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Performance Evaluation of Hybrid Propellant Rocket Engine

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ABSTRACT
Hybrid Propulsion Technique is more suitable to applications requiring throattling and restart of long duration missions. Such application would include primary boost propulsion for vehicles of space launch, upper stages and satallite maneuvering systems. In hybrid techniques speed regulation is possible by regulating the supply of oxidizer. Higher fuel density can attained in case of hybrid propellant rocket system. In hybrid propellant Rocket system the fuel and oxidizer are stored separately, there is no chemical deterioration between fuel and oxidizer. Hybrid rockets are light weight when compared to liquid propellant and solid propellant rocket engines. Higher amount of theoretical specific impulse can be achieved compare to solid and liquid propellant engines. When more number of components available in engine construction, it is very difficult to provide a cooling to Engine thrust chamber. There is no universal well defined procedure or design method for making a design of hybrid propellant rocket engine. In order to rectify such a problems we should take almost care on propellant selection and propellant Incredients. In this paper we discuss in detail about evaluation of hybrid propellant rocket engine by theoretical approach. The design and operational performance of a rocket engine is purely depends on the combustion characteristics of a propellant, its burning rate, burning surface area, nozzle exit area ratio, grain geometry and thrust coefficient.

Keywords – Propellant burning rate, Grain geometry, Burning surface, Thrust coefficient, Nozzle exit area ratio.

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I. INTRODUCTION

Hybrid Propellant is a Combination of Solid fuel (Beryllium Hydride, Lithium Hydride, Hydrocarbon) and Liquid oxidizer (Fluorine, Chlorine Trifluoride, Nitrogen Tetroxide. The Hybrid rocket engine combine the advantages of both solid and liquid propellant rockets. In this type solid fuel along with liquid oxidizer is used as a propellant. Solid fuel is packed inside the combustion chamber and the liquid oxidizer is stored in the separate tank. Solid fuels are used because of their higher density than liquid fuels and liquid oxidizers are used because of solid oxidizers are problematic and lower performing than liquid. The main advantage of this rocket are mechanically simple and denser fuels. Hybrid rockets consists of pressure vessel contains liquid oxygen. The combustion chamber contains solid fuel and valve isolate the two and when thrust is desired a suitable ignition is introduced into combustion chamber. The liquid propellant flows into combustion chamber where it is vaporized and it reacts with the solid propellant occurs in a boundary layer diffusion flame adjacent to the surface of the solid fuel. Oxidizers are used to throttle the rockets and to restart the motor. Pressurizing the liquid oxidizer is not an important element of hybrid technology. The main advantage of hybrid propellant rocket engine is highly theoretical specific impulse of the rocket can be achieved compare to the compare to both the solid propellant and liquid propellant rocket engine. The operation of the hybrid propellant rocket engine is purely depends on the combustion characteristics of the propellant, its burning rate, burning surface area and nozzle exit area ratio.
The main objective of present work are:
a) To study the complete structural design of hybrid propellant rocket engine along with its components.
b) To study the complete operational phenomena of hybrid propellant rocket engine.
c) To find out the parameters, which are necessary to analyze a rocket with aid of theoretical approach.

1.1 Hybrid Propellants

Hybrid propellant rocket engines generally uses combination of solid fuel along with liquid oxidizer, which are the working substance of rocket engine constitute the fluid that undergoes chemical and thermodynamic changes. Many early hybrid rocket propellants were aimed at target missiles and low-cost tactical missile application purpose only. In recent years development focused on boosters prototypes for space launch applications. Selected propellants included a nitrogen Tetroxide/ nitrous oxide oxidizer and a hydrocarbon fuel grain composed of polymethylmethacrylate and magnesium. More practically although lower energy, upper stage hybrid propellant system is 90 to 95% hydrogen peroxide oxidizer combined with hydroxyl terminated polybutadiene (HTPB) fuel. In a hybrid, HTPB becomes the entire fuel constituent because of its low cost, processes easily and will not self-deflagrate under conditions. For large hybrid boosters application is liquid oxygen (LOX) oxidizer and HTPB fuel. Liquid oxygen is a commonly used oxidizer in the space launch industry, is relatively safe and delivers high performance at low cost. This hybrid propellant combination produces a nontoxic, relatively smoke-free exhaust. Where a smoky exhaust is not a detriment, hybrid propellants for certain applications may benefit from the addition of powdered aluminum to the fuel. This increases the combustion temperature, reduces the stoichiometric mixture ratio, and increases the fuel density as well as overall density-specific impulse. Some fuel-oxidizer combinations for hybrid propellant rockets are Beryllium hydride – Fluorine, Lithium hydride – Chlorine trifluoride, Lithium hydride – Nitrogen Tetroxide, Hydrocarbon – Nitrogen Tetroxide.

