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DESIGN AND TESTING OF THE IGNITION SYSTEM FOR HYBRID ROCKET MOTOR

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ABSTRACT

The design of an efficient and reliable ignition system represents a challenge for propulsion engineers especially for hybrid rocket motors where the fuel grain and oxidizer are separated. In the current work, a simple design model is used to size the igniter, and to select the solid propellant type and mass. The analysis is based on the heat transfer model of the main rocket motor grain, thermochemical model of the ignition process and an empirical regression rate model of the grain. Following this analysis, a design point of the igniter including the chamber pressure, igniter dimensions and a propellant type are selected to perform the required task and to ensure the ignition of the main engine. Series of experiments are performed for different propellant types, different propellant mixture ratios and different grain size to select the optimum igniter. The flame length, temperature and ignition time are observed to assess the performance of the igniter. Finally, the selected igniter design is tested successively in the hybrid rocket motor.

KEYWORDS

Solid, propellants, adiabatic, flame, grain, heat transfer, chemical equilibrium, and igniter geometry

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NOMENCLATURE

a	Burning rate constant
A_t	Nozzle throat area of the igniter
A_b	Burning area of the propellant inside the igniter
C^*	Characteristic velocity of the exhaust gases from the igniter
D_p	Fuel grain port diameter
$\dot{m}_g$	Mass flow rate of the igniter nozzle
n	Regression rate exponent
P_{ig}	Igniter chamber pressure
$\dot{r}$	Fuel grain regression rate
T_g	Temperature of the exhaust gases from the igniter
ρ_f	Fuel grain density

1. INTRODUCTION

Release of the chemical energy in hybrid rocket motors is initiated by an ignition system. Selection of the ignition system depends on the nature and phase of the rocket propellants, the necessity to restart the engine, the system safety of the main engine, the compatibility of the igniter with the overall engine and weight and space considerations. All ignition methods have a common requirement of rapid and reliable engine ignition.

Although the igniter is a key element in any rocket motor, it appears that only a minimal effort is usually extended on igniter designs particularly relative to motor designs. Under ordinary conditions, igniter designs may not present a problem. However, when special requirements arise or when problems are encountered with standard igniter designs, igniters should be treated carefully.

The current work is motivated by the special restrictions imposed on igniters for hybrid rocket motors where the igniter geometry, propellant type, duration of ignition and igniter location in the motor are very effective for the rocket motor performance. The igniter should be placed carefully in the combustion chamber such that it will not affect the flow of the oxidizer from the feeding system. In addition, igniter designs must be simple and less expensive especially if it will be used in a simple system such as the hybrid rockets.

Different systems of the igniters have been proposed for rocket motors including pyrotechnic, hypergolic propellants, fuse wire, magnetic strip, catalytic bed and spark torch igniters [1-4]. In the current work, a solid-propellant igniter is proposed to heat up the fuel grain in the hybrid motor. The solid propellant igniter consists of a chamber and a convergent nozzle. The igniter geometry, type and amount of the propellant and the ignition time are determined based on the analysis of the heat transfer in fuel grain, the combustion process of the igniter propellant and steady flow of the hot gases from the igniter nozzle. Another approach is proposed by Kuna [5] to predict the performance of a pyrogen igniter for solid rocket motor using the form function technique.

Two different designs of the igniters for two different propellants are tested in a series of experiments. The flame length, the ignition time and the flame temperature are observed to assess the performance of the igniter.

2. HEAT TRANSFER MODEL

A heat transfer model for the main engine grain is used to compute the grain temperature distribution. The temperature distribution in the fuel grain depends mainly on the mass flow rate of the exhaust gases from the igniter and the ignition time. The ignition time and mass flow rate are computed such that the surface temperature of the grain is greater than the ignition temperature. The heat flux inside the grain is assumed to be negligible in the axial direction since the flame temperature and velocity of the igniter are assumed to be uniform in the axial direction. Therefore, the unsteady temperature distribution in the fuel grain is governed by the following equation:

$$\frac{\partial T}{\partial t} - \alpha_f \left(\frac{\partial^2 T}{\partial r^2} + \frac{1}{r} \frac{\partial T}{\partial r} \right) = 0 \tag{1}$$

where T is the fuel grain temperature, r is the radius of the grain.

