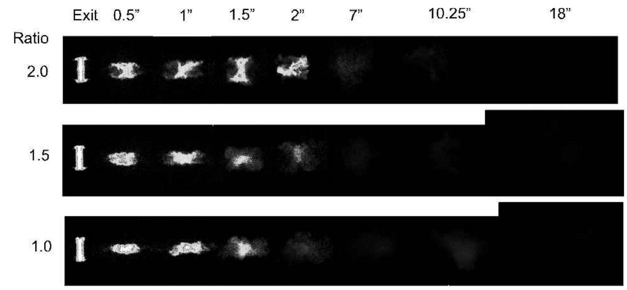

Fig. 4 Normalized images of the mixing.

Instruments (WinView 1.6.2) used to operate the camera was also used for further processing of the images. The background reflections were removed by subtraction, then the images were normalized to the total fluorescence signal. All the negative values in the image, caused by small differences between the background image and the background of the fluorescence image, were first set to 0 to further reduce the influence of the background. The total signal was then determined by adding the values of all the pixels with a value of over 10% of the maximum value in the image. The threshold was set at 10% to minimize the contribution of the background to the total.

Results

The resulting images can be seen in Fig. 4. These images have been resized to the same scale to enable comparisons of the turbine flow distribution. The pressure ratio ($p_{\text{turbine}}/p_{\text{rocket}}$) for each test is shown on the left and the image locations are measured from the exit plane of the strut.

In all cases, the conditions at the exit plane are quite similar. The turbine flow is clearly expanding into the rocket nozzles, forming a shape best described by the letter "I." At a pressure ratio of 1.0, the turbine nozzle flow is compressed vertically and experiences a strong lateral expansion. The flow appears almost completely dispersed at a distance of 5.1 cm. As the pressure ratio increased, the vertical expansion of the turbine flow (in the plane of the nozzle wall divergence) appears enhanced. The greater the pressure ratio, the greater the lateral expansion at the top and bottom of the "I," where the simulated rocket and turbine flows intersect. This lateral expansion of the turbine flow appears better defined, in general, for the pressure ratio 2.0 case and remains more or less intact to a distance of at least 5.1 cm downstream. The pressure differential would appear to be initiating a vortex flow at the point of intersection with the rocket nozzle flow. In all cases, the turbine flow is virtually dissipated by the time it reaches the 17.8 cm downstream location, but the higher the pressure ratio, the less the dispersion of the turbine flow into the surrounding nozzle and ingested air flows.

Conclusions

The operating pressure ratio ($p_{\text{turbine}}/p_{\text{rocket}}$) had a measurable effect on delaying the mixing distance for the one specific configuration of the rocket and turbine exhaust nozzles tested. The turbine exhaust gas exhibited increasingly well-defined confinement of the core flow and greater distance required until full dispersion of the exhaust was realized as the pressure ratio was increased from 1.0 to 2.0. This may be explained by either of two phenomena: 1) Because the rocket nozzle pressure was controlled by the plenum pressure, the nozzle mass flow rates varied inversely with the pressure ratio between the turbine exhaust and the rocket. Because the nozzle pressure ratios (stagnation to the pressure at the point of intersection) remained constant and the static temperature of the flows was held constant, the momentum of the nozzle flow also decreased in direct proportion to the increase in the pressure ratios. Consequently, the greater momentum of the nozzle flow in the case where pressure ratio equals 1.0 serves to confine the vertical expansion of the turbine flow, with consequent increased lateral expansion and more rapid dissipation. 2) The vortex flow initiated where the flows intersect may play a role in containing the central flow of the turbine flow to a greater distance downstream than is the case for the pressure ratio of one.

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Modular Ignition System Based on Resonance Igniter

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Introduction

FURTHER space exploration is centered on rocket engines that operate with powerful propellants based on oxygen. Because propellant combinations involving oxygen are nonhypergolic, the provision of engines by a simple and reliable ignition system is one of several topical problems. Additionally, where engines for upper stage and space system applications are concerned, the introduction of an ignition system with the possibility of multiple starts is also fundamental.[1]

In liquid-oxygen (LOX)-based engines ignition is usually realized through a torch, where spark ignition is considered as the most suitable for multiple starts. However, it is provided at the cost of large amounts of hardware for two independent subsystems fluid and electrical, and hence assurance of the system reliability is not an easy task. In 1970 Phillips et al.[2] produced a torch igniting a gaseous oxygen–hydrogen mixture inside a resonance tube (resonance ignition). By that time it was known that a gas jet accelerated through a sonic nozzle could provoke inside a deep cavity (resonator) shock-wave oscillations with heat release sufficient to ignite propellant mixtures.[3–5]

The resonance ignition is attractive because of the extremely simple configuration and possibility of multiple ignitions. However, operating the ignition system with gaseous hydrogen (GH_2) is coupled

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with an excessively sized GH_2 tank and its components of feeding line.

In the present work, considering that any torch is suitable for propellants ignition, a resonance ignition system, adjusted to operate with gaseous oxygen (GO_2) and liquid fuel instead of GH_2, was designed and tested for evaluating its potential for multiple starts application.

Ignition System Configuration and Characteristics

The new scheme appears in Fig. 1. As shown, compressed GO_2 is used not only as an oxidizer but also as a pressurizing gas for the liquid fuel tank. Such an integration of oxygen and fuel lines allows reduction in the number of fuel line components and, consequently, improves the reliability and decreases the dry mass of the system. The use of liquid fuel, such as ethanol or kerosene, also contributes to reduce the dry mass of the system, as fuel tank and valves have minor volume and mass.

Operation of the system is the following: once the oxygen valve is opened, the oxygen gas passes through the pressure regulator and comes both into the fuel diaphragm tank and into the igniter nozzle. Pressurized oxygen is accelerated through the nozzle and directed into the resonator as an underexpanded jet. This oxygen jet excites shock-wave oscillations and provokes a fast heating of oxygen blocked inside resonator. After some elapsed time, when the oxygen temperature raises to a level enough for ignition, the fuel valve is opened. Distinctly from GO_2-GH_2 igniter,[2,6] where GH_2 is injected into the resonator directly through a sonic nozzle, the fuel on Fig. 1 is injected into igniter chamber and then the drops of fuel are entrained by oxygen jet into the resonator. This design provides ignition of drops by contact with hot oxygen.

In this ignition system there is no need of electrical high-voltage equipment or any other redundant subsystem because the propellant feeding, heating, and ignition are provided only at the cost of potential energy and chemical activity of the stored compressed oxygen. This system can be designed and operated as a modular unit because of its independence from engine propellant lines and minor sizes of elements.

Experimental Setup and Results

The setup shown in Fig. 1 is assembled around the available resonance igniter.[7] However, ethanol instead of kerosene, with mass fraction of 92,8% was selected as a fuel, because it allows operation without soot accumulation in a broad range of torch mixture ratio. The igniter has a body made from cooper as a cylinder block of 60 mm diam and 30 mm height. Inside the igniter there is a combustion chamber of 20 mm diam, with an outlet hole 10 mm diam. On the chamber wall there are a sonic nozzle, a fuel injector, and a resonator. The resonator has a cylindrical cavity with convergent entrance to accelerate the oxygen heating. The inner igniter geometry is designed according to theoretical considerations[8,9] to obtain fast heating inside a resonator with a minimal oxygen flow.

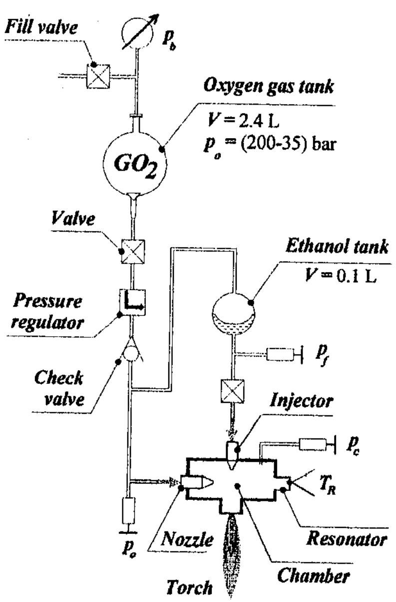

Fig. 1 Scheme of the modular ignition system based on resonance igniter.

As shown in Fig. 1, manometer p_b, transducers p_o, p_f, and p_c were used to detect respectively, pressures in the oxygen tank, in the lines of oxygen and fuel, and in the igniter chamber. Thermocouple T_R was used to detect the resonator's outer surface temperature. All of the experiments were conducted under ambient conditions with an initial temperature of about 300 K. Thus, to realize efficient heating with a minimum oxygen flow the values of oxygen pressure were limited in the range $p_o = (8.0–12.5)$ bar, with the associated oxygen flow rates within the range (25.0–39.0) 10^{-3} kg/s. The present investigation was oriented to a main propellant ignition in a LOX/LH_2 combustor. In this case, as it was shown before,[10] the torch should contain excess of oxygen to improve propellant ignition. Thus, the fuel injector was regulated to produce a torch with mass mixture ratio oxidizer/fuel (O/F) in the range 8.0–10.0.

The computer-controlled measurement system recorded information with an accuracy of 1% and also commanded the oxygen and fuel valves. The following sequence was used for the operation: open the oxygen valve, open the fuel valve, close the fuel valve, and close the oxygen valve. The time interval between the opening of oxygen valve and the opening of fuel valve, called igniter preparation time, was limited within the range (0.1–0.5) s, and the minimum value equal to 0.1 s was dictated by dynamics of feeding lines. The duration of igniter firing operation was fixed as 2.0 s for all tests.

Figure 2 depicts a typical example of record of pressures p_o, p_f, and p_c, and temperature T_R obtained in a single experiment. In this test feeding pressures p_o and p_f were adjusted to 9.5 bar, corresponding to a torch flow rate of $33 \cdot 10^{-3}$ kg/s, and the preparation time was 0.3 s. After ethanol injection the ignition occurred immediately. As result of ignition, the pressure p_c increased up to 4.5 bar, and the torch was observed. In all tests values of time delay of ignition and T_R did not exceed the values of 0.05 s and 100 °C.

The present procedure was used to check the workability of ignition system at several combined values of p_o and preparation time. The mapping of these combinations is shown in Fig. 3. It is seen that with preparation time equal 0.1 s, ignitions occurred with $p_o > 9.5$ bar. At reduced values of p_o, the heat released inside resonator was not sufficient for fast oxygen heating caused by atmospheric pressure influence. Thus, to produce a torch with $p_o < 9.5$ bar the preparation time should be prolonged.

The ignition test was repeated using an oxygen tank of 2.4 L volume under initial pressure of 170 bar. It was demonstrated that ignition system allowed up to eight ignitions with conditions of Fig. 2. The tested ignition system was assembled with components for industrial application, except the specially designed diaphragm-tank and the available resonance igniter. The system dry mass was

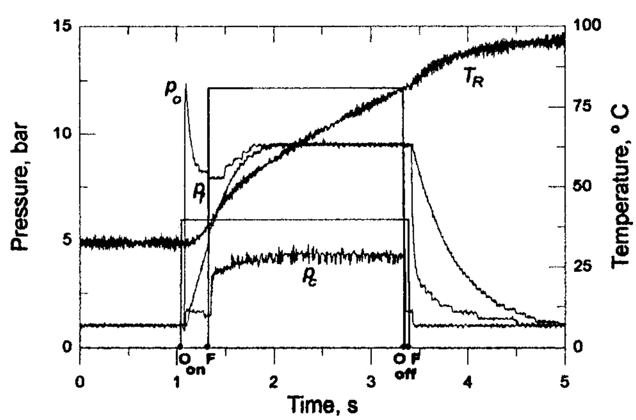

Fig. 2 Record of pressures p_o, p_f, and p_c, and temperature T_R during an ignition test.

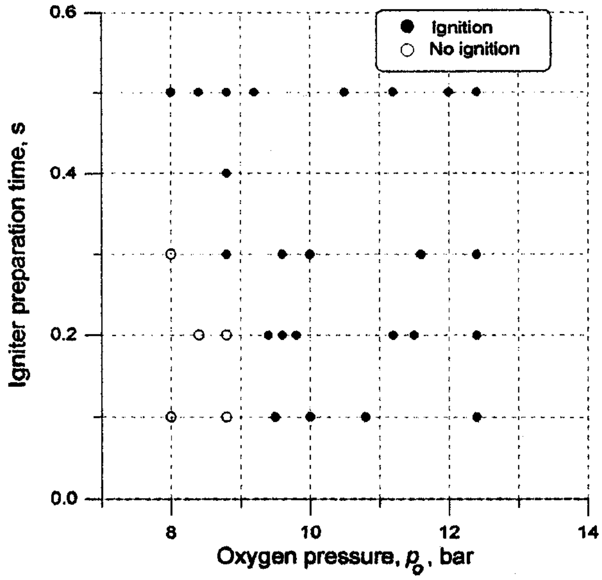

Fig. 3 Effect of pressure p_o and igniter preparation time on ignition.

about 7 kg, and the 2.4 L oxygen tank was the heaviest component: 1.66 kg designed for nominal pressure of 200 bar. Hence, the ignition system took no account of a compromise between mass and number of ignitions, and, therefore, it can be lighter for flight applications.

Conclusions

An ignition system based on resonance igniter was designed and tested, which 1) can operate independently of engine propellant lines only at the cost of the energy of stored compressed oxygen; 2) contains reduced number of components and can be designed as a modular unit, for a wide range of mass flow rate and mixture ratio; and 3) allows up to eight ignitions with an oxygen tank of 2.4 L under 170 bar, for 2 s of torch duration and $33 \cdot 10^{-3}$ kg/s of torch flow rate.

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Experimental Investigation of Pulsatile Flow in Circular Tubes

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Nomenclature

d	=	tube diameter
Q	=	volume flow rate
Re	=	Reynolds number
r	=	distance from tube axis
T	=	period
v	=	velocity
α	=	frequency parameter
λ	=	flow ratio
ν	=	kinematic viscosity
ρ	=	fluid density
σ	=	Reynolds normal stress
ω	=	angular frequency

Introduction

THE investigation of pulsatile Newtonian fluid flow in circular rigid pipes was performed by the authors. The aim of the research was to deepen the knowledge of impact of model geometry and flow characteristics on origination and development of Reynolds stress and on the value of energy loss during pulsatile flow. Interesting results that were obtained will be used in practice. Velocity profiles were measured with a laser-Doppler anemometer. The assemble-average velocity profiles and the Reynolds normal stress have been experimentally evaluated. Knowledge of origination and development of turbulent disturbances helps us to create an image of transformation of flow into turbulence. The evaluation of the Reynolds normal turbulent stress helps us to determine the values of Reynolds tangential stress, which causes hydraulic loss in the tubes. In hemodynamics the Reynolds tangential stress has a great importance for possible damage of blood elements and inner surface of blood vessels.

At the same time the pressure loss in circular tubes of a constant cross section was measured, and it was evaluated in dependence on parameters of pulsatile flow. The result of this measurement was the determination of dependence of loss coefficient on flow characteristics. Knowledge of these relationships can result in the minimization of the energy loss in pulsatile flow.

We have also evaluated velocity profiles, Reynolds stress, and the pressure loss in singularities formed by sudden expansion and consequential sudden contraction of the tube. The result is dependence of loss coefficients on the pulsatile flow parameters. Knowledge of these relationships provides for better understanding of the pulsatile flow mechanisms exploitable in industrial applications as well as in hemodynamics, and it can result in remarkable energy savings.

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