COB09-1735 CHARACTERIZATION OF ROCKET FOR RESCUE SYSTEM

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Abstrac. *This paper describes the design of a rocket used in the rescue system. This system is designed for quick deployment to enable the rescue of store launch from the lowest possible height. In traditional systems the parachute is gradually pulled out from the top, exposing the length of the material to distortion by air currents and damage by contact with airframe or detached parts thereof. In the system proposed, the canopy is kept contained in a harness until the suspension ropes are fully extended at security distance above the store, where it is then extracted for safe inflation. This design minimizes the danger of damage to the fabric and suspension ropes during deployment. Inflation of the canopy starts in 4.5 to 5.0 seconds from activation of the system.. In order to investigate the performance of rocket used in the recuperation system, it is presented a test matrix and respectively results obtained during the development of the prototype of the motor. The requirements of design were defined based on the similar products.*

Keywords*: Propulsion, parachute extraction, rocket design, SIRAC*

1. INTRODUCTION

This paper describes the design of a rocket used in the rescue system named SIRAC (Sistema de Recuperação Aérea de Carga), project (2003). This system is designed for quick deployment to enable the rescue of store launch from the lowest possible height. In traditional systems the parachute is gradually pulled out from the top, exposing the length of the material to distortion by air currents and damage by contact with airframe or detached parts thereof. In the system proposed, the canopy is kept contained in a harness until the suspension ropes are fully extended at security distance above the store, where it is then extracted for safe inflation. This design minimizes the danger of damage to the fabric and suspension ropes during deployment. Inflation of the canopy starts in 4.5 to 5.0 seconds from activation of the system. The system is designed with reserve capacity to work even under extreme conditions.

This is the second work published about SIRAC system. Silva and Silva (2005) describes a model of experiment designed to investigate the dynamic of flight of the rescue system based in this rocket. The launch conditions has been specified through of the simulation of certain number of launchings in different flight conditions in order to estimate design parameters, which as, horizontal range, opening shock, terminal velocity, time of flight and variation in angle of trajectory. The shortcoming of those initial estimates is the point-mass approximation. The simulations of the trajectory of the center of gravity fail when the relative motion between the parachute and the store needs to be considered. However, the technique adopted display a lot of the features of dynamic of flight of the CG of parachute/payload system that can be used for pre-specify the velocity, parachute area and altitude of launch. One of the principal constrictions of that simulation was the thrust of solid rocket. In this context, in order to investigate the performance of rocket used in the recuperation system, it is presented in this work a test matrix and respectively results obtained during the development of the prototype of the motor. The extraction system is depicted in the Fig. 1. The gas generator is to ignite the solid rocket. The requirements of design were defined based on the similar products. The principal purpose of this work is evaluate the solid rocket (see Figure 1) proposed to achieve the following requirements: burn time: 0.75 s and thrust: 65 kgf in order to produce a total impulse of 460 Ns.

