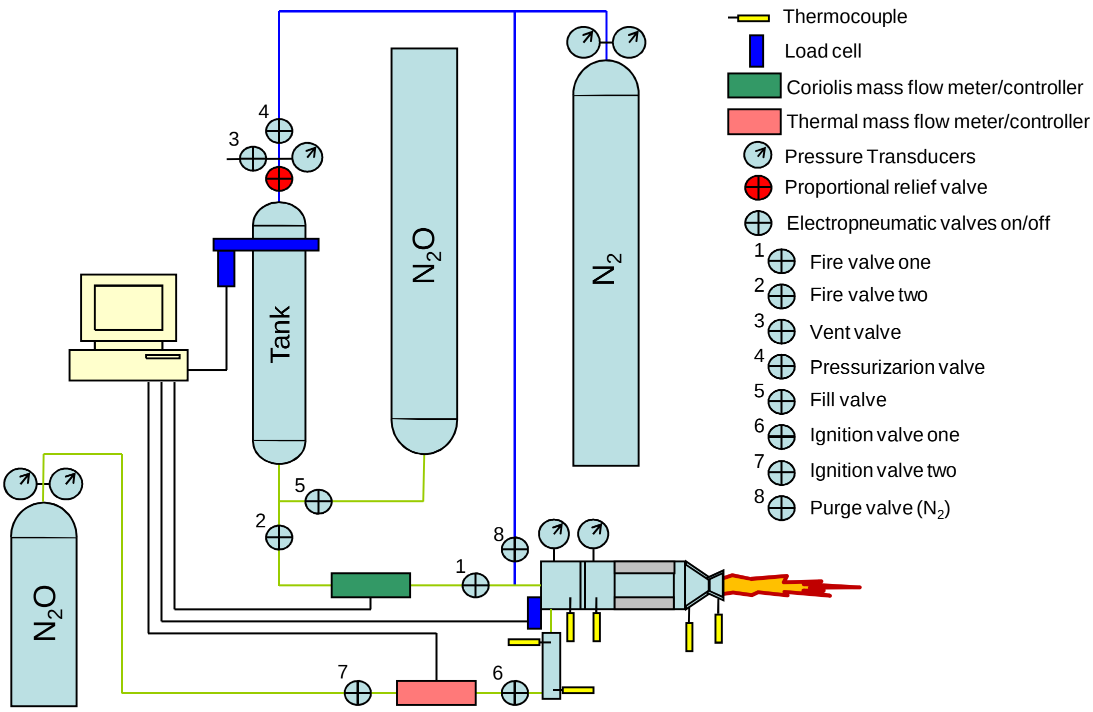
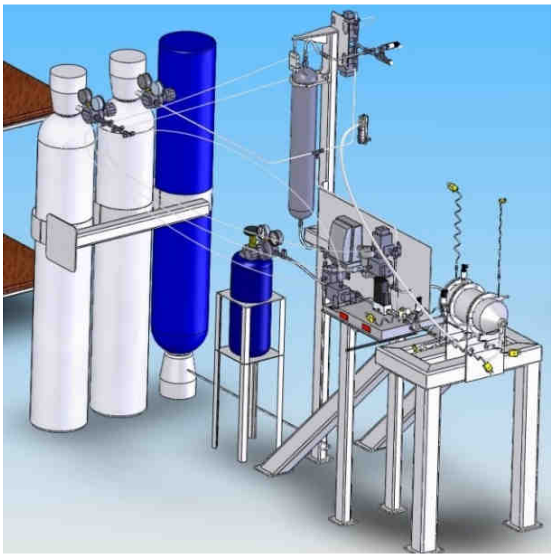


Figure 7. Scheme of the test bench with the thruster assembled.



a)3D view of test bench and thruster

b) Photo of the test bench and thruster

Figure 8. Test bench with the thruster.

The thrust is measured by a 1000 N load cell fastened to the injection section. The thruster is supported by flexible blades which are deflected during the thruster firing.

Pressure transducers are placed before the injection plate, at the pre-combustion chamber and at the oxidizer tank. Thermocouples measure inlet and exit temperatures of the cooling water, the oxidizer temperature before and after injection and the oxidizer temperature at the exit of the catalytic chamber. The oxidizer and pressurizant flows are controlled by 8 electro-pneumatic on/off valves. A Coriolis meter and controller is used to control the mass flow rate of the liquid oxidizer flowing in the thruster.

The signals from the transducers and thermocouples are sent to a data acquisition system operated by LabView software. The valves are opened or closed by the same software.

