Preliminary Rocket Exhaust Plume Analysis and Compatible Acoustic Ignition Technology

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Abstract— Liquid rocket engines experience various difficulties, including ignition for engine restart in extreme low temperature and vacuum conditions, as well as acoustic noise and combustion instabilities, which can even couple with the fuel inflow and magnify themselves. The report aims to discuss the rocket plume analysis and its subsequent application for the design of an Acoustic Ignition setup. An acoustic engine ignition setup uses a resonance tube to not only ignite the propellant, but also to mitigate combustion instability. By placing the resonance tubes at appropriate locations within the combustion chamber, combustion instabilities can be avoided and acoustic noise can be reduced. Using the resonance tube for ignition, also allows for more efficient use of the fuel by enabling ignition to occur within milliseconds. This is advantageous especially for engines that require frequent restart at extremely low temperatures, such as Attitude Control Systems (ACS) in space. A typical ACS is turned on and off frequently in the extreme conditions of space while in service. Thus, current attitude control systems avoid using liquid bi-propellant engines, and instead use the cold gas thruster or the hypergolic engine with lower performance. By eliminating the slow ignition problem of the liquid propellant engine, this allows it to be used in the ACS efficiently.

Keywords— Acoustic Ignition, Attitude Control system, Bi-Propellant engines, Resonance tube.

I. INTRODUCTION

PROPULSION of aircraft and spacecraft through the combustion of fuels has both an extensive history and a wide range of applications. For reasons of production cost, low maintenance, simplicity, scalability and safety, advanced technologies based on corrective rocket exhaust analysis has derived premium importance, as in ballistic missile defence systems, acoustic ignition technology and similar applications where the lack of throttle control or reigniting capabilities are outweighed by cost savings or safety advantages over liquid propellant rocket engines.

II. ROCKET EXIT PLUME FLOW PROPERTIES

The first step in the analysis is to determine the properties of the rocket engine exhaust at the nozzle exit. The essential data to carry out this calculation are the chemical composition of the rocket propellant including the amount of alkali metal contaminants, the rocket chamber pressure, and the detailed geometrical configuration of the engine chamber, nozzle throat, and nozzle contour.

The characterization of the rocket exhaust flow properties begins with a calculation of the rocket engine chamber flow properties and composition by assuming chemical equilibrium to prevail at the specified chamber pressure at a constant enthalpy, because the flow velocity is low and the chamber temperature is high, the combustion time is very short in comparison with the time required for the combustion products to flow through the entire chamber. Under this condition, chemical equilibrium is attained. [1]

As the combustion products flow into the entrance region of the nozzle throat, their velocities increase rapidly and deviation from chemical equilibrium starts. Therefore, the calculation must consider finite chemical reaction rates as the engine exhaust flows and is expanded in the rocket nozzle. Since a large amount of particulates are present in the combustion products of solid propellants, it is often necessary to treat two-phase flow in the throat and nozzle calculations.

The gas velocity is subsonic in the chamber and becomes supersonic downstream of the nozzle throat. The method of computing the transonic flow as the two-phase rocket exhaust passes from the chamber to the nozzle. As the exhaust passes through the throat, an uncoupled particle-gas flow with constant fractional thermal and velocity lag was assumed. [2]

A. Plume Flow Field

The next step in the analysis is to take the rocket engine exhaust flow properties at the nozzle exit plane outside the nozzle, as part of build up to considering advanced design consideration such as Acoustic ignition. The approximations in analysis are required to assume that the gas has been sufficiently expanded outside the nozzle to approach a source-like flow, and the detailed mechanisms of the gas expansion in the vicinity of the nozzle are of minor importance. The interaction between the plume gas and the ambient atmosphere becomes important in affecting the plume plasma properties.
The interaction generates the bow shock and the barrel shock in the plume. Its effect on the plume shape was considered. The method of characteristics (MOC) could be employed to analyse the inviscid flow field of rocket plume and plasma properties. This also has the capability of treating the barrel shock formation while assuming modified Newtonian pressure at the plume boundary.

More ionization in addition to that occurring in the nozzle is observed when the region just outside the nozzle near the nozzle lip the expansion is more rapid and can cause the chemical composition to freeze momentarily. Thus downstream of the exit plane, radiations of the chemical composition exist. This aspect of the additional ionization has not been fully investigated and would require additional analysis. At present, we assume the electron mole fraction in the whole plume to be at the nozzle exit value. The electron density anywhere in the plume can then be computed from the gas density distribution. Together with the plume temperature determination, it enables the collision frequency to be calculated. These are the plasma properties required for determining the interference of the electromagnetic waves by the rocket plume.

B. Effects of Ionic Interaction Rate

In order to aid effective investigation of rocket plume, finite rate chemical reactions must be included in the rocket nozzle flow calculation in order to determine the electron mole fraction at the nozzle exit. Both neutral and ionic reactions need to be considered. The neutral reactions can affect the flow field temperature and the concentrations of the neutral species that enter into ionic reactions. Both have effects on the ionic reaction rate and, therefore, the electron concentration. For a reaction in which the reactants A and B are converted into the products C and D

\[ A + B \rightarrow C + D \]

The specification of the forward reaction rate coefficient \( K_f \) is sufficient, since the reverse rate coefficient \( K_r \) can be computed from the equilibrium constant \( K_r \).