1.2 Structural Design

The structural analysis begins when all the loads can be identified and quantified. The kind of loads and timing of these loads during a life of a hybrid propellant rocket engine have to be analyzed for each application and every rocket because of hybrid propellant rocket engine has different phases (solid and liquid phase). Ignition and acceleration causes high stresses and strains. The maximum stress and strains can be accepted by the propellant under various conditions. The failure criteria are derived from cumulative damage tests, classical failure theories, actual rocket failures and fracture mechanics.

1.3 COMBUSTION

combustion chamber is the important component in hybrid propellant rocket engine because of 90% of the operation can be done in combustion chamber itself. The combustion process of hybrid propellant depends on what type of solid fuel used. The combustion taking place after the ignition process, where the gasified solid fuel and injected liquid oxidizer mixed and reach as per stoichiometric ratio. The flame is located within the dynamic boundary layer. The combustion is sustained by the thermal energy by convection and radiation from the diffusive flame to the burning surface of the solid grain. The combustion products are expanded in the nozzle for thrust generation. The temperature grows from the wall temperature to the free stream temperature and has a maximum value of the flame.

The summary of the hybrid propellant rocket engine combustion process are as follows:
a) Thermal heating and pyrolysis of solid fuel.
b) Diffusion of fuel spices toward the flame zone.
c) Formation of boundary layer near surface regions of the solid fuel grain.
d) Fuel surface regression due to continue heating from the turbulent diffusion flame.
e) Increase of axial mass flux along the port of the fuel grain due to mass addition.
f) Reduction of axial mass flux as port area increases in the later stage.
g) Potential ejection of unburned sliver residues.

One of the physical phenomena that limits the burning rate in hybrid rocket is “blocking effect” that caused by the high velocity injection.

1.4 Methods For Regression Rate Increase

1) Use of Energetic additives: A chemical approach used to increase the regression rate consists of adding high energy fuel ingredients to the solid fuel grain.
2) Turbulence Generators: A variety of devices or configurations can be used to generate a high level of turbulence at the regression surface, that increasing the heat exchange.
3) Droplet Entrainment: This technique is based on the use of fuels forming a thin liquid layer on their regression surface, allowing liquid droplets to be entrained by the gaseous oxidizer flow.

1.5 Combustion Instability

The hybrid combustion process tends to produce somewhat rougher pressure versus time characteristics than either liquid or solid rocket engines. However a well designed hybrid will typically limit combustion roughness to approximately 2 to 3% mean chamber pressure. Hybrid motors have been exhibited two basic type instabilities in static test environments. Oxidizer feed system induced instability (non-acoustic) and flame holding instability (acoustic). Oxidizer feed system instability is essentially a chugging type and arises when the feed system is sufficiently “soft”. Flame holding instabilities in hybrids are typically manifested at acoustic frequencies and appear in longitudinal modes. No acoustic instabilities in hybrid motors have been observed in higher frequency tangential or radial modes such as in solid or liquid engines. Flame holding instabilities arise due to inadequate flame stabilization in the boundary layer and are not associated with feed system perturbations.

1.6 Nozzle

Nozzle is used to increase the velocity and decrease the pressure of the gases. The nozzle provides for the expansion and acceleration of the hot gases and has to withstand the severe environment of high heat transfer and erosion. The following type of nozzles are widely used in rocket engine.

a) Fixed Nozzles: fixed nozzles are generally not submerged and do not provide thrust vector control.

b) Movable Nozzles: movable nozzles can provide pitch and yaw control and two are needed for roll control. Movable nozzles are typically submerged and use a flexible sealed joint or bearing with two actuators.

c) Submerged Nozzles: A significant portion of the nozzle structure is submerged within combustion chamber or case.

d) Extendible Nozzle: The extended nozzle improves specific impulses by doubling or tripling the initial expansion ratio thereby significantly increasing the nozzle thrust coefficient.

II. INDENTATIONS AND EQUATIONS

The following formulas were used to find out the parameters, which are necessary to investigate the entire process of hybrid propellant rocket engine.

The mixture ratio r is defined as the ratio of the oxidizer mass flow rate m_o and the fuel mass flow rate m_f or

$r = m_o / m_f$ (1)

The thrust produced is given by

$F = m_p{*}Cj + (P_e – P_a) A_e$ (2)

If q is the heat supplied in the form of chemical energy per unit mass of propellant, we get

$Q = Cp(T02 - T01)$ (3)

Specific impulse of a rocket engine is given by

$Isp = Cj / g$ (4)

Specific Propellant Consumption is given by

$SPC = WP / F$ (5)

Weight floe coefficient is given by

$Cw = Wp / Po A{*}$ (6)

Thrust coefficient is calculated through

$CF = F / Po A{*}$ (7)

Characteristic velocity of a rocket is given by $C{*} = CJ / CF$ (8)

when steady state is reached total mass flow rate will be $m = m_o + m_f = P_1A_{t / C}{}^{*}$ (9)

The thrust of a hybrid rocket motor will be

$F = (m_o + m_f) I_s g_o$ (10)

The mass production rate of fuel is given by

$m_f = density * A_b r$ (11)

Where A_b combustion port surface area.