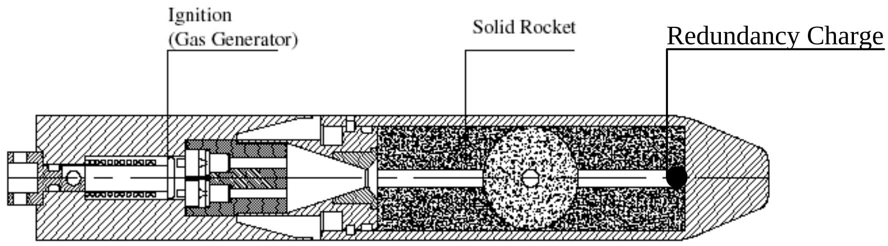


Figure 1. Rocket Motor

2. TEST MATRIX

Figure 1 illustrates a first idea about the geometric configuration of the motor. Two types of energy conversion processes occur in any propulsion system, namely, the generation of energy which is really the conversion of stored energy into available energy and, subsequently, the conversion to the form in which a reaction thrust can be obtained. The kinetic energy of ejected matter is the form of energy useful for propulsion. In this context, the principal concern in this development is to identify, exactly, the thrust contribution of the isolated solid rocket since there are more than one thrust contributions in the whole system. In fact, it is possible to identify three (3) contributions in terms of thrust. Obviously, the first one is the thrust due to the solid rocket. The second is related to the ignition system. This system will supply hot gas to the ignition of grain. The procedure to start the gas generator is based on the impact of a piston with two projectiles filled with explosive charge. With the impact it is released a great quantity of hot gas, which is driven to the chamber combustion of solid rocket. Its energy will ignite the grain. Thus, the second contribution is due to the spring which acts on the piston and the sudden release of energy in the gas generator system. Finally, the last contribution is associated to the redundancy charge (BP – Black Powder). It is put next to the nose of the rocket in order to guarantee the ignition of the system. In another words, it is redundancy system. Based on the above mentioned, it was defined the matrix test which capture all the contributions in an isolated way in accordance to the Tab.1

Figure 2 depicts the results obtained for thrust and chamber pressure. It was used a pressure sensor WTP-4010 (2901psi). This test does not include ignition system (gas generator), just powder to initiate the propellant. It is good to remember that the integrated area under the chamber pressure curve is proportional to the total impulse. So, the total impulse (or the chemical energy released) in combustion stays essentially constant during the test. Figure 3 shows the picture with the characteristics of the fire test. It is important to stress here that it was achieve a stable combustion, without overshoot, in terms of thrust and pressure, and the thrust attained the requirements defined for extraction of parachute. This test will be used as a reference in the characterization of rocket motor, item 3 of this report.

Table 1. Test Matrix (throat nozzle: 6 mm).

Test Configuration	Characteristics	Figure
Rocket Without Ignition System	Ignition: Powder Burning time: 1,1708 s Thrust: 44,967 kgf Chamber pressure: 81,34 bar	1 and 2
Rocket With Ignition system	Ignition: Gas Generator Burning time: 0,7688 s Thrust: 68,099 kg Chamber pressure: 106,92 bar	3
Rocket plus With Ignition system plus Powder redundancy	Ignition: Gas Generator Burning time: 1.1125 s Thrust: 53,308 kg Chamber pressure: 84,572 bar	4

