DESIGN AND FLIGHT TEST OF A PARAFFIN BASED HYBRID ROCKET

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Abstract. *This paper shows the current status on hybrid propulsion technology carried by the Laboratory of Energy and Environment from University of Brasilia. Specifically we describe the development of a new hybrid motor, based on paraffin as the solid fuel and N_2O as the oxidizer. In order to demonstrate the technology, a twelve months project was conceived to design, build and test a hybrid rocket capable of reaching 2600 meters altitude. The project was divided in three main phases: (i) static tests of the 500 N hybrid motor, (ii) design and construction of a hybrid rocket and (iii) launching of a model hybrid rocket. The regression rate tests confirmed that paraffin is a promising fuel for hybrid rockets. We were able to get reliable ignition and operation in the majority of the firings. Two rockets were then built and launch. The rockets flight and operation was considered a great success. To our knowledge, this was the first hybrid rocket ever built and launch by a research group in Brazil. The group is now leading a project sponsored by the Brazilian Space Agency (AEB) to design, integrate and test a hybrid sounding rocket capable of reaching 20000 m with avionics and recovery system.*

Keywords*: hybrid rocket, paraffin, nitrous oxide, launching.*

1. Introduction

Hybrid rockets use liquid and solid propellants. The oxidant agent, generally liquid, is injected in the combustion chamber containing a solid grain as fuel, developing a turbulent chemical reacting boundary layer. Hybrid rockets have regain attention from the research community due to some advantages over liquid and solid technology, such as: (i) the solid fuel is considered as non-hazardous and non-toxic (ii) there is a larger margin for safety in storage and operation of the fuel; (iii) in contrast to solid rockets, hybrid can have variable thrust and, more important, allowing multiple stops and reignitions; (iv) the products of the combustion of hybrid propellants are considered less harmful; (v) hybrids are more reliable than solid and liquid rockets; (vi) compared to liquid rockets, hybrids need half of the complex system of propellant feeding, since only the oxidant is in the liquid phase; (vii) cracking in a hybrid rocket is nor as catastrophic as in solid rockets and more importantly, (viii) in general hybrid rockets are cheaper than solid and liquids rockets. As a significant disadvantage, hybrids do not generate as much energy per mass of propellant as liquid propellant engines and they are more complex than solid fueled engines. Hybrid suffered from very low regression rates with conventional fuels, like HTPB corrected by using multiple combustion ports. This has limited the application of the technology to small sounding rockets. Notwithstanding, hybrids are under investigation in various research centers. Manufactures, mainly in US are testing hybrid system for different application. A large hybrid rocket was recently tested as a part of the Falcon Small Launch Vehicle development program. When integrated the engine should be capable of placing small satellites in 185 km orbit (www.globalsecurity.org, 2005).

Recently, Ames Research Centre and Stanford University identified a new class of hybrid fuel based on paraffin (Aerospace America, 2002). In their study the fuel regression rate was three to four times faster than conventional hybrid fuels. They claimed that paraffin-based solid fuels could be used in rocket booster since only one combustion port would be necessary, rather than multiple ports for standard hybrid fuels.

Thin-film instability, above the solid fuel, form waves that collapse in small droplets which are entrained and react in the central oxidizer flow. Such process is not observed when burning HTPB, polyethylene and other typical fuels for hybrid propulsion. This additional burning mechanism explains the higher regression rate observed in their experiments.

Many aspects of hybrid propulsion have been under investigation, covering pyrolysis behavior of the solid fuel (Chiaverini et al., 1999), regression rate (Chiaverini et al., 2000) and performance characteristic (Korting, et al., 1987; Risha et al., 2001, Risha et al., 2002; Chiaverini et al., 2001; George et al., 2001) More recently, Karabeyoglu et al. (2004) conducted experiments in paraffin-based hybrid fuels with gaseous oxygen. They confirmed the high regression rate of paraffin compared to other standard fuels.

Hybrid rocket propulsion has been under investigation by the Energy and Environment Laboratory – University of Brasília, since the year 2000 (Viegas & Salemi, 2000; Navarro & Pereira, 2002; Fraga 2003). Ignition, combustion stability and overall performance were first investigated by Viegas & Salemi (2000). Navarro & Pereira (2002) estimated the solid-fuel (polyethylene) regression rate in a broad range of burning conditions.

