

DESIGN, MODELLING, FABRICATION AND TESTING OF HYBRID ROCKET ENGINE AND EVALUATION OF BURNING RATE FOR DIFFERENT SOLID FUELS

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ABSTRACT

A hybrid rocket is a rocket engine which uses propellants in two different states of matter- one is in solid and the other either gas or liquid. A hybrid rocket exhibit advantages over both liquid rockets and solid rockets especially in terms of simplicity, stop-start-restart capabilities, safety and cost. This paper deals the design and development of a hybrid rocket having paraffin wax as solid fuel and liquid oxygen as oxidizer. Due to variation of pressure in combustion chamber there is significantly change in mass flow rate, burning rate and uneven regression along the length of the grain.

Key Word: Hybrid rocket, Rocket Engine, Hybrid Propellant, Combustion Chamber.

1. INTRODUCTION

This project describes the working model of a hybrid propellant rocket motor. We have designed a hybrid rocket thrust chamber based on the predetermined combustion chamber pressure and the properties of hybrid propellant. This project is all ready in working condition with normal oxygen injector. Now we have planned to modify the injector design to improve the combustion property. We will use spray type injector for injecting the oxidizer. This idea will increase the performance followed by the regression rate of the solid fuel. By employing mass conservation law, oxygen mass flux, oxidizer/fuel ratio and regression rate the thrust coefficient can be obtained for our current design. CATIA V5 R20 is our design software for the complete setup. This project is fully based on experimental evaluation and the collection of combustion and flow parameters. The thrust chamber is made of stainless steel and the duration of test is around 15-20 seconds (Maximum). These experiments indicates that paraffin based fuel provides the

opportunity to satisfy a broad range of mission requirements for the next generation of the hybrid rocket system.

1.1 HYBRID ROCKETS

Rocket propulsion concepts in which one component of the propellant is stored in liquid phase while the other is stored in solid phase are called hybrid propulsion systems. Such systems most commonly employ a liquid oxidizer and solid fuel. Various combinations of solid fuels and liquid oxidizers as well as liquid fuels and solid oxidizers have been experimentally evaluated for use in hybrid rocket motors. Most common is the liquid oxidizer solid fuel concept. The oxidizer can be either a non cryogenic (storable) or a cryogenic liquid, depending on the application requirements.

In the hybrid motor concept, oxidizer is injected into a pre-combustion or vaporization chamber upstream of the primary fuel grain. The fuel grain contains numerous axial combustion ports that generate fuel vapor to react with the injected oxidizer. An aft mixing chamber is employed to ensure that all fuel and oxidizer are burned before exiting the nozzle.

1.2 ADVANTAGES OF HYBRIDS

1. Compared to solids:
 - Chemically simpler and tolerant to processing errors.
 - Reduced chemical explosion hazard.
 - Thrust termination and abort possibility.
 - Better I_{sp} performance.
 - Throttling/ restart capability.
 - Reduced environmental impact.
2. Compared to liquids:
 - Mechanically simpler.
 - Tolerant to fabrication errors.
 - Reduced fire hazard.
 - Less prone to hard starts.
 - Higher fuel density.
 - Easy inclusion of solid performance additives (Al, Be)
3. Cost wise:
 - Reduced development costs are expected.

- Reduced recurring costs are expected.

1.3 HYBRID ROCKET FUELS

- Polyethylene – high density polyethylene has a density of 960 kg/m^3 and a heat conductivity of $0.23 - 0.29 \text{ W m}^{-1} \text{ K}^{-1}$
- Polymethacrylate-(CH₂C=CHCO₂R)_n poly – methacrylate has a density 1683 kg/m^3 and a molecular weight of 114
- Polyvinyl chloride (PVC) has a density of 1380 kg/m^3 and a thermal conductivity of $0.16 \text{ W m}^{-1} \text{ K}^{-1}$
- Hydroxyl terminated poly butadiene – butadiene – HTPB has a density of 930 kg/m^3 and thermal conductivity of $0.217 \text{ W m}^{-1} \text{ K}^{-1}$

1.3.1 PARAFFIN WAX

High regression rate hybrid fuels such as paraffin have made it possible to design single-port hybrid rocket has replacements for more conventional solid and liquid rockets. Paraffin wax is colorless or white consisting of mixture or solid straight-chain hydrocarbons ranging in melting point from about 48°C to 66°C . Paraffin wax is obtained from petroleum by de-waxing light lubricating all stocks. It is insoluble in water, but soluble in ether, benzene and esters. It is an excellent electrical insulator, with electrical resistivity of 10^{13} to $10^{17} \text{ ohm-metre}$. This is better than nearly all materials except some plastics. It has a specific heat capacity of 2.14 to 2.9 J/gK , and heat of fusion of 220 J/g .