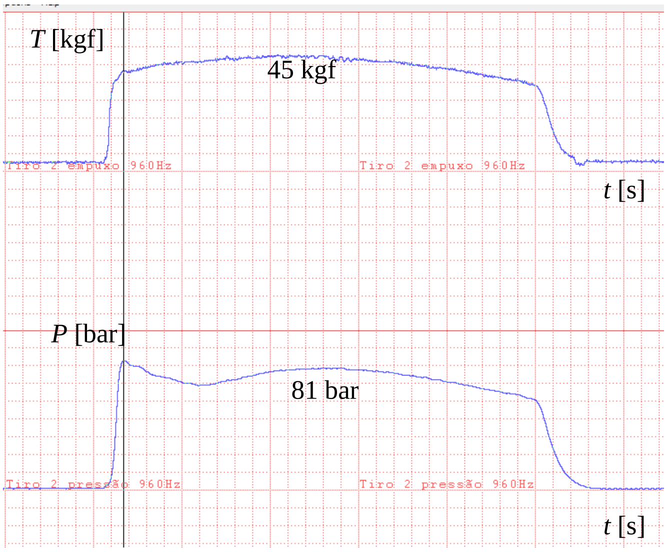


Figure 1. Shot without ignition system

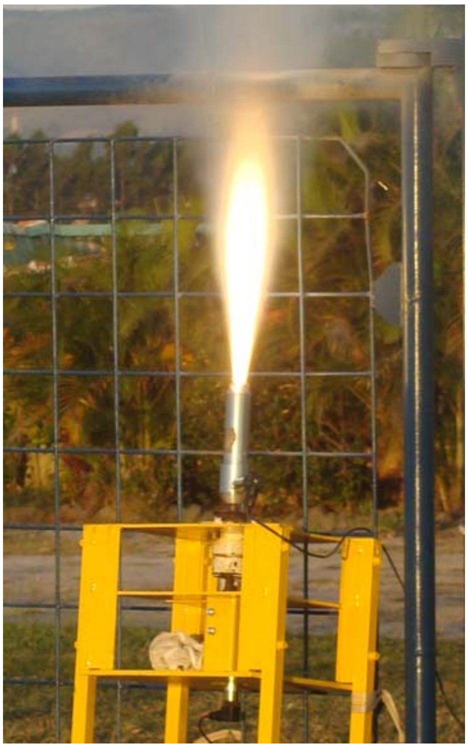


Figure 2. Picture of test without ignition system

Figure 3 shows the test with ignition system. It is expected that the thrust obtained in this test be greater than those from the test without ignition system since the gas generator contribute with the increase of chamber pressure. In this context, it is expected the reduction of burning time too. The main objective of SIRAC system is to extract the parachute from the aircraft using a solid rocket. In fact, the solid rocket will pulled out the harness with the parachute from the top and, naturally, it will expose all the system to the plume as a result of the combustion process. Thus, the reduction of burning time is a good result since the thrust requirements are achieved. It has been found an overshoot in thrust (167.762 kg) and chamber pressure (126.1984 bar) during the test with ignition system. In accordance with experience of group, this overshoot is not a big problem since the aircraft structure is robust enough to resist this load. But, the oscillatory behavior after the overshoot is a concern which must be studied during this development. It is associated to the expulsion of ignition system during the test bank.

Another important observation is associated to the effect of ignition system at the solid rocket. The results plotted at the Figs. 1 and 3 indicate that the propellant had an increase of temperature after the explosive charge from ignition system. This can be a problem since the system will be used in different conditions of temperature (depending on the place of use).

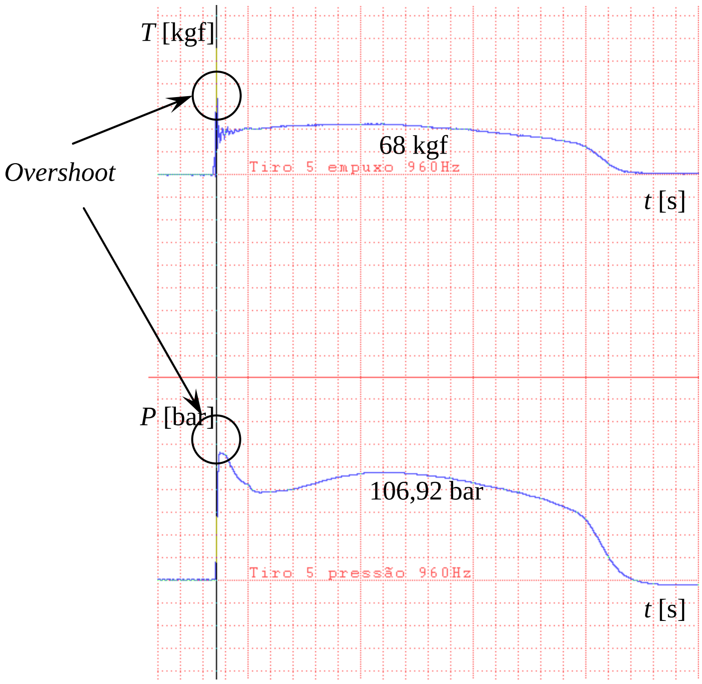
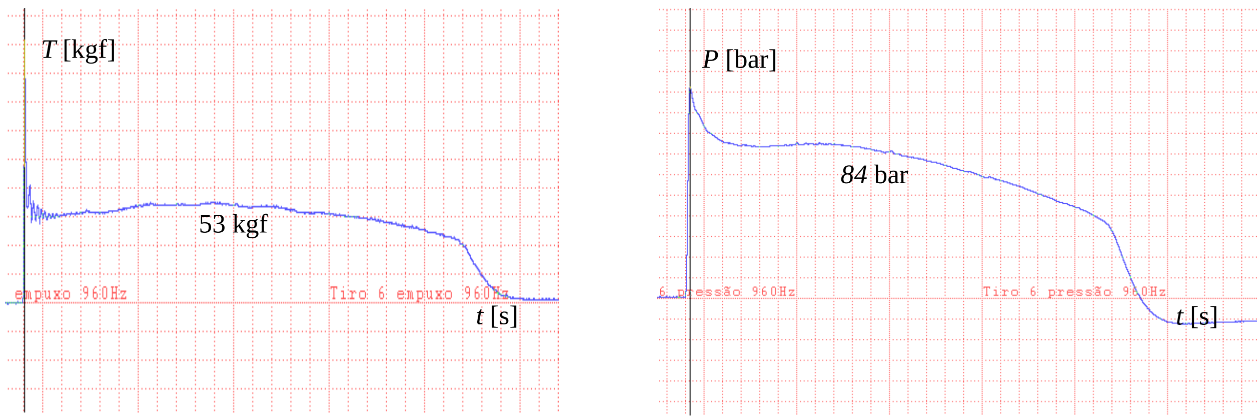