Fraga (2003) tested the burning characteristics and performance of paraffin based solid fuels with oxygen as the oxidizer. A detailed fuel processing procedure was implemented in order to have the paraffin based solid grain.

With the experience gained in static tests it was proposed a twelve month project to design, build and test a model rocket based entirely in hybrid technology employing paraffin as the solid fuel and nitrous oxide as the oxidant. This paper presents an overview of the project, from conception to launching. To our knowledge, this attempt was the first ever conducted in Brazil. The challenge was proposed, as an undergraduate project (course completion requirement), and was carried out by two students from the Mechanical Engineering Department, University of Brasília, supervised by a professor from the same department.

2. Objectives and Success Criteria

The objective is to conceive, build and flight test a paraffin-based hybrid rocket with nitrous oxide. A 300 N, average, motor was set as the basic parameter. Maximum rocket altitude would be then a consequence. The project was divided in many phases with the more important listed in the following:

- Extent literature review with emphasis on similar projects;
- Adapt the test stand to work with liquid oxidizer (N_2O);
- Improve the fuel processing technology;
- Improve the ignition system;
- Implement mathematical models for rocket motor performance analysis;
- Identify and acquire model rocket design and flight simulation system;
- Static test the 500 N thrust engine;
- Design the model rocket;
- Design the oxidizer filling and ignition system;
- Static test the model hybrid rocket;
- Integrate all the subsystem and flight test the rocket;
- Performance analysis.

3. Static tests

Figure 1 shows the motor assembly for static tests. It consisted of a combustion chamber, made of ABNT 1020 steel, with flanges placed in the tube endings. Motor internal diameter was 75.2 mm; a pre-chamber (50 mm) was placed between the oxidizer injector and the paraffin grain. A post combustion chamber was introduced in order to favor burning completion of the reactants, before nozzle expansion.

Burning of the solid fuel is interrupted after closing the oxidizer valve followed by nitrogen injection in the motor. Nitrogen helped to freeze the reactions and, additionally, prevented unintentional self-ignition.

Paraffin was purchased, in granulated form, in candle store. Tubes made of plastic (PVC – polyvinylchloride) with threaded caps in both ends were employed as the grain casing. Melted paraffin, with a blackening agent, was poured in the plastic casing and was allowed to rotate along its longitudinal axis for about one and half hour. The fuel grain port was obtained in a straightforward manner, by virtue of phase changing shrinking and the amount poured. Therefore, no post machining was necessary. Figure 2 summarizes the main parameters of the 500 N engine. The ignition system consisted of a black powder layer added to the wall oxidizer entrance. A wire, through which an electric current was imposed, ignited the black powder. Temperature dissociation of nitrous oxide is about 560 °C. To guarantee this level of temperature we used a steel wool near the main oxidizer entrance that was ignited by the black powder. Part of the heat of the product gases heated the grain port, thus allowing the initiation of the combustion process (reacting turbulent boundary layer).

Two main data had been collected for the motor performance validation: (i) the pressure in the combustion chamber and the thrust. Thrust was measured by a cylinder-piston device used in car brakes, previously calibrated to convert pressure to force (load cell). We used two pressure sensors, model Wika ECO-1. The signal from the pressure transducers were converted to voltage and routed to a HP model 35665 Dynamic Signal Analyzer. The data are recorded and later converted to an ASCII file that can be processed by spreadsheet software.

Our test stand allowed engine firings in the horizontal position, common for the type of investigation. In the qualification phase the model rocket was tested in a vertical stand ("test as you fly"). With nitrous oxide we tested the engine injecting in both phases, liquid and gaseous. In gas phase, the flow of nitrous oxide was controlled by nozzle with throat diameter of 5.9 mm. This would give a 0.8505 kg/s of nitrous oxide.

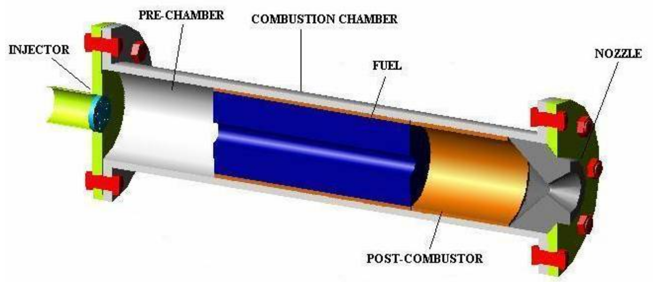

Figure 1: Hybrid motor assembly.