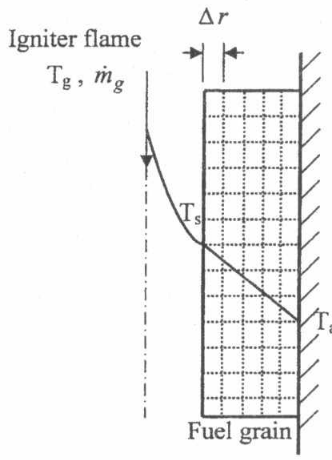


Fig. 1. Schematic of the hybrid fuel grain and the heat transfer process in the ignition.

The fuel grain thermal diffusivity α_f is given by

$$\alpha_f = \frac{K_f}{\rho_f C_{pf}} \tag{2}$$

where K_f, ρ_f and C_{pf} are the thermal conductivity, density and specific heat of the fuel grain.

Different boundary conditions are used to compute the temperature distribution in the grain as shown in Fig. 1. Since the ignition process is done in a very short time, the outer wall of the grain is assumed to be at the ambient temperature, while the inner wall temperature is computed using the energy equation.

$$h_g (T_g - T_s) = \rho_f C_{pf} \frac{\Delta r}{2} (\frac{\partial T}{\partial t})_1 - K_f (\frac{\partial T}{\partial r})_1 \tag{3}$$

where T_g is the temperature of the igniter exhaust gases, T_s is the surface temperature of the fuel grain at the inner radius, Δr is the grid size at the inner wall and the subscript 1 refers to the first grid point near the inner surface of the fuel grain.

The convective coefficient of heat transfer h_g is computed using the following empirical formula for rocket motors according to Barrere [6]

$$N_u = 0.0162 \Pr^{0.82} \mathrm{Re}^{0.82} (\frac{T_g}{T_s}) \tag{4}$$

where Nusselt number N_u , Prandtle number Pr and Reynolds number Re are defined as follows:

$$N_u = \frac{h_g D_p}{K_g} \tag{5}$$

$$\mathrm{Re} = \frac{4 \dot{m}_g}{\pi D_p \mu_f} \tag{6}$$

$$\Pr = \frac{\mu_g Cp_g}{K_g} \tag{7}$$

2.1 Numerical Discretization of the Heat Transfer Model

The heat transfer equations (1) and (3) are discretized using a second-order central difference formula for the spatial derivatives and a first-order forward Euler for the time derivatives. The discretized equations are written as follows:

$$T_{n+1,j} = T_{n,j}\{1-(2+\frac{\Delta r}{r_j})\lambda\} + \lambda\{(1+\frac{\Delta r}{r_j})T_{n,j+1} + T_{n,j-1}\} \quad (8)$$

$$T_{n+1,1} = \frac{2h_g \Delta t}{\rho_f C_{pf} \Delta r}(T_g - T_{n,1}) + 2\lambda(T_{n,2} - T_{n,1}) \quad (9)$$

where the parameter λ is given by;

$$\lambda = \frac{\alpha_f \, \Delta t}{\Delta r^2} \quad (10)$$

The condition of stability of equation (8) is given by;

$$1-(2+\frac{\Delta r}{r_j})\lambda \geq 0 \quad (11)$$

$$\text{or } 0 < \lambda \leq \frac{1}{2} \quad (12)$$

3. IGNITER MODEL

A regression rate model for the igniter propellant similar to the regression rate of the solid propellant combustion is used in the current study. The igniter grain in is regressing axially from the lower end to the upper end of the grain where the regression area will be constant. The regression rate of solid propellants depends on the chamber pressure as given by the following relation [7]:

$$\dot{r} = a\, p_{ig}{}^{n} \quad (13)$$

For the design purposes, the igniter flow is assumed to be in a steady state condition. Therefore, the nozzle mass flow rate is given by the following relation:

$$\dot{m}_g = \frac{p_{ig}\, A_t}{C^*} \quad (14)$$

where the characteristic velocity C^* is given by

$$C^* = \frac{\sqrt{\gamma_g\, R_g\, T_g}}{\Gamma} \quad \text{and} \quad \Gamma = \gamma(\frac{2}{\gamma+1})^{\frac{\gamma+1}{2(\gamma-1)}} \quad (15)$$