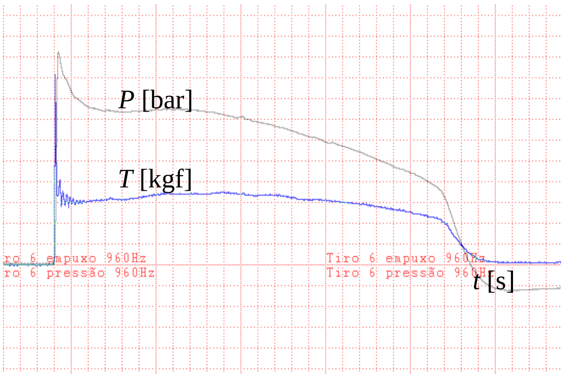
Figure 3. Shot with ignition system

Finally, Fig.4 illustrates the test results obtained for the standard solid rocket, which is, ignition system plus solid rocket plus ignition redundancy. It has been found an overshoot in thrust (138.577 kg) and chamber pressure (114.62 bar) during the test execution. It is interesting to note that the presence of ignition redundancy reduced the overshoot in thrust and chamber pressure and, in consequence, it increase the burning time. This phenomenon can be explained from the influence of redundancy explosive charge. The heat generated from the ignition system is directly transferred to the propellant grain and throat. The material of throat is not thermally resistant. During the tests with the full configuration (igniter + redundancy charge + rocket motor) it was possible to note that the nozzle had been eroded (increasing 10% in area). In this context, the pressure chamber has fallen down significantly and the burning time increase when it is compared with the test without redundancy system.



(a) Thrust

(b) Chamber Pressure



(c) Thrust and Chamber Pressure

Figure 4. Shot: Standard Rocket

3. CHARACTERIZATION OF ROCKET MOTOR

A Forerunner rocket motors used to be propelled by BP (black powder) and even now, smallest ones for whether military or civilian purposes are still mostly powered by it. However, in 1960's a breakthrough was achieved when a novelty, the oxidizer AP (Ammonium Perchlorate) came out in the mid of the Cold War. The AP-based propellant can deliver more specific impulse and energy than BP used to do. Since then, just after the debut, AP turned out to be the workhorse for the solid rocket industry over the twenty century. By now, manufacturers set about making use of this state-of-art oxidizer in small rockets to extract parachute from the housing.

In this section, rocket propulsion calculation is carried out for a parachute AP-based extractor. The pressure vs. time experimental plotting that come of the motor's fire-test is the starting-point for the calculations herein, the further information is provided from the grain's geometry and the propellant formulation. The aim is to compare the experimental thrust to the theoretical one in order to learn the actual performance of this kind of motor.

3.1. Propellant grain

The propellant is cast in cylindrical cases by gravity and under vacuum. After cured, the grain is machined at the core to get the hole done. Two cylindrical grains are lined up through the ends and set up into the motor housing according to the Fig. 5. The casing wall is in blue and the propellant grain in red-brown. The propellant formulation is given in the Tab. 2.