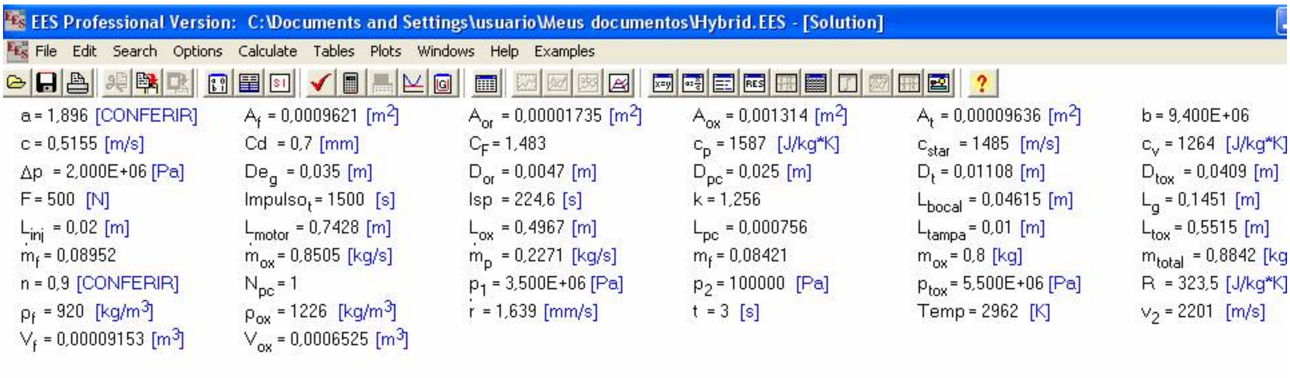

Figure 2. Rocket main parameters as shown by the EES application.

Figure 3 shows thrust and chamber pressure measurements against time. The measured regression rate was estimated a 2.2 mm/s. It can be seen that the motor delivered in the first five seconds more than 200 N of thrust and then with the drop of pressure, thrust decays to less than 200 N. The reason for such a low thrust lies in the chamber pressure miscalculations. Discharge coefficient value and pressure balance between oxidizer tank and combustion chamber was misestimate. The theoretical pressure was set to 35 bar. Total pressure, in Fig. 3, should be obtained by summing the dynamic pressure, which was estimated to be less than 1.0 bar. Important conclusion, though, could be draw regarding motor operation. The rocket parameters were transferred to the Hybrid Design Program, corrected for the measured chamber pressure. The thrust against time curve can be seen in Fig. 4. When correcting the observed pressure, theoretical thrust was in a better agreement to the measured one.

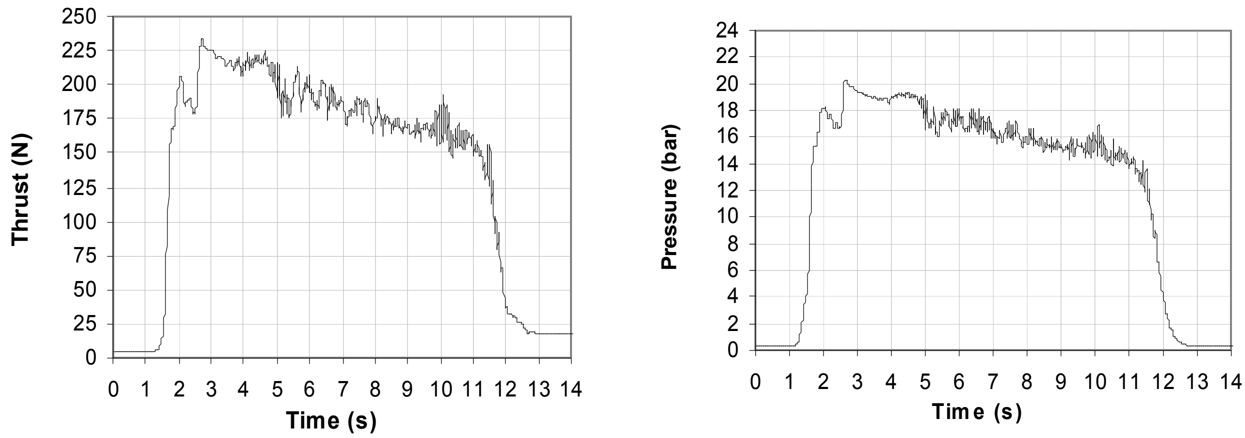

Figure 3. Motor thrust and chamber pressure.