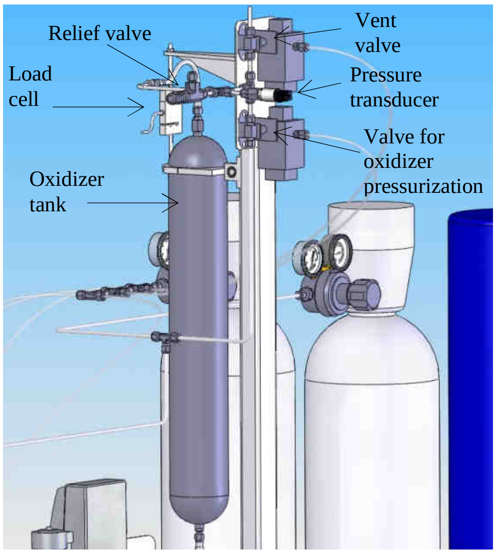


Figure 9a. Propellant storage system.

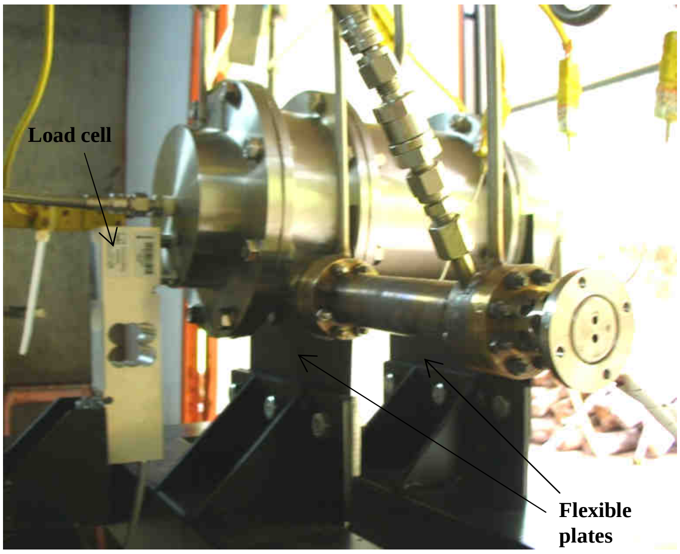


Figure 9b. Thrust balance.

3. RESULTS AND DISCUSSION

Preliminary tests of the ignition system were made to verify the repeatability and efficiency of ignition. Several tests of the hybrid thrusters were also performed. Figure 10 presents a photo of the opened engine for test of the ignition system. Figures 11 shows a photo of a test.

Figure 12 and 13 show the pressures, oxidizer mass flow rate and the thrust curves measured in preliminary tests of the hybrid thruster. The initial spike in the curves is depicted inside the blue circles in both figures. It is caused by the Coriolis meter and controller which it was totally open before firing opening of the on/off valves 1 and 2. This problem was solved by keeping the Coriolis meter and controller closed before opening of the on/off valves. In the test depicted in Figure 13 the cooled nozzle was used, but without the water flow which caused overheating and damage to the nozzle throat. The air kept in the nozzle jacket, instead of the water flow, behaved as a thermal barrier. The red line in the figure indicates the moment of the throat rupture. It can be seen in both figures 12 and 13, that all curves follow the adjustment of the mass flow rate made by the Coriolis meter and controller up to the maximum 540 kg/h set before test. The test shown in Fig. 12 was set for 7 s oxidizer flow whereas the test depicted in Fig. 13 was stopped just after 13 s.

Figures 14 and 15 show, respectively, specific impulse, *Isp*, and thrust coefficients, C_F, calculated from the tests in the time interval considered for analysis. In the test depicted in Fig. 14 the average *Isp* was about 230 s whereas the average C_F was about 1.13. In the test depicted in Fig. 15 the average *Isp* was about 226 s whereas the average C_F was about 1.18. In both Figures it can be observed a slight decrease in thrust coefficient as the oxidizer flow rate increases, caused by a higher increase in chamber pressure than in thrust when the mass flow rate of oxidizer is increased.

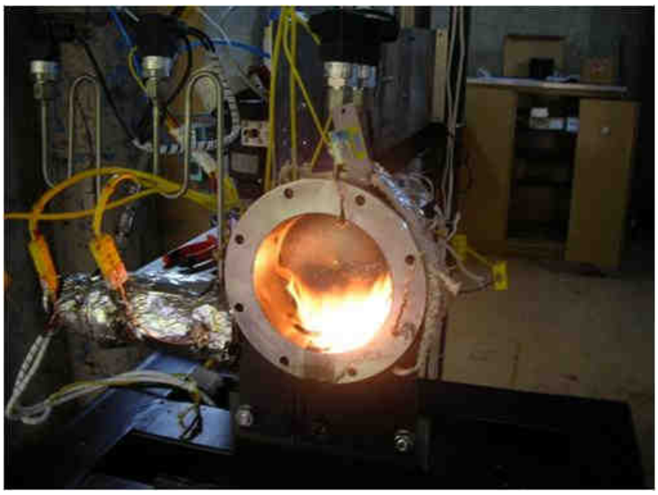


Figure 10. Test of the electrothermal catalytic ignition system.