Changing of thrust or throattling of a hybrid is achieved by changing oxidizer flow rate, usually by means of a throattling valve in the oxidizer feed line. The fuel flow is the function of the oxidizer flow but not necessarily a linear function. For circular port geometrics with radius R,

$r = a (m_o / 3.14 R^2)^n$ (12)

Where m_o is the oxidizer mass flow rate.
R is the radius.

III. FIGURES AND TABLES

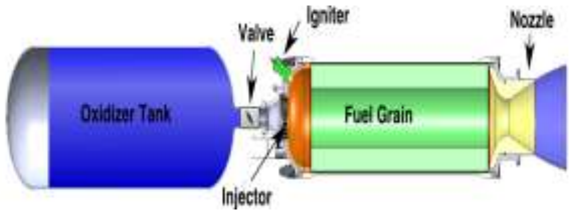

Figure 1. Schematic of Hybrid Propellant Rocket Engine.

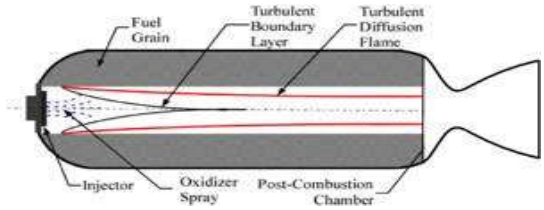

Figure 2. Types of Combustion Instability of Hybrid Propellant Rocket Engine.

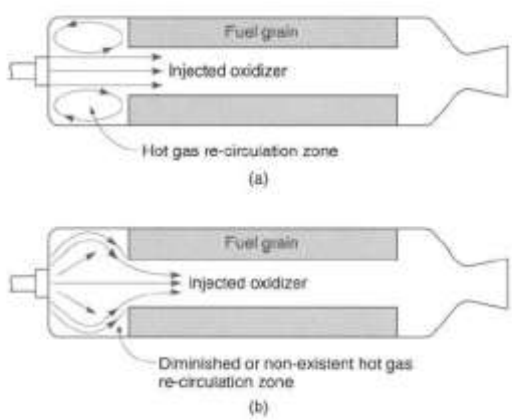

Figure 3. Axial and Conical Injection of Oxidizer of Hybrid Propellant Rocket Engine.

To design a hybrid propellant rocket engine the following procedure can be carried out with help of following inputs as a example.

S.No	Parameter	Quantity
1	Average Molecular Mass	24kg/kg-mol
2	Chamber Pressure	$2.53{*}10^6 N/m^2$
3	Atmospheric Pressure	$0.09{*}10^6 N/m^2$
4	Chamber Temperature	2900K
5	Throat Area	$0.00050m^2$
6	Specific Heat Ratio	1.3

Table 1. Inputs for Calculation

From the above inputs, after making a theoretical calculation by using above standard formulas, We have the following results.

IV. CONCLUSION

From the above theoretical work we can concluded the following aspects as a result of a present work.

1) Higher value of theoretical specific impulse can be achieved in the case of hybrid propellant rocket engine compare to both the solid propellant rocket engine and liquid propellant rocket engine.

2) When hybrid propellant rocket engine is in motion condition, its combustion process is controllable, which means speed regulation is possible by regulating the supply of liquid oxidizer.

3) High load capacity can be evaluated through the hybrid propellant rocket engine and hybrid rockets are light weight compare to liquid propellant rocket engine because of less components available with the rocket system. So vibration problems are also less compare to above cases.

4) In hybrid propellant rocket engine, mixture ratio and hence specific impulse will vary somewhat during steady state operation and throattling. Lower density specific impulse than solid propellant system.

5) Some of the Fuel sliver may be retained in the combustion chamber at the end of burning which slightly reduces the motor mass fraction.

S.No	Parameter	Quantity
1	Characteristic Velocity	1502.07 m/s
2	Mass flow rate of the Propellant	0.841 kg/s
3	Exit Mach Number	2.78
4	Exit Pressure	$91*10^3$ N/m^2
5	Exit Velocity	2159.86 m/s
6	Exit Area	$2.0125*10^2$
7	Thrust	1818.45 N
8	Specific Impulse	220 s
9	Throat Mach Number	2.029

Table 2. Final Results

6) The above results were find out after done a theoretical analysis of a hybrid propellant rocket engine. From the result of a present work, The specific Impulse of a hybrid propellant rocket engine is high compare to both the solid and liquid propellant rocket engine. Considerable amount of thrust can be achieved for very short value of throat area.

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