1.3.2 HYBRID ROCKET OXIDIZERS

- Nitrous oxide (N₂O) also known as di-nitrogen mono oxide laughing gas has a boiling point of -89.5°C at 1 atm and is normally maintained as a liquid at a pressure of 54 bar nitrous oxide has molecular weight of 44.0 and a density of 1222 kg/m^3 at 20°C . The theoretical and a temperature of nitrous oxide is 7.27 Mpa and 36.6°C
- Gaseous oxygen
- Hydrogen peroxide

- Liquid oxygen has also chemical name as Lox having boiling point -1836°C at 1 atm liquid oxygen has a molecular weight of 32.0 and a density of 1265kg/m^3 at a 20°C .
- Nitrogen peroxide has a boiling point of 21.2°C and 1 atm nitrogen tetroxide has a molecular weight of 46.01 and a density of 1903kg/m^3 at a 20°C
- Nitric acid (HNO_3)

In this project, we used liquid Oxygen as a liquid Oxidizer. Liquid oxygen has a pale blue color and it is strongly paramagnetic and can be suspended between poles of powerful horseshoe magnet. It has a density of 1.141g/cm^3 . It is cryogenic with a freezing point of 50.5K . Liquid Oxygen has an expansion ratio of 1:861 under 1 standard atmosphere (100kpa) & 20°C , because of this; it is used in some commercial and military aircrafts as transportable source of breathing oxygen. Liquid oxygen is also a very powerful oxidizing agent: organic materials will burn rapidly and energetically in liquid oxygen. Further, if soaked in liquid oxygen, some materials are just called briquettes; carbon black, etc can detonate unpredictably from sources of ignition such as flames sparks or impact from light glows.

2. EXPERIMENTAL SET UP

The total length of motor design is and is made of stainless steel to withstand a chamber temperature of 3000k and pressure 10bar

2.1 COMBUSTION CHAMBER

The combustion chamber in a hybrid rocket motor not only provides the location for propellant combustion but also contains the whole fuel grain. The length of the combustion chamber is determined by the fuel grain configuration. Also, the longer the combustion chamber, the more stable the combustion, since the propellant has the more opportunity for even mixing. The combustion chamber, motor casing, is a vital part in the safety and performance of the hybrid rocket motor. The length of the chamber is around 180mm with diameter of 60mm & thickness 20mm

2.2 FUEL GRAIN CONFIGURATION

Tubular grain is being used as its length remains constant during burning if the end surfaces are insulated and constant oxidizer to fuel ratio of 3.456 can be maintained. In order to improve the grain strength during the heating period, the fuel is fabricated by mixing actuated charcoal or

aluminum powder with the paraffin. A series of paraffin based hybrid rocket fuels were tested in laboratory-scale motor that was designed similar to the classical one with an aft mixing chamber. In general however, for high power or amateur rocketry, the single cylindrical port geometry configuration is probably the best choice for most hybrid propulsion applications

2.3 REGRESSION RATE

In a hybrid rocket motor, liquid oxidizer is fed into the combustion chamber from the oxidizer tank, where it is ignited by an ignition source such as a pyrotechnic igniter. The fuel is then ignited and burnt in the presence of the oxidizer, where it vaporizes, and burns along the length of the fuel grain. The rate at which the fuel burns, is called the regression rate, and is measured in meters per second (m/s). The combination of fuel burn rate and oxidizer flow rate is called the mass flux, is measured in ($\text{kg}/\text{mm}^2\text{s}$). The regression rate then, is dependent on the mass flux and the length along the fuel grain port.

2.4 INJECTOR

For hybrid rocket motor motors on the high power and amateur rocketry level, where single circular port geometry is most frequently used, direct injection of oxidizer is the best approach, hence the method of injection is direct injection into fuel grain port.