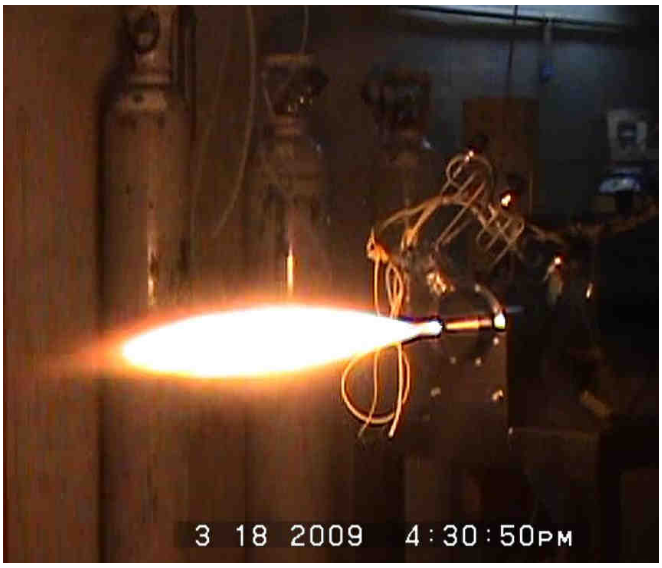


Figure 11. Exhaustion plume of the hybrid thruster.

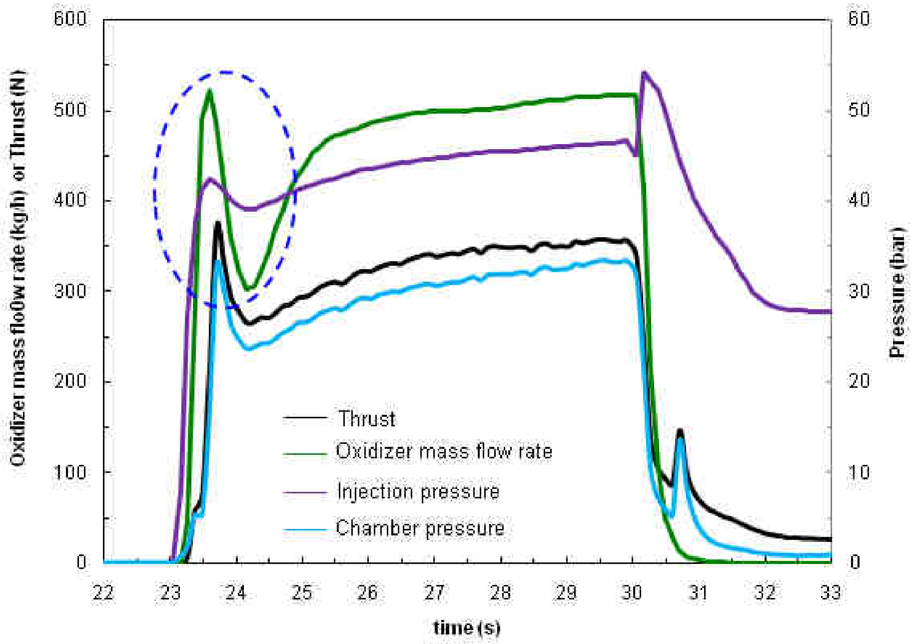


Figure 12. Test results of the hybrid thruster without the jacked nozzle.

