Performance and Stability Characteristics of a Uni-Element Swirl Injector for Oxygen-Rich Stage Combustion Cycles

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ABSTRACT

A uni-element liquid propellant combustion performance and instability study for liquid RP-1 and hot oxygen-rich pre-burner products was conducted, at a chamber pressure of about 1000 psi, using flush and recessed swirl injectors. High-frequency pressure transducer measurements were analyzed to yield the characteristic frequencies which were compared to expected frequencies of the chamber. Modes, which were discovered to be present within the main chamber included, the first longitudinal, detected at approximately 1950 Hz, and the second longitudinal mode at approximately 3800 Hz. An additional first longitudinal quarter wave mode was measured at a frequency of approximately 23000 Hz for the recessed swirl injector configuration. The characteristic instabilities resulting from these experiments were relatively weak averaging 0.2% to 0.3% of the chamber pressure.

INTRODUCTION

In the 1940's and 1950's there was a shift from ethanol based fuels to kerosene based fuels due to the heavy-lift requirements of ballistic missile missions. Although by the 1960's Liquid-oxygen (LOX)/RP-1 systems were discontinued due to storability issues, many of the rockets which were subsequently developed, the U.S. Saturn, Delta, and Atlas launch vehicles all use LOX/RP-1 in the high-thrust first-stage booster engines. LOX/RP-1 is still regarded as a promising propellant combination relative to other storable propellants because of its high theoretical specific impulse, and impressive bulk density [1]. Before new systems can be integrated with this fuel combination, one major problem needs to be addressed scientifically. This problem involves combustion instabilities and the effect that instabilities have on the structural integrity of the combustion chamber and the overall performance of the rocket engine.

Combustion instability, the result of coupling between the fluid mechanics/acoustics of the chamber and the combustion process, is a very complicated and intricate problem. In order to solve the combustion instability problem the physics of the phenomenon must be fully understood. Instability issues have plagued the rocket community since the 1950's when investigations of the Thor and Atlas ballistic missiles were conducted [2-4]. Since then, many costly investigations have been completed and yet, an understanding based on the fundamental physics of the problem still eludes us. Many stable engines have been developed through trial and error, or by simply introducing changes to the geometry of the system, using baffles, resonators, or other devices [5-7]. An attempt to create a comprehensive resource of all information concerning combustion instabilities was commissioned in 1967 by NASA and the U.S. Armed Forces and two scientists, Harrje and Reardon, took on the challenge of compiling this comprehensive reference. The final result was NASA SP-194 published in 1972 [4].

Key aspects included in NASA SP-194 are the characterization of instabilities into two categories, acoustic and non-acoustic instabilities, recommendations for stable chamber design, including baffle and acoustic cavity geometries, and simple computational models such as Priem and Heidmann's drop vaporization model [8]. In addition, there were injection element design guidelines with references to particular oxidizer/fuel combinations.

Although NASA SP-194 is clearly a comprehensive reference, there are two limiting factors of this report. The first factor was the lack of accurate computational models on combustion instabilities due to the limited computing resources available in the late 1960's and early 1970's. The second is the absence of correlations between the influence of injectors and chamber design on the stability of the system [1]. The latter missing component from NASA SP-194 is the key research objective that needs to be addressed.

There have been many advances since NASA SP-194 both experimentally and computationally. Some of the major contributions are, Pavli, in association with NASA Lewis Research Center (currently the NASA Glenn Research Center), who conducted a series of tests to evaluate several injector elements designs [9], the LOX/Hydrocarbon Injector Characterization Program which took an analytical approach in evaluating LOX/HC engines [10,11], Rocket Engine Analytical Design Methodology Development Program [12], the Heavy Hydrocarbon Main Injector Technology Program [13], and the Pressure Fed Booster project built by Aerojet for NASA Marshall Space Flight Center [14].

The Rocket Engine Advancement Program 2 (REAP2) initiated in March 2003 is a NASA funded university consortium with a mission focused on research and education in advanced propulsion for space exploration that was created to address specific propulsion research issues through a coordinated university/NASA team. The REAP2 program includes leading researchers at University of Alabama, Huntsville, Penn State University, Purdue University, Auburn University and Tuskegee University who have assembled a coordinated research program that aims to scientifically investigate the problems of combustion instability and thrust chamber cooling.