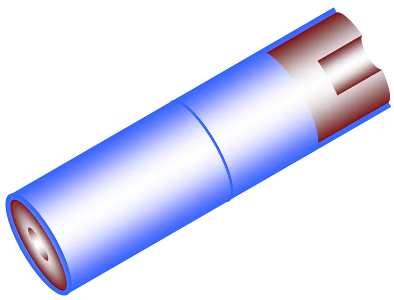


Figure 5. Twin Rocket Motor

Table 2. Propellant formulation.

Ingredients	**Description**
Oxidizer	Ammonium Perchlorate
Binder	HTPB
Fuel	Aluminum
Catalyst	Iron Oxide

3.2. Fire test

The fire-test results are detailed in the next item. Nevertheless, two outcomes are displayed in the Fig. 6, namely, the chamber pressure vs. time and the motor thrust vs. time. The very first peak on the pressure graph can hardly be ascribed to the igniter peak or erosive burning as usually end up being made. In fact, it is likely to be caused by a catalyst in the propellant formulation. Furthermore, this peak may bring about blast if the igniter goes off at higher temperatures.

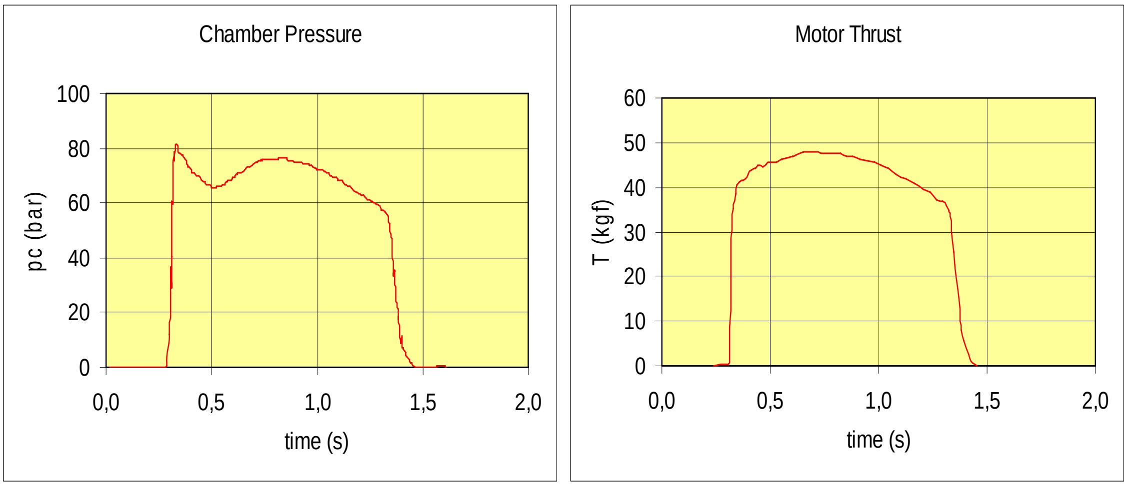


Figure 6. Fire-test pressure and thrust

3.3. Burning surface and burning rate

In order to characterize the rocket motor, it is necessary to known the burning surface and the burning rate. The burning surface (S_b) is obtained from the geometric characteristics of motor. For the twin cylindrical motors S_b is given by the Eq.(1), where *D* is the outer grain diameter, *d* is the initial inner grain diameter, *e* is burned surface thickness and l is the grain length. All units are given in millimeters.

$$S_b = 4 \cdot \frac{\pi \cdot \left[D^2 - (d+e)^2\right]}{4} + 2 \cdot \pi \cdot (d+e) \cdot (l - 2 \cdot e) \tag{1}$$

The burning surface of a propellant grain recedes in a direction essentially perpendicular to the surface. The rate of regression, usually expressed in mm/s, is the burning rate. In this work, the burning rate (r_b) is given by the classical power expression, which is:

$$r_b = a \cdot P^n \tag{2}$$

Where the variable *a* is the burning rate for pressure equal to unit and *n* is the burning exponent. The chamber pressure (*P*) is expressed in MPa. Table 3 shows the values estimated in this work.

Table 3. Regression rate parameters.