The burning area can be computed using the following relation:

$$A_b = \frac{\dot{m}_g}{\dot{r}\, \rho_p} \quad (16)$$

where ρ_p is the propellant density in the igniter and A_b is the burning area or the cross-sectional area of the igniter chamber.

The length and mass of the grain in the igniter is computed assuming a steady state chamber pressure in the igniter

4. THERMOCHEMICAL MODEL

A thermochemical model is required to compute the combustion temperature of the igniter propellant and to compute the properties of the exhaust gases. In the current work, a chemical equilibrium model developed by Gordon and McBride [8] is used to determine the properties of the exhaust gases. The equilibrium model assumes that the reaction time of the propellants is very small compared with the flow residence time. This assumption is adequate for the igniter since the propellants used in igniters usually have a very small reaction time.

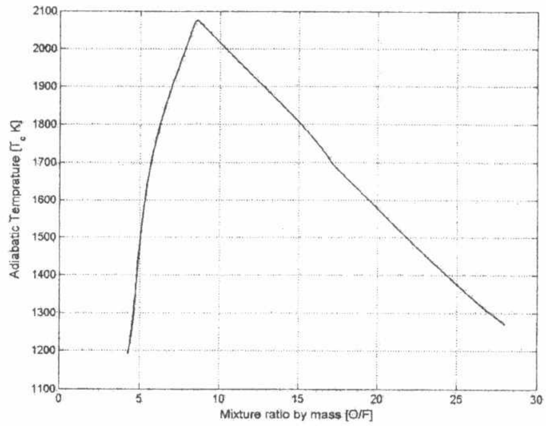


Fig. 2. The adiabatic flame temperature of potassium nitrate and sucrose versus the oxidizer to fuel ratio (combustion pressure=20 bar).

In this study, two different propellants are used to design the igniter. The first propellant used to design the igniter is a combination between potassium nitrate (KNO3) serving as the oxidizer and sucrose ($C_{12}H_{22}O_{11}$) serving as the fuel and binder. The properties of this propellant are computed using the chemical equilibrium method and the variation of the adiabatic flame temperature with the mixture ratio is presented in Fig. 2. The maximum temperature for this propellant is obtained at oxidizer to fuel ratio of 8.6.

The second propellant used to design the igniter is a mixture of Sulfur, Charcoal and Potassium Nitrate with the mass ratios of 10:15:75 respectively. This propellant is known as black powder. The charcoal serves as the fuel and potassium Nitrate serves as the oxidizing element while the sulfur is used to stabilize the reaction and produces the flame.

5. DESIGN PROCEDURES OF THE IGNITER

The design of the igniter depends on many parameters such as the propellant type, grain size, propellant mixture ratio, location of the igniter with respect to the grain, and the direction, shape and temperature of the ignition flame. The proposed igniter in this work is a solid propellant gas generator which consists of a combustion chamber and exhaust nozzle. The required igniter is used in a hybrid rocket motor. The fuel grain in this motor is the polyethylene and the oxidizer is the gaseous oxygen. The design process starts with the selection of the propellant type. The thermochemistry code is used to determine the optimum mixture ratio of the propellant and its properties such as the flame temperature, the specific heat, the thermal conductivity and the characteristic velocity. Using the thermochemical properties of the igniter exhaust gases, the heat transfer model is used to obtain the temperature distribution in fuel grain. A parametric study is done to determine the required igniter mass flow rate which heats up the fuel grain to a certain temperature in a certain time.