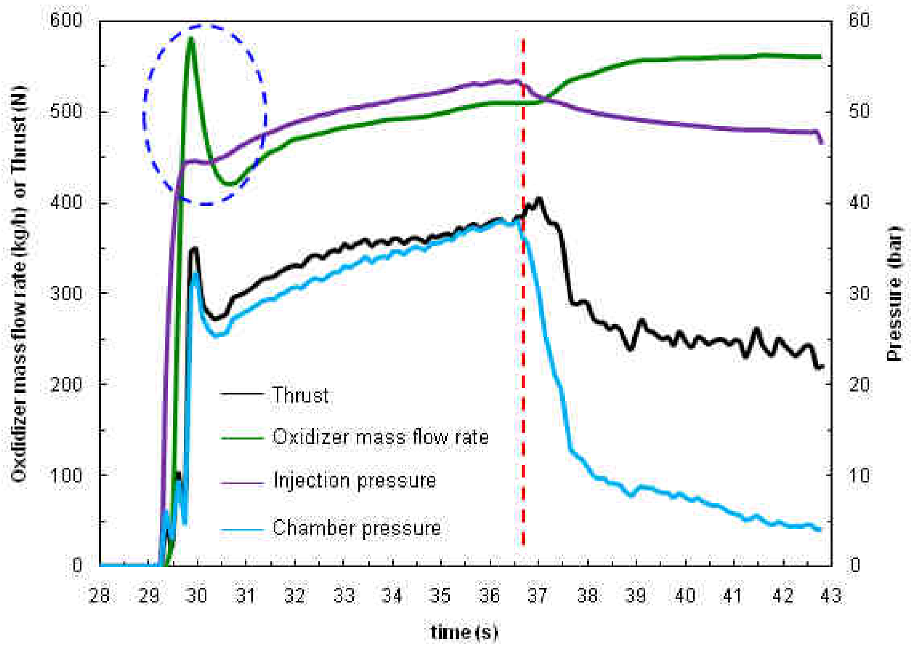


Figure 13. Test results of the hybrid thruster with the jacked nozzle.

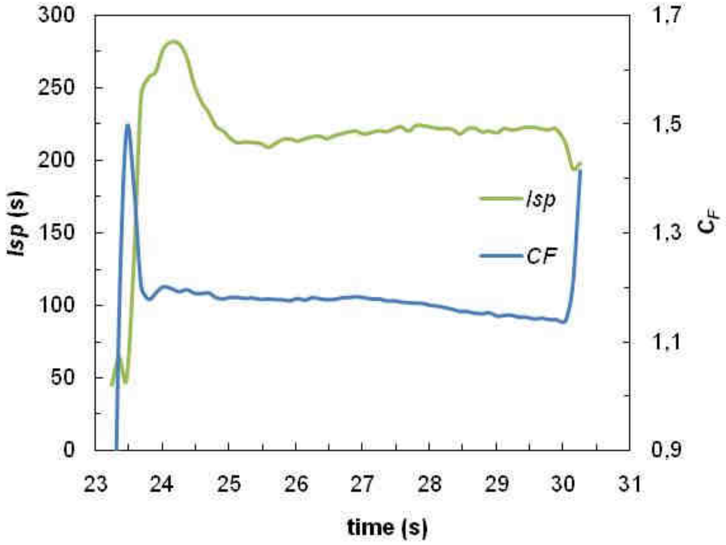


Figure 14. ***Isp*** **and** C_F **calculated from tests without the jacked nozzle.**

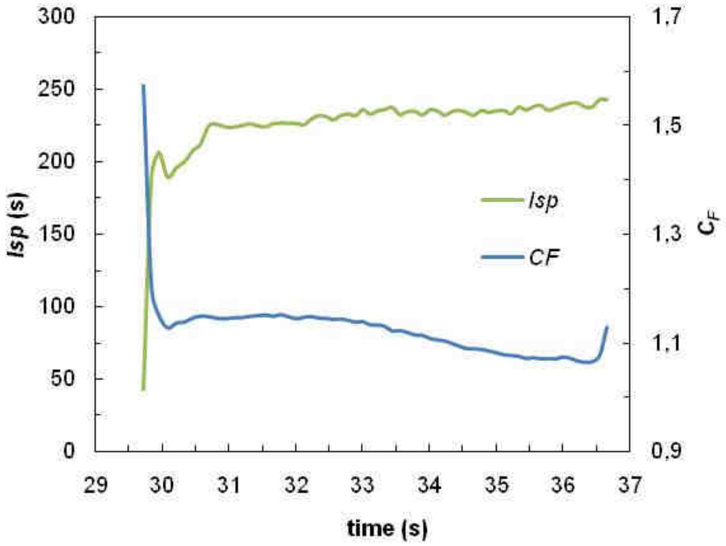


Figure 15. ***Isp*** **and** C_F **calculated from tests with the jacked nozzle.**

4. CONCLUSIONS

This paper described the development of a bench for testing hybrid thrusters up to 500 N and the design of a 400 N hybrid thruster for initial tests. The bench main systems were presented, including propellant storage, pressurization, ducting, flow control, data acquisition, thrust balance and diagnostics. The design process of a simplex pressure swirl injector for nitrous oxide at 80 bar was also presented. Initial tests of the 400 N thruster using paraffin and nitrous oxide as propellants yielded specific impulses 226-230 s with thrust coefficients 1.13-1.18.

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6. ACKNOWLEDGEMENT

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