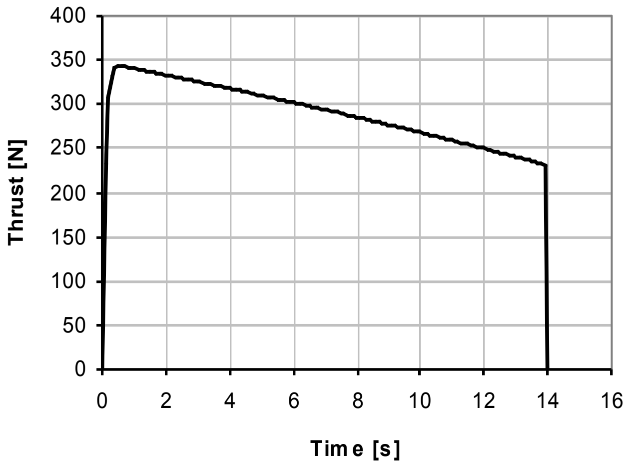

Figure 4. Thrust curve against time for the 500 N hybrid motor.

4. The rocket engine

The rocket motor conception was entirely based on the engine tested statically, but with another oxidizer injector. Based on the experimental results and with the simulation shown on Fig. 3, it was decided to build a 300 N hybrid motor. A rocket was then conceived for such engine. Some important parameter necessary to design the motor are listed in Table 1. More detail of the motor design can be seen in Cozac e Santos (2004).

Table 2: Base project parameters.

Fuel	**Paraffin $(C_{20}H_{42})_n$**
Oxidizer	**Nitrous oxide (N_2O)**
Thrust, average	**300 N**
Oxidizer tank pressure, initial	**50 bar**
Combustion chamber pressure, initial	**20 bar**
External pressure	**1 bar**
Burning time	**3 s**

The main engine performance parameters were estimated by a computer code written in EES platform as well as with the aid of the "Hybrid Design Program - HPD" - version 1.1.0. Table 3 presents the main parameters of the engine.

Table 3. Rocket engine main parameters.

Propellant	Combustion	Nozzle
N_2O mass = 0.8 kg	Characteristic velocity = 1485 m/s	Throat diameter = 11.1 mm
Fuel mass = 0.0765 kg	Initial chamber pressure = 20 bar	Entrance diameter = 30.9 mm
Ox/fuel = 9.80	Specific Impulse = 224.6 s	Exit diameter = 22.36 mm
Fuel Density = 0,745kg/m^3	Total Impulse = 1500 Ns	Convergence angle = 24°
Total mass = 0.8265 kg	**Oxidizer Tank**	Divergence angle = 95°
Grain length = 150 mm	Initial tank pressure = 50 bar	Length = 45.56 mm
Port Diameter = 25 mm	Oxidizer flow = 0,180 dm^3/s	External diameter = 40.9mm
Fuel = solid paraffin	Material = Al alloy 6101 T6	Material = graphite

These parameters were calculated as follows.

The total impulse of the engine is given by

$$I_t = F.t\,. \tag{1}$$

In Eq. (1), F is the rocket thrust and t is the burning time.

The specific impulse, considering the coefficient of thrust, can be found through the following equation:

$$I_{sp} = \frac{C_F . c^*}{g_0} \tag{2}$$

where g_0 is the gravity acceleration at sea level, c* is the characteristic velocity, given by

$$c^* = \frac{v_2}{C_F} . \tag{3}$$

In Eq. (3) v_2 is the expansion velocity of the gases, which equation is

$$v_2 = \sqrt{\frac{2k}{k-1} R T_1 \left[1 - \left(\frac{p_2}{p_1} \right)^{\frac{(k-1)}{k}} \right]} \tag{4}$$

In Eq. (4) T_1 is the chamber temperature and R is gas constant.

The equation for the coefficient of thrust is

$$C_F = \sqrt{\left(\frac{2k^2}{k-1} \right) \left(\frac{2}{k+1} \right)^{(k+1)/(k-1)} \left[1 - \left(\frac{p_2}{p_1} \right)^{(k-1)/k} \right]} + \frac{p_2 - p_3}{p_1} \frac{A_2}{A_3} \tag{5}$$

where k is the ratio of specific heats estimated with the help of a equilibrium program (GASEQ, 2005), p_1 is the chamber pressure and p_2 is the nozzle exit pressure and p_3 the atmospheric pressure. For simplicity the expansion was considered ideal. In this work, C_F was found to be 1.483.