5.1 Objective

Design an igniter for a hybrid rocket motor. The hybrid rocket grain geometry is as follows:

Fuel grain port diameter : 2 cm
Fuel grain outer diameter : 5 cm
Fuel grain length : 45 cm
Fuel grain type : Polyethylene

5.2 Potassium Nitrate and Sucrose Igniter

5.2.1 Design Steps

1. Igniter propellant is potassium nitrate (KNO3) and sucrose ($C_{12}H_{22}O_{11}$).

2. The regression rate constants of eq. (13) for this propellant are n=0.319 and a=0.0665 as indicated by Humble [7].

3. The thermochemical properties of the propellant combustion is obtained from the thermochemistry code which assumes chemical equilibrium of the exhaust gases composition. An optimum mixture ratio of 8.6 for this propellant is obtained from Fig. 2 and the flame temperature will be 2050 K.

4. The heat transfer model is used to compute the required mass flow rate from the igniter which heats up the fuel grain in a certain time to a temperature of 700 K. The results are presented in Fig. 3 for different flame temperatures. It is clear from the figure that a mass flow rate of 15 gm/sec is required to heat up the fuel grain in 1 second if the flame temperature is 1000 K.

5. Using the steady flow equations (14-16), the igniter geometry for a selected chamber pressure is determined. Using equation (14), the variation of the igniter mass flow rate with the chamber pressure and throat diameter is presented in Fig. 4. For a chamber pressure of 20 bar, the throat diameter should be 3 mm to obtain a mass flow rate of 15 gm/sec. From Fig. 5., the grain burning diameter is 35 mm for a mass mass flow rate of 15 gm/sec. The grain length is computed such that grain is consumed in 1 second. The regression rate is obtained using equation (13) at a chamber pressure of 20 bar. Therefore, the grain length is estimated to be 15 mm to produce exhaust gases for more than 1 second as shown in Fig. 6.

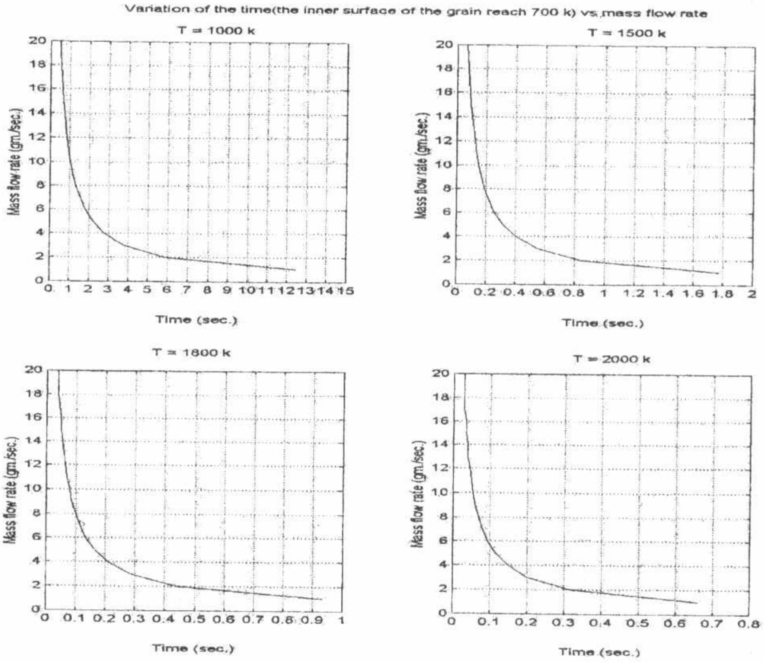


Fig. 3. Variation of the igniter mass flow rate versus time required to heat the fuel grain to 700 K for different combustion temperature of the potassium nitrate and sucrose igniter.

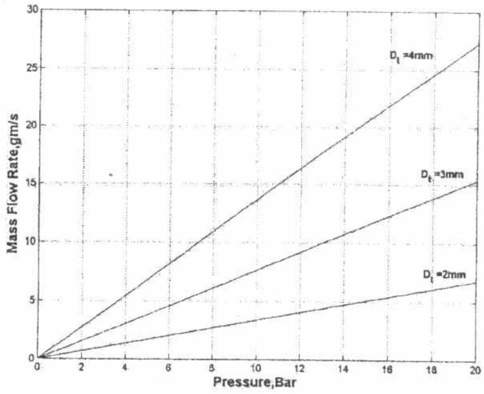


Fig. 4. Variation of the nozzle mass flow rate with the chamber pressure for different throat diameter.