Parameter	Value	Unit
a	5	-
n	0.4	-
P	-	MPa
r_b	-	mm/s

The two equations above in conjunction with the fire-test pressure curve plus grain's geometry and propellant's ballistic information were used to obtain the temporal variation of burning surface and burning rate, given in the Fig. 7.

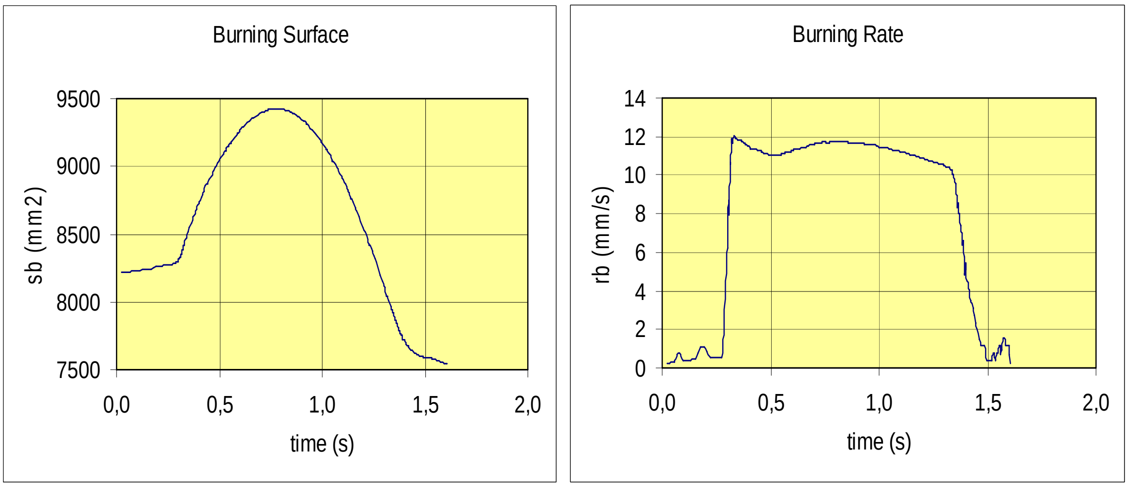


Figure 7. Burning surface and burning rate.

3.4. Mass flow and specific impulse

The burning rate of propellant in a motor is a function of many parameters and at any instant governs the mass flow rate ($\dot{m}$) of hot gas generate and flowing from the motor. The classical expression derived for $\dot{m}$ obtained during a stable combustion is given by Eq. (3). Here, ρ_p is the propellant density.

$$\dot{m} = S_b \cdot r_b \cdot \rho_p \tag{3}$$

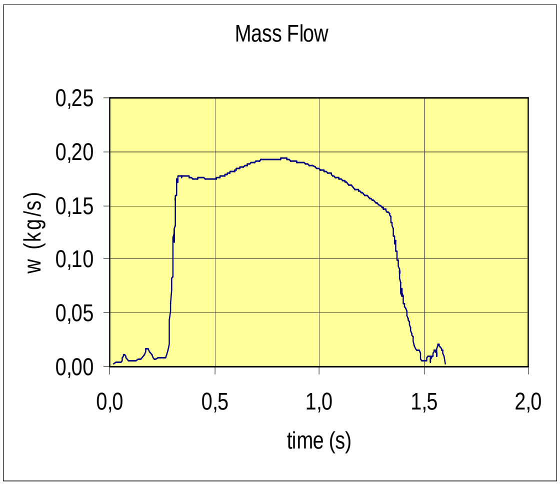
The specific impulse (I_S) is the total impulse per unit weight of propellant. It is important figure of merit of the performance of a rocket propulsion system. If the total mass flow rate of propellant is $\dot{m}$ and thrust developed at the standard acceleration of gravity at sealevel g_0 is 9.8066 m/s^2 is constant, it is possible to write:

$$I_S = \frac{F}{\dot{m} g_0} \tag{4}$$

By using the ICT® Thermodynamic software, the specific impulse it is as a function of the chamber pressure and the propellant formulation. The results were roughly correlated by the expression (5):

$$I_S = 43{,}308 \ln(P) + 79{,}36 \tag{5}$$

with the chamber pressure (P) expressed in bar. One may notice that the chamber pressure can not be zero; the fact of the matter is the specific impulse has got to be determined by assuming the unit for the lowest pressure. Figure 8 shows the temporal variation of mass flow and the specific impulse, respectively.