The propellant mass flow rate in the exhaust nozzle is given by:

$$\dot{m} = \frac{p_1 A_t}{c^*} \tag{6}$$

where A_t is the nozzle throat area, which is estimated:

$$A_t = \frac{F}{C_F p_1} \tag{7}$$

4.1. Structural project

The rocket engine was conceived to be a mono-tube type, in which the oxidizer tank and combustion chamber share the external surface. The vessel should stand structural tensions whilst keeping weight to a minimum. Safety was always a primary concerned, overruling rocket performance and a safety factor of two was used. The tank and combustion chamber were separated by a piston-like device which also hosted the oxidizer injector and the filling system. This mono-tube was made of aluminum, 6101-T6 alloy. Aluminum is chemically compatible with the nitrous oxide and is a low density metal with good structural resistance. The thickness of the mono-tube was calculated as follows.

For a cylindrical vessel with a wall thickness t, internal radius r submitted to a pressure p, an element of such vessel is subject to the normal stress σ_1 in the tangential direction and σ_3 in the axial direction, defined by

$$\sigma_1 = \frac{pr}{t} \tag{8}$$

$$\sigma_3 = \frac{pr}{2t} \tag{9}$$

For a tank pressure of 55 bar, aluminum with a yield strength of 75 MPa and an internal diameter of 40.9 mm, the minimum thickness calculated for the tank wall was 1.5 mm. For the combustion chamber, with an operating pressure of 35 bar, the minimum wall thickness found was of 0.95 mm. These values of wall thickness had been calculated using the theory of the maximum shear stress or Tresca yield criterion and confirmed by the Von Mises theory of the maximum distortion energy. Using Tresca

$$\left|\sigma_1 - \sigma_3\right| < \frac{S_y}{N}, \tag{10}$$

where S_y is the aluminum yield strength and N is the safety factor (two).

Using Von Mises theory:

$$\sigma' = \frac{\sqrt{\sigma_1^2 - \sigma_1\sigma_3 + \sigma_3^2}}{N} \tag{11}$$

where σ' is the operating stress in the tube, which must be lesser than the aluminum. For the oxidizer tank a value of operating tension of 13.3 MPa was found. For combustion chamber a value of 4.21 MPa was found. These values are below the aluminum yield tension causing safety margin of 200%. The selected pipe had 48.26 mm of external diameter and a wall thickness of 3.68 mm.

4.2. Motor subsystem

The oxidizer tank was a 561.5 mm aluminum pipe, 48.26 external diameter, composed of a welded end cap and the piston-like injector, which also worked as a boundary for the combustion chamber. Figure 4 shows a photograph of the motor.

Figure 4. Hybrid motor, monotube.

The length of the oxidizer tank was calculated using the density of the nitrous oxide, at ambient temperature and the total mass of the oxidizer. This mass was estimated as 0.8 kg. The minimum length was set to 500 mm, including 5% of ullage, for gaseous nitrous oxide, the remaining is filled with liquid N_2O. The oxidizer injector is an aluminum piston-like device with 30 mm of length and 40.9 mm of external diameter. A compression fitting is threaded in the injector plate through which passes a nylon hose that is responsible for transferring nitrous oxide to the combustion chamber. The seal of the tank and the combustion chamber was provided by a 35 mm diameter Viton O-ring inserted in a groove of 2.8 mm. Viton was chosen because of good resistance to the heat and nitrous oxide, associated to excellent mechanical properties.

The ignition system was placed in the injector. It consisted of black powder, steel wool and an electric wire. This system was assembled in such way that at the moment of the firing, the heat generated was transferred partially to the nylon hose that is severed allowing the oxidizer to flow towards the combustion chamber. Some of heat generated for the burning of the powder initiated vaporization of the solid fuel as well as for chemical dissociation of nitrous oxide.

The combustion chamber, placed in the opposite end of the monotube was composed by the pre combustion chamber (30 mm), the fuel grain (150 mm) and a post combustion chamber (50 mm). PVC was used for the pre and post combustion chambers. This was a key element for thermal protection for the combustion chamber walls. The exhaust gases nozzle was constructed of graphite. The nozzle was machined with a 90° convergence angle and 24°.divergence angle. An O-ring was placed in the nozzle for sealing the combustion chamber. For closing the combustion chamber, an end cap was threaded. Figure 5 shows the rocket main components.