Thus,

\[ K_r = \frac{K_f}{K} \]

Where \( K_r \) varies with the gas temperature \( T \) as follows:

\[ K_f = AT^n \exp(-E/RT) \] (2)

Where, \( R \) is the gas constant and the constants \( n \) and \( E \) are determined from experimental reaction rate data. Liquid propellants generally do not form condensed solid particles except perhaps from erosion of the nozzle throat. In that case, the amount of particulates carried by the exhaust gas is so small that effects on the flow field can be neglected. Thus it simplifies the plume analysis to utilise the exhaust for acoustic ignition as discussed in the following section.

III. ACOUSTIC IGNITION

Rocket engines require an ignition source, or igniter, to initiate combustion. A variety of igniters are available for this use, such as hypergolic, pyrophoric, spark torch and catalytic. Most of these are viewed as high cost and operationally complex. Acoustic igniters offer an alternative. The acoustic igniter relies on the interaction between a supersonic jet flow issuing from a nozzle and a resonance tube placed closely downstream to produce a dramatic increase in temperature in the resonance tube. The igniter also includes an enclosure surrounding the gap between the nozzle and resonator. This system eliminates the need for electrical energy to drive spark systems to initiate combustion in liquid-propellant rockets, therefore reducing complexity, cost and weight. While the acoustic igniter concept has been in existence for years, only recently have commercial attempts been made to design and manufacture them for use in space propulsion. Most research in acoustic igniters has been experimental. In this project we are exploring whether computational techniques can be used to predict the behavior in an acoustic igniter.

The igniter behaves in either the regurgitant or screech mode depending on where the shock cell is interrupted due to the existence of the resonance tube. If the first shock cell is interrupted in the stable portion of the shock cell then the regurgitant mode is achieved. In this mode, temperature oscillations in the resonance tube occur on a time scale related to the axial extent of the tube. If the first shock cell is interrupted in its unstable region, the screech mode occurs. In this mode, intense heating at the end of the resonance tube is achieved and the oscillations in the tube are at very high frequency corresponding to the shock oscillation. Figure 2 provides a simple schematic of the shock wave phenomenon in the acoustic resonator.

A method of ignition using an acoustic igniter for igniting a mixture of propellants in a combustion chamber of a liquid propellant rocket engine.

An inert gas under pressure is injected via a propellant injection nozzle having an outlet opening out into a precombustion chamber, so that a portion of the jet of expanded inert gas at the outlet from the propellant injection nozzle penetrates into an opening of an acoustic resonator defining a cavity opening out into the precombustion chamber, opposite the propellant injection nozzle. And another portion
of said jet penetrates a housing around said acoustic resonator, whereby said portion of the jet of expanded inert gas having penetrated into the acoustic resonator is heated by oscillations due to shock waves until it reaches a temperature suitable for igniting said mixture of propellants that is to be injected during a second step.

The second step which occurs when said temperature has been reached and which consists in injecting an oxidizer through the propellant injection nozzle and a fuel through a fuel injector disposed inside said propellant injection nozzle on the axis thereof, whereby said oxidizer and fuel cooperate with the inert gas to form a mixture of propellants which penetrates into the acoustic resonator to catch fire on contact with the hot inert gas and create a flame that then ignites the mixture of propellants throughout the precombustion chamber[6].

Different possible configurations may be opted depending upon the requirement, such as use of the inert gas helium or nitrogen. Alternatively the oxidizer and the fuel may be injected simultaneously respectively through the propellant injection nozzle and the fuel injector or the oxidizer and the fuel may be injected in alternation respectively through the propellant injection nozzle and through the fuel injector as part of better optimization.

A. Preliminary Design Considerations and Limitations:

Acoustic ignition involves minor alteration to the prevailing thrust chamber design as shown in Figure 2. Fuel may be considered to be fed into cylindrical precombustion chamber courtesy an axially set injection nozzle. A hollow acoustic resonator is placed across the second main face of the precombustion chamber facing the injection nozzle. It is compatible for the acoustic resonator to include a frustoconical first portion that converges from the inlet opening thereof, and a cylindrical second portion that defines a cavity which terminates in an end wall of the cylindrical portion. Burning mixture after ignition may be fed towards a downstream chamber which communicates with a main combustion chamber of rocket engine, courtesy orifices placed at suitable locations.

To facilitate acoustic ignition, a gaseous component, an oxidizer, is required to be injected under pressure into an injection nozzle where it would be subjected to acceleration. Simultaneously, another component, fuel, would be injected. Hence the injections would serve to form a propellant mixture in the precombustion chamber. A portion of the mixture may penetrate into the cavity of the acoustic resonator and prevail therein. The resulting shock waves heat up the portion of the mixture situated in the acoustic resonator[7]. The mixture inflicts fire on reaching its ignition temperature. The flame leaves the cavity of the acoustic resonator and ignites all of the mixture in the precombustion chamber, after which it is to be let out via an outlet orifice towards the another downstream chamber which is set to suitably with main combustion chamber of the rocket engine through an orifice.