This paper describes the current status of an on-going research effort at Penn State on combustion instability for oxygen/RP-1 injector configurations. The program is also closely coordinated with a complementary modeling effort at Purdue University [15]. This synergistic experimental/modeling program is geared towards understanding combustion instability in general, but specifically, for main chamber injectors in LOX/RP-1 rocket engines utilizing an oxidizer-rich preburner. For such an engine, the preburner propellants are LOX and liquid RP-1 at oxidizer-rich conditions (O/F~50) and the main chamber propellants are the preburner gaseous products (~92% gaseous oxygen (GO_2) and 8% carbon dioxide (GCO_2) and water vapor (GH_2O) at 1100-1300 R) and liquid RP-1. Injector designs for these propellants, viz. liquid RP-1 and "hot" vitiated oxygen have not been investigated thoroughly in the US, and consequently the stability characteristics of such injectors are relatively unknown. The goal of the effort at Penn State is to experimentally assess the combustion performance and stability characteristics of this "type" of injector under both uni-element conditions where longitudinal instabilities may be prevalent, and multi-element conditions where transverse instabilities could be more prevalent. This paper describes the combustion performance and stability research conducted to date on a uni-element main chamber injector configuration.

EXPERIMENTAL

The current experiments were conducted at the Cryogenic Combustion Laboratory (CCL) housed in the Propulsion Engineering Research Center (PERC) at Penn State. This facility is a unique university facility where small scale propulsive concepts have been tested for over a decade. Since its inception, the laboratory has been vigorously involved in uni-element rocket [16-18] and Rocket Based Combined Cycle (RBCC) rocket-ejector [19-20] experimentation. The maximum flowrate capabilities of the laboratory are 1.0 lbm/s for both GO_2 and LOX, 0.25 lbm/s for gaseous hydrogen and 0.5 lbm/s for liquid hydrocarbons. The maximum tested chamber pressure to date is 1400 psia, although the majority of the experiments have been conducted at chamber pressures below 1000 psia.

UNI-ELEMENT ROCKET SETUP

The experiments reported here were conducted using an optically accessible rocket chamber integrated with an oxidizer-rich preburner. A schematic of the integrated assembly and a photograph of a firing for the current experiments are shown in Figures 1 and 2, respectively. The Oxygen Free High Conductivity (OFHC) copper heat-sink main rocket chamber was designed in a modular fashion to