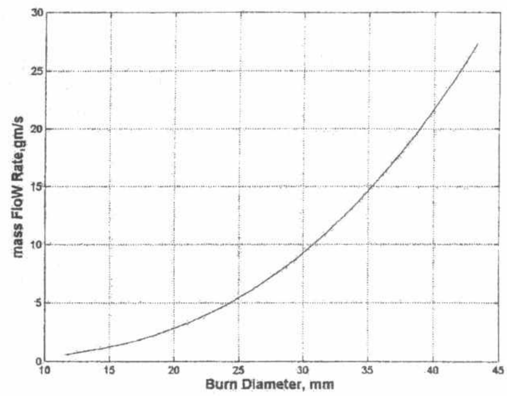
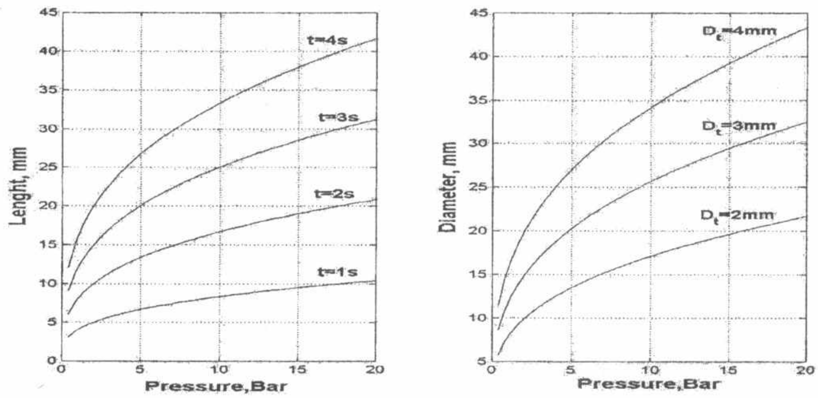


Fig. 5. Variation of the grain burning diameter with the mass flow rate for the specified propellant type and propellant mixture ratio.



a   b

Fig. 6. The effect of the chamber pressure on the igniter dimensions for different ignition durations and throat diameter a) Grain length versus chamber pressure b) Burning diameter versus chamber pressure.

5.2.2 Design Point and Igniter Geometry

The design point of the potassium nitrate and sucrose igniter can be summarized in the Table 1.

Table 1. Design point of the igniter.

Igniter chamber Pressure	20 Bar
Throat diameter	3 mm
Grain burning diameter	35 mm
Grain burning length	15 mm
Propellant mass	14 gm
Propellant mixture ratio	8.6
Igniter material	STEEL 40/70

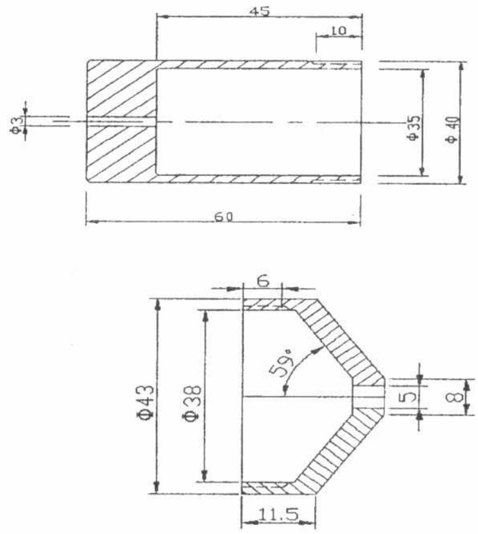


Fig. 7. Schematic of combustion chamber and nozzle of the potassium nitrate and sucrose igniter.