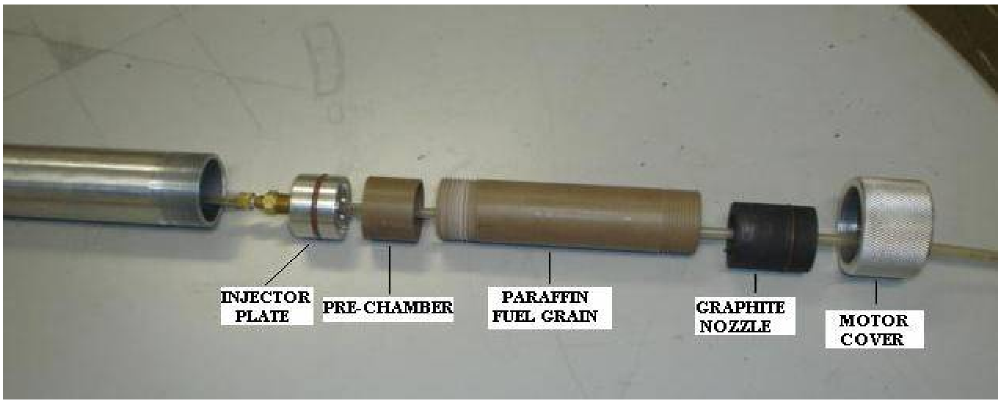

Figure 5. Engine subsystem.

4.3. Motor test

To test the structural quality of the engine, the ignition system and to measure the thrust, a vertical test stand with a spring like dynamometer was constructed. The vertical test stand can be seen in Fig. 6 with the motor ready for firing.

Figure 6. Vertical test stand picture.

Two tests were conducted with acceptable overall performance. We thus considered the motor qualified for launching. Next step then was to integrate the motor to a model rocket. The model rocket main parameters were obtained with the help of a commercial code. Next section explains the methodology.

4.4. Rocket final conception

The rocket final conception was done with the help RockSim version 7.0.4.5 code, from Apogee Components Incorporation. This program is an application for model rocket design and flight simulation system.

The dimension and location of the rocket components can be set, so that the static stability can be inferred. Based on rocket motor external diameter, a rocket caliber was chosen and the main components for stability were added. After conception the rocket can be set on flight in such way that maximum altitude, speed, acceleration are estimated and plotted for analysis. Static stability was set to 2.29. This is a somewhat conservative margin but was kept by virtue of the lack of previous experience of the group in such activity. The fuselage body was made of a PVC pipe, 60 mm external diameter and 53 mm internal diameter. The nose cone was made of Nylon, machined, with a hollow to have the minimum possible weight. The fin sets were fabricated on 6101-T6 aluminum, with 1.0 mm of thickness. A pair of launch lugs was added to the rocket external fuselage. The final conception culminated with the following characteristics and dimensions:

- Rocket length → 1100 mm;
- Rocket diameter → 60 mm;
- Rocket material → PVC;
- Number of fins → 3;
- Fin material → aluminum;
- Span diameter → 200 mm
- Rocket mass → 5410 g;
- Rocket nose → conic.

Figure 7 shows a flight simulation for the rocket final conception. As it can be seen, dynamic stability was obtained. A launch guide angle of zero, straight up configuration, was chosen and no wind conditions. An altitude of about 2500 meters should be expected at apogee. No more than six G´s of acceleration and a maximum velocity of 250 m/s would be observed.