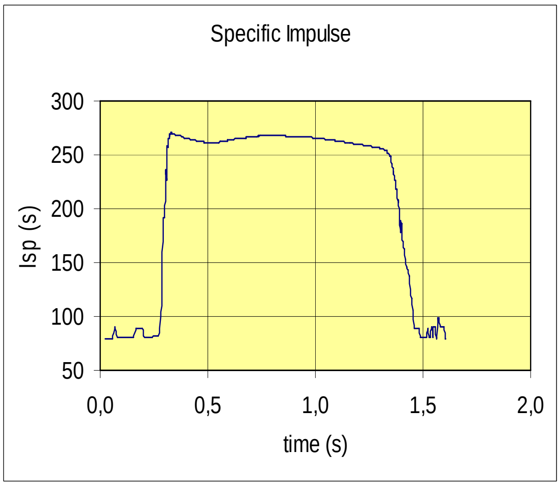


Figure 8. Mass flow and specific impulse.

3.5. Thrust

The thrust (*T*) is the force produced by the rocket acting upon a vehicle. In a simplified way, it is the reaction experienced by its structure due to the ejection of matter at high velocity, Eq. (6), (Sutton, 2001):

$$T = I_S \cdot \dot{m} \tag{6}$$

where *T* is expressed in kgf, specific impulse in second and $\dot{m}$ is expressed in kg/s. The fire-tested thrust agrees with the theoretical results as shown in the Figure 9, despite the absence of the initial peak in the experimental curve. The likeness between the two curves comes down to the closeness of the motor performance to the one with adjusted nozzle.

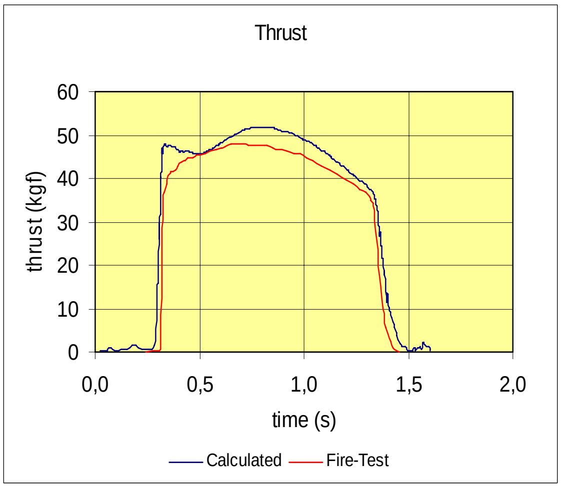


Figure 9. Thrust.

4. CONCLUSIONS

This paper describes the characterization of a solid rocket used in a rescue system. It has been investigated the performance of rocket and it is defined a test matrix which should be used during the development of the prototype of the motor in order to isolates the different contribution of ignition system, solid rocket and ignition redundancy thrust. It can be conclude that the performance of rocket defined in terms of burn time (1.11 s) and (thrust: 53,33 kgf) achieved the requirements desired, which are: burn time: 0.76 s and thrust: 65 kgf. It has been found thrust and chamber pressure overshoot, however, it is not a problem since the aircraft structure is robust enough to resist this load, moreover, the system has been tested in a real flight conditions. However, it was identified any problems which must be studied in the future developments:

- The ignition system has a significant influence in the burn process. The parameters that driven this influence must be identified and quantified;
- The overshoot in pressure can be a problem if the rescue system is used in tropical regions (high temperature). It is important to test the system with another initial conditions of temperature;
- The redundancy system has a significant influence in the heat transfer at the throat of nozzle. From the experimental results it is interesting to investigate the possibility of change the nozzle material;
- The requirements are defined in terms of burning time and maximum thrust. It is necessary evaluate the impact of total impulse in the requirements specified for the rocket.

5. REFERENCES

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