5.2.3 Igniter Testing

A series of experimental tests are performed to assess the performance of the potassium nitrate and sucrose igniter. The flame temperature is measured by a thermo-couple. The flame length and ignition time are observed. The ignition is started using an electric wire and Ni-Cr coil. Different propellant mixture ratios and propellant mass are used in different experiments. The snapshots of the igniter flame are shown in Figs. 8-9 for two different experiments. The experimental results for this igniter indicate that the flame temperature is less than 400 K and it may not be enough to start the rocket motor. The low temperature measured can be attributed to the moisture content in the propellant due to the mixing process and the nature of this propellant. Therefore, a different igniter design based on a different propellant is suggested.

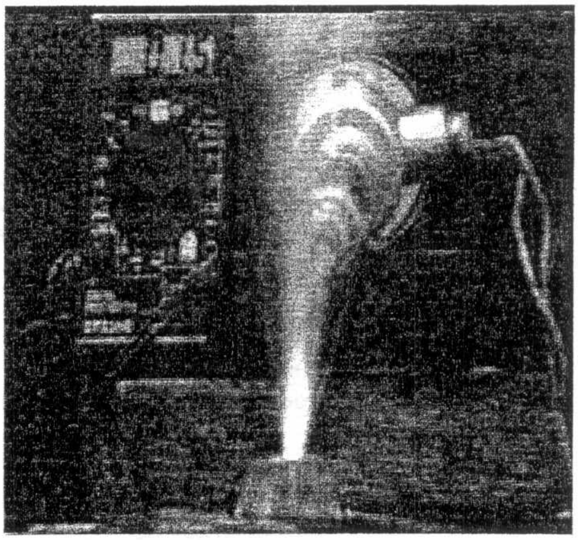


Fig. 8. Snapshot of the flame at 0.9 seconds (14.6 gm KNO3 and 5.4 gm sucrose)

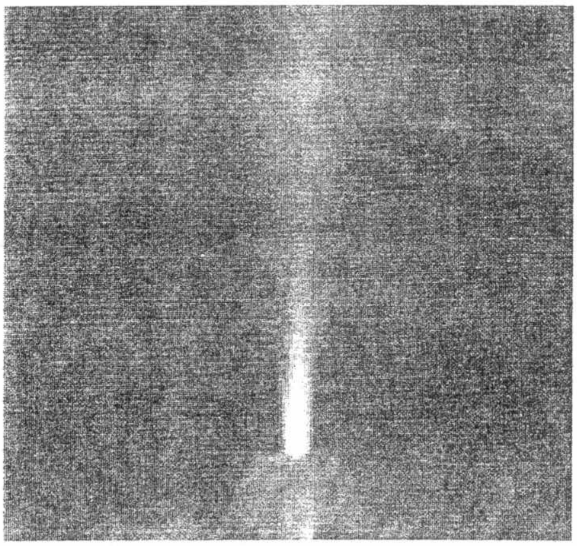


Fig. 9. Snapshot of the flame at 0.9 seconds (42 gm KNO3 and 22.6 gm sucrose)

5.3 Black Powder Igniter

5.3.1 Design Steps

The black powder igniter design steps are the same as those of the potassium nitrate and sucrose igniter except that a different propellant type is used. The black powder is commercially known as Gun powder. It is a pyrotechnic compound used in fire works and in manufacturing gun bullets. It has a very high flame temperature and very high regression rate. Black powder is composed of elemental Sulfur, Charcoal and Potassium Nitrate with the ratios are 10:15:75 by weight respectively. Charcoal is serving as the fuel and potassium nitrate is serving as the oxidizer. Sulfur is used to allow for a stable reaction and produces flame.

5.3.2 Design Point and Igniter Geometry

The design point of the black powder igniter is summarized in Table 2.

Table 2. Design point of the black powder igniter.

Igniter chamber Pressure	20 Bar
Throat diameter	2.5 mm
Grain burning diameter	20 mm
Grain burning length	60 mm
Propellant mass	9 gm
Igniter material	STEEL 40/70

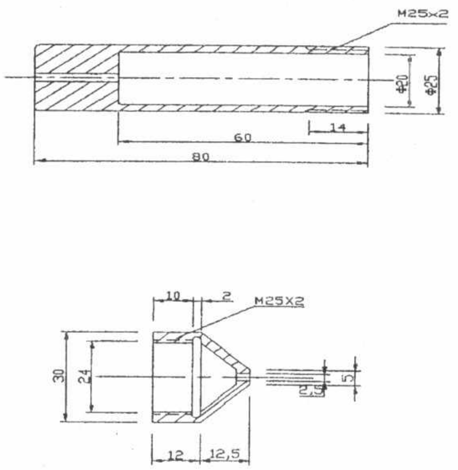
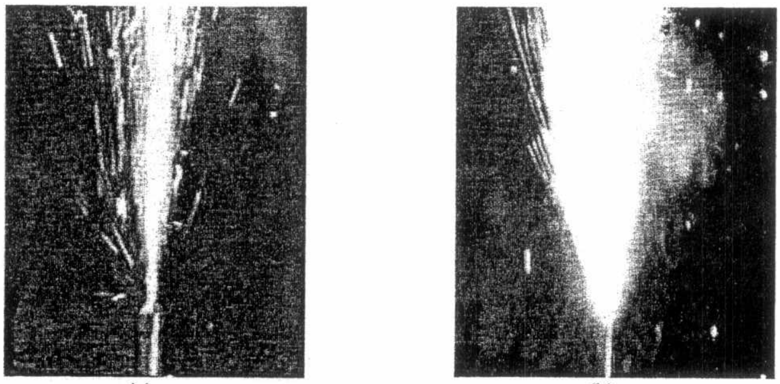


Fig. 10. Schematic of combustion chamber and nozzle of the black powder igniter.

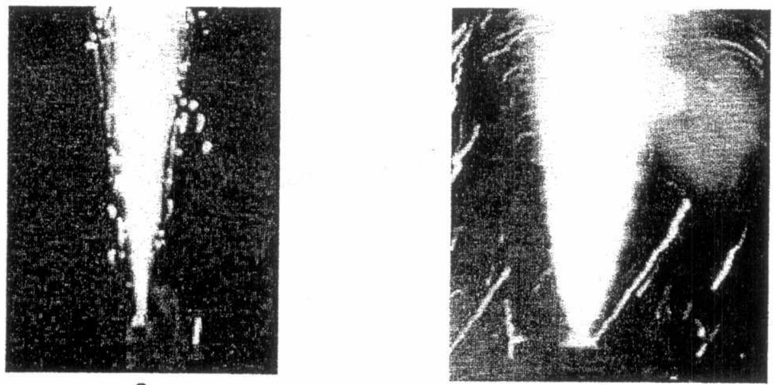
5.3.3 Igniter Testing

A series of experimental tests are also performed for the black powder igniter to determine the propellant mass for an optimum performance of the igniter. The ignition is started using an electric cupper wire and Tungsten coil. Different propellant mass are used in different experiments. The snapshots of the igniter flame are shown in Figs. 11-12 for two different experiments. A larger flame is noticed for this igniter and the thermocouple recorded higher flame temperature than the first igniter. The ignition continued for more than one second. So, the conclusion is that this igniter with a propellant mass of 9 gm will be suitable to initiate the hybrid motor combustion.



(a) (b)

Fig. 11. Snapshot of the flame for 3 gm black powder at a) 0.5 second b) 1 second.



a b

Fig. 12. Snapshot of the flame for 9.65 gm black powder at a) 0.5 second b) 1.5 second.

6. CONCLUSIONS

Theoretical and experimental methods are used to design an igniter for hybrid rocket motor. The theoretical method is based on the heat transfer analysis of the fuel grain, the thermochemistry of the ignition and the steady flow of the igniter. Two different propellants are used to design the igniters. A series of experimental tests are done to assess the igniter performance before installing it in the rocket motor. The experimental observations show that the black powder igniter produces the required flame length and temperature in the required ignition time. The potassium nitrate and sucrose igniter produced a very low flame temperature because of the moisture content in the propellant. Therefore, design of the black powder igniter is selected for the hybrid motor and this igniter is tested successively on the motor.

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