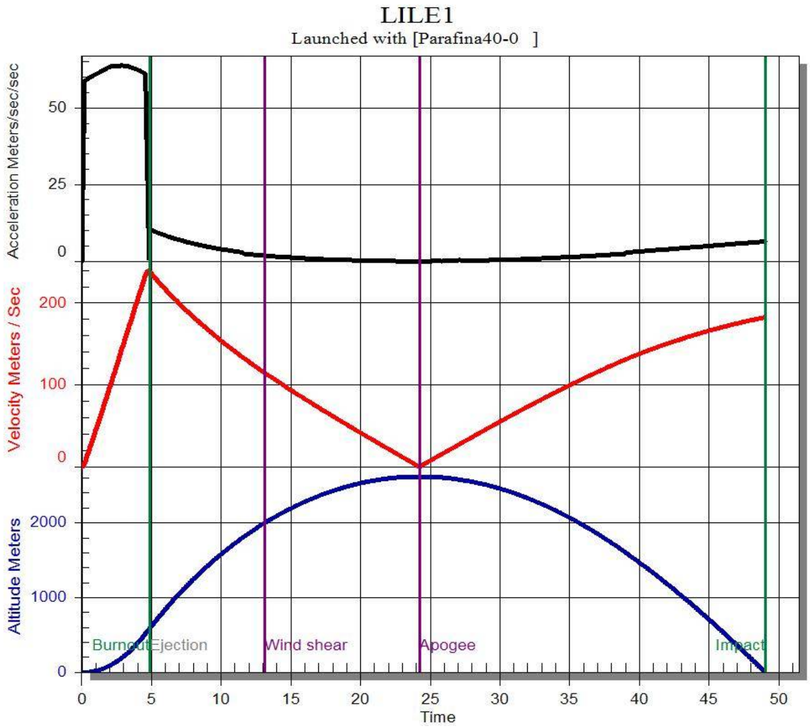

Figure 7. Rocket flight simulation (▬ acceleration; ▬ velocity and ▬ altitude).

For the oxidizer load in the rocket, a Nylon hose for high pressure is used, that is connected to the injector plate. The tank receives a small puncture in its superior extremity that serves of gaseous nitrous oxide purges of the tank and relief of the pressure in its interior. The ignition system is composed of black powder and a resistance as described previously. When it initiates the release of nitrous oxide the heat generated in the ignition is enough for its dissociation. The oxygen released in this breaking feeds the combustion of paraffin, thus generating the thrust of the engine.

The rocket motor setting sequence is described as follow: the nylon hose is connected to the injector plate; the ignition system is mounted in the injector plate; insertion of the injector plate in the interior of the motor body; insertion of the combustion pre-chamber and fuel grain; insertion of the nozzle; closing of the aluminum tube with a threaded cover.

5. The Flight Test

With the rocket subsystems assembled the group left for launching in a small farm in the Santo Antônio do Descoberto – GO. The ignition system was set just prior of the flight test.

After a check list, the rocket was placed in the launch pad and the launching was ready to take place.

The weather was cloudy, mildly windy. The launch angle was estimated in less than 5 degrees bending towards the wind direction (Figure 8). The group was protected by a small construction that worked as a bunker for safety precautions.

Launching started by opening, up side down, the nitrous oxide commercial tank (4 kg capacity) valve, followed by the filling valve. The rocket tank had a vent hole that blows off nitrous oxide when the tank is completely filled. The valve is closed and the ignition system is turned on. After ignition the rocket left the launch pad and could be observed in a stable and rapid flight for only two to three seconds, the observers could follow its ascending trajectory for a couple of seconds, and then they lost sight of it. Very little smoke emission occurred, as consequence of a near stoichiometric burning. The main products of paraffin and nitrous oxide combustion are CO_2, H_2O e N_2.

There was no recovery system or electronic device as to track altitude or other flight parameters. Video imagery was used for initial flight performance analysis.

The rocket crashed about 700 m, in the direction of the wind and the launch pad angle. The motor was damaged in the upper part, due to the impact. It was deformed by compression forces. It was concluded that rocket hit the ground in a ballistic trajectory, flying nose down. The audio, from de video camera, was analyzed indicating a strong signal for about 3.2 s. The signature of the sound was that of the operating motor. Despite the absence of other means form performance comparison, the launch was considered a great success. A week later another rocket (Lile2) was also launched. The amount of oxidizer was not enough to fill the rocket tank. The launch however took place and, due to much lower acceleration and altitude it was possible to follow the flight trajectory, upwards and descending. Again, the rocket presented a dynamically stable flight indicating a proper design.

The group is now designing a sounding rocket which planed altitude is above 20000 meters. The flight test is schedule to take place in Natal, Brazil (*Centro de Lançamento da Barreira do Inferno*) by September 2006.

Figure 8. LILE 1 and LILE 2 in the launch pad.

6. Conclusions

Two hybrid rockets were successfully designed and launched, demonstrating the feasibility hybrid technology. As far as we know, this was the first of its kind in Brazil. Paraffin-based hybrid rocket may be an important propellant for the Brazilian Space Program.

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