A hybrid motor differs fundamentally in terms of combustion behavior compared with solid and liquid rockets, in that the oxidizer to fuel ratio (O/F), varies along the length of the hybrid fuel grain, i.e., it has an axial dependency. In a liquid rocket, the injectors generally inject both the fuel and the oxidizer at one end of the combustion chamber thus there is no axial dependency. In a solid rocket motor, there is no injector head, and every particle is bound of fuel and oxidizer, thus ensuring the O/F remains pretty much constant.

2.5 IGNITER

Pyrotechnic igniter is electrically initiated slow burning pyrotechnic torches. Burned duration is in the range of 2 to 10s, depending on engine requirements. It uses solid explosives as the heat producing materials. They can be made sufficiently strong to withstand environmental condition. Their disadvantage is hardware weight is high

2.6 NOZZLE

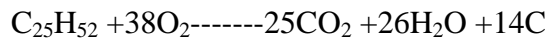
The rocket nozzle regarded as the most difficult portion of the design. It is also the most important, having to a 30% effect on the thrust capability of the engines. Prior to optimize of the nozzle, certain temperatures and pressures within the system are needed and hence were calculated for a C-D nozzle. The convergent length is taken as 20mm with an angle of 50° and divergent length is 60mm with an angle of 9°

2.7 FEED SYSTEM

The feed system comprises of a liquid oxygen tank that delivers the liquid oxidizer to the rocket chamber. There are valves to regulate the flow rate and even to keep check of the pressure. Other components include filters, pressure relief valves& non-return valves (NRV). NRV’s are used to stop any kind of back flow of liquid oxygen

3. INITIAL CALCULATION

Specific heat constant (γ) =1.3



Molecular weight= 28.33g/mol

Gas constant(R) = 293.46J/Kgk

3.1 CALCULATION OF THROAT DIAMETER

S.N	Outer diameter(d_0) (mm)	Port diameter (d_i) (mm)	Oxidizer /Fuel Ratio	m_{max} kg/s	Throat diameter(d^*) (mm)
1	70	20	3.456	0.396	7.93
2	45	20	3.456	0.009	3.74
3	68	20	3.456	0.138	4.67

3.2 VARYING AREA RATIO CALCULATION

$$\frac{A_2}{A^*} = \frac{1}{M_2} \left[\frac{2}{\gamma + 1} + \frac{\gamma - 1}{\gamma + 1} M_2^2 \right]^{\frac{\gamma + 1}{2(\gamma - 1)}}$$

$$\frac{A_2}{A^*} = 5,10,15,20,30,40$$

3.3 TABULATION FOR EXIT CONDITIONS

S.N O	Area Ratio(A_2/A^*) (mm^2)	A^* (mm^2)	D_2 (m m)	M_2	V_2 (m/s)	Thrus t(T) (N)	SF C
1	5	10	22.3	2.7	2146.5	113.4	180
2	10	10	31.6	3.2	2524.8	113.7	180
3	15	10	38.7	3.5	2741.0	103.5	165
4	20	10	44.7	3.7	2876.1	88.3	140
5	30	10	54.7	4.0	3107.8	55.24	88
6	40	10	63.2	4.2	3266.1	17.49	27

3.4 NOZZLE DESIGN

Exit Diameter = 31.29mm

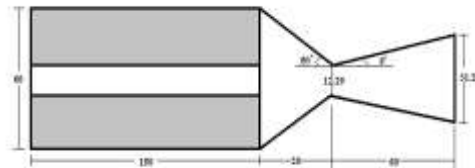
Throat diameter= 12.29mm

Area Ratio, (A_2/A^*)=6.48

Length of the divergent nozzle = 60mm

Angle of the divergent nozzle = 9 degree

Length of the convergent nozzle = 20mm



3.5 TABULATION OF NOZZLE LENGTH

S.NO	Area ratio(A_2/A^*) (mm^2)	Throat diameter(d^*) (mm)	Divergent angle (degree)	Length of divergent nozzle (mm)
1	4	10	5	57.15
2	4	10	10	28.35
4	10	10	10	61.35

4. CONCLUSION

- The maximum burning rate for pure wax is 1mm/s at 1 bar. But for charcoal and wax combination is less.
- For pure wax burning rate is high, as we added charcoal and sugar in it, the burning rate decreases. So, according to our result we got burning rate and mass flow rate for pure wax at 12bar pressure.

S.NO	Composition	Burning time (sec)	Burning rate (mm/s)	Mass flow rate (kg/s)
1	Paraffin wax	40	1	0.035

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