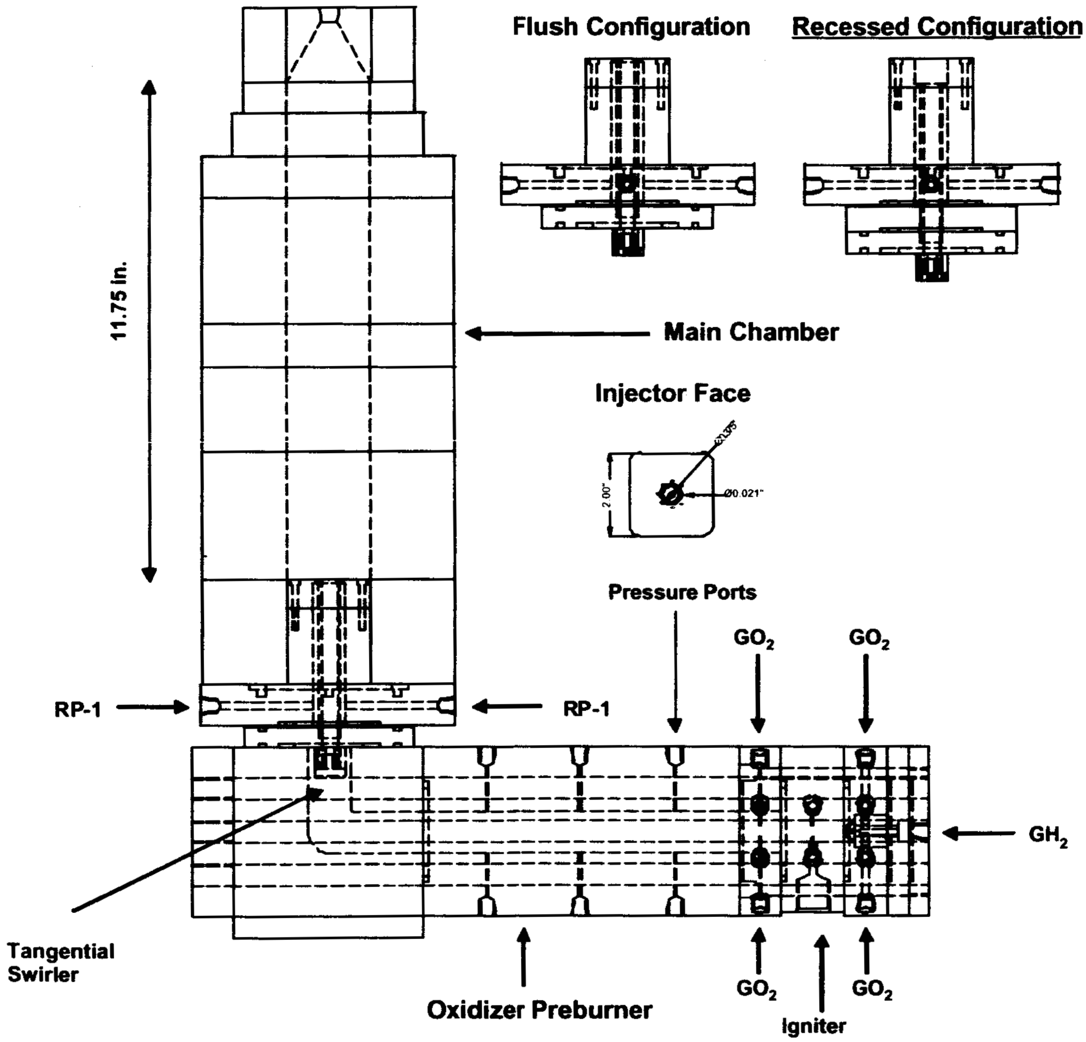


Figure 1. Schematic of the main rocket chamber integrated with the oxidizer-rich preburner. The oxidizer-rich preburner utilizes a near stoichiometric core/downstream dilution design approach for providing oxidizer-rich products to the main injector. The preburner products are tangentially swirled through the central passage of the main injector. Liquid RP-1 fuel is introduced into the RP-1 manifold and emanates into the main chamber through eight holes that circumferentially surround the central oxidizer circular passage. The injector face and both flush and recessed injector configurations are also depicted in this schematic. Additional details on the injector geometry are provided in Table 1. Optical access and pressure transducer instrumentation are not shown in figure.

Figure 2. Representative photograph of a rocket firing.

easily provide optical access along the chamber length. The main rocket chamber is comprised of several sections that include the injector assembly integrated with the oxidizer-rich preburner, gaseous oxygen/hydrogen igniter, window and blank sections, and a nozzle assembly. These sections are held together by a hydraulic jack that allows for ease of assembly and arrangement of the various sections. The chamber length can be varied by inserting or removing blank sections. For the present experiments, the chamber length from the main injector face to the nozzle entry was 11.75 in. Two diametrically opposed windows, 2 in. in diameter and 1 in. thick, provide optical access into the 2 × 2 in. main rocket chamber. All windows are protected from the hot combustion gases by a gaseous nitrogen (GN_2) curtain purge which flows across each of the interior window surfaces. Lastly, the water-cooled nozzle assembly is also modular in design. Nozzles of different throat diameters can be interchanged, thus providing the capability for varying the chamber pressure.