In particular, in the case of acoustic ignition, a flow of cold gas is circulated through an orifice and through a downstream chamber prior to ignition, where cold gas is in contact with the outer wall of the acoustic resonator. This contributes to reducing the development of heat inside the acoustic resonator and thus lengthens the time required for ignition.

The operation of that acoustic igniter is improved by the fact that the fuel gas injector opens out into the converging portion of the injection nozzle, upstream from the outlet of an injection nozzle, thus obtaining a mixture that is more uniform and increasing the stability of the ignition process. Nevertheless, the acoustic igniter is still not sufficiently reliable, and in particular it does not enable the maximum length of time required for ignition to be reduced.

B. Objective and Brief Description Of Technology

The present concept seeks to remedy the above-mentioned drawbacks, and in particular to provide an acoustic ignition presenting oscillations of greater intensity due to shock waves, smaller thermal losses that are from the acoustic resonator, and in general, operation that is optimized so as to be more stable and more reliable than existing devices, thus contributing in particular to reducing the time required for ignition.

These objects are achieved by processing a design of the acoustic igniter for igniting a mixture of propellants in the combustion chamber of a liquid propellant rocket engine, the igniter comprising a cylindrical precombustion chamber having a cylindrical wall and first and second end walls, a propellant injector nozzle opening out into the precombustion chamber through the first end wall via an orifice of minimum diameter, a fuel injector disposed inside nozzle placed axially, at least one outlet orifice of minimum diameter, formed through the cylindrical wall, and an acoustic resonator defining a cavity opening out into the precombustion chamber opposite the nozzle through the second end wall via an opening of a valve with as particular diameter, the igniter being characterized in that the acoustic resonator is surrounded by a housing which defines an auxiliary chamber around the acoustic resonator, the auxiliary chamber being closed with the inside thereof being in communication only with the precombustion chamber via at least one duct[8].
According to a preferred characteristic, the acoustic resonator presents, running from its opening: an essentially frustoconical converging portion extended by a cylindrical portion and closed by an end wall that is essentially parallel to the second end wall of the precombustion chamber, and the wall of the cylindrical portion of the acoustic resonator is made of a metal material having relatively less thermal conductivity, that is less than 25 W/m°C.

To add to the conduciveness, the converging portion of the acoustic resonator is proposed to have a convergence angle lying in the range 10° to 24°. and the cylindrical portion diameter of the acoustic resonator lying in the range 0.15 to 0.35 times the diameter resonator’s opening, and a length lying in the range one to three times the diameter of said opening.

In an optimized embodiment, the precombustion chamber is required to have a diameter greater than 2.2 times to that of nozzle to aid for enhanced combustion. Preferably, the downstream end of the fuel injector is situated in the converging portion of the nozzle.

C. Detailed Description of Particular Embodiments

The shock waves created within the hollow acoustic resonator are usually all the more intense in that, running from its opening, an essentially frustoconical converging portion extended by a cylindrical portion of an optimum inside diameter and closed by an end wall which is essentially parallel to the end walls and of the chamber, would serve the purpose of ignition by acoustics [9].

Ameliorating on the convention, operation of the acoustic resonator can be improved by surrounding a housing to define a closed auxiliary chamber around the resonator, with the inside thereof being in communication only with the precombustion chamber, and only via one or more ducts of small section. An empty space is preferred to prevail in the auxiliary chamber between the wall thereof and the outer wall of the resonator. The housing prevents heat generated within the resonator being dissipated to the outside environment. Ducts may be used to enable acoustic oscillations to penetrate into the inside of the housing, thereby providing additional heating of the gas that has penetrated into the empty space between the wall of the housing and the wall of the resonator. This helps reduce the time required for ignition.

D. Advantages of Acoustic Ignition:

1) No moving parts.
2) High pressure ignition is achieved within milliseconds, even in the cold vacuum environment of space.
3) Combustion instabilities are eliminated, and acoustic vibration is reduced by proper location and sizing of the resonance tubes within the combustion chamber.
4) Ability to perform over the life of the engine.

IV. CONCLUSION

Advanced propulsion is becoming a necessity, for both economic reasons and mission requirements. Advances in propulsion systems such as acoustic ignition and employment of rocket exhaust plume, mentioned above will ultimately reduce the cost of launching payloads into orbit. But the technology also requires in-depth analysis as discussed in the paper. The advanced systems as such will also reduce the propulsion system mass for satellite orbit maintenance and attitude control will be easier to maintain for extended periods of time, and reduce the cost of Low Earth orbit (LEO) to geosynchronous Earth orbit (GEO) orbit transfers. Advanced propulsion will extend our ability to explore the solar system and ultimately, enable interstellar missions.

REFERENCES