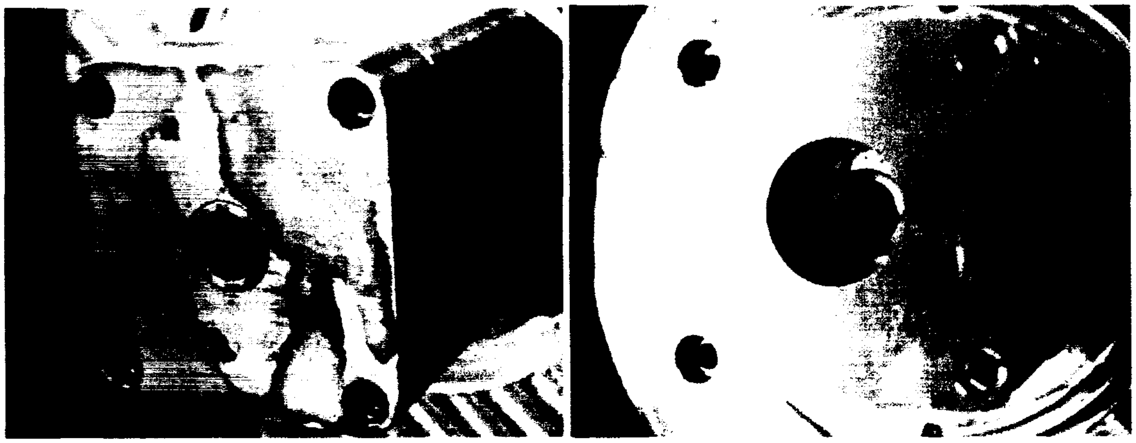
INJECTOR DESIGN AND RUN CONDITIONS

The design of an injector element for liquid RP-1 and vitiated hot oxygen is a new frontier for U.S. engines. Injector element designs in the U.S. can be broadly categorized into three categories, viz. impinging, shear coaxial and swirl coaxial. For propellant combinations where one of the propellants is a liquid and the other is a gas, the latter two configurations are more amenable for design. In comparison to swirl coaxial injectors, shear coaxial injectors are typically poor atomizers but in general exhibit lower injector face heat transfer characteristics. However, in terms of designing compact engines, the swirl injector is more attractive since the combustion zone is reduced. For the present injector, the target conditions were a mixture ratio of approximately 2.9 and injector propellant pressure drops between 10 and 15% of the main chamber pressure, with the main chamber pressure targeted to be around 1000 psia (~1/3 full scale). These guidelines were chosen for the injector design to be scalable to full scale rocket conditions of 3000 psia. With these design guidelines, central swirling of the vitiated hot oxygen flow with an annular flow of RP-1 represented the ideal solution. Additional swirling of the RP-1 was also considered but proved to be a design challenge due to the extremely small passage required to maintain the desired pressure drop. A conventional annular passage surrounding the central swirler was theoretically feasible but the pressure drop requirement yielded a design with a very small annuluar gap (< 0.010 in.) that would be hard to manufucture within tolerance. The final chosen design was centrally swirled vitiated hot oxidzer flow with multiple individual orifices arranged circumferentially around the central swirler for the RP-1 propellant. The tangential swirler design for the central oxidizer flow was based on the incompressible design guidelines provided by Doumas and Laster [21]. A summary of the

Table 1. Geometry specifications and target flow conditions.

Preburner Propellants	
GO_2 Flowrate (lbm/s)	0.3941
GH_2 Flowrate (lbm/s)	0.0023
Total Flowrate (lbm/s)	0.3964
O/F	170
Preburner Products	
GO_2 Flowrate (lbm/s)*	0.3757
GH_2O Flowrate (lbm/s)*	0.0207
Total Flowrate (lbm/s)	0.3964
Product Temperature (R)**	1110
Main Injector (Oxidizer Path; From Preburner)	
Swirl Injector Post Diameter (in.)	0.375
# of Rectangular Tangential Inlet Slots	3
Rectangular Tangential Slot Width (in.)	0.0625
Rectangular Tangential Slot Length (in.)	0.35
Swirl Angle (°)***	61
Oxidizer Tangential Velocity (ft/s)***	233.1
Oxidizer Axial Velocity (ft/s)***	392.5
Oxidizer Total Velocity (ft/s)***	456.5
Oxidizer Pressure Drop (psia)***	126
$\Delta P_{oxidizer}/P_c$	13.2
Main Injector (Fuel Path)	
RP-1 Flowrate	0.1277
# of circular fuel holes	8
Fuel hole diameter (in.)	0.021
Fuel Velocity (ft/s)	156.7
Fuel Pressure Drop (psia)	107.3
$\Delta P_{fuel}/P_c$	11.2
Main Injector	
Injector Oxidizer/RP-1 Flowrate Ratio	3.1056
Injector GO_2 in Oxidizer/RP-1 Flowrate Ratio	2.9433
Main Chamber	
Chamber Length (in.)	11.75
Chamber Cross-section (in. × in.)	2 × 2
Nozzle Throat Diameter (in.)	0.38
Chamber Pressure, P_c (psia)	955

*based on CEA calculations [22]
**based on CEA calculations [22] and previous experience with heat loss to walls in preburner
***design numbers based on Doumas and Laster [21]



(a) (b)

Figure 3. (a) Flush and (b) recessed injector configurations. For each configuration, the oxidizer-rich preburner products are swirled through the central passage, whereas liquid RP-1 flows through the eight smaller circumferentially arranged orifices. Dimension details of the injector are provided in Figure 1. For the recessed injector, the circular mixing cup has a diameter of 0.75 in. and a depth of 0.6 in.

final design is shown in Figure 1 with injector design characteristics highlighted in Table 1. The injector was designed such that both "flush" and "recessed" configurations could be investigated. For the flush injector configuration, both propellants are injected at the injector face, whereas for the recessed injector configuration, both propellants can partially mix and combust in a mixing cup before emanating from the injector face plane. Photographs of both configurations are presented in Figure 3.

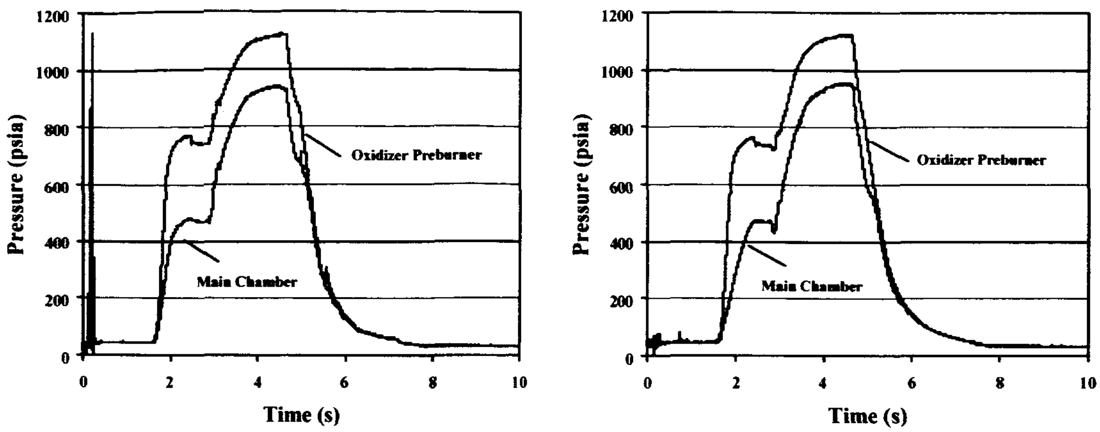
OXIDIZER-RICH PREBURNER DESIGN

As mentioned earlier, the oxidizer-rich preburner for the full scale rocket engine would employ LOX and RP-1, with the gaseous products being mostly hot oxygen with a small percentage of GCO_2 and GH_2O. Due to the complexity of running LOX and RP-1 in the oxidizer-rich preburner, a simpler approach was utilized. The preburner used for the current experiments operated on GO_2 and GH_2 propellants with product constituents of gaseous hot oxygen and GH_2O. The overall mixture ratio of the oxidizer-rich preburner was chosen such that the mass fraction of the oxygen in the products as well as the product temperature mimicked oxidizer-rich LOX/RP-1 combustion. Target flow conditions for the oxidizer-rich preburner are also summarized in Table 1.

The preburner was designed to integrate with the main chamber injector as also shown in Figure 1. Since the mixture ratio of the preburner was extremely, high, viz. 170, the design was based on the near-stoichiometric core/downstream dilution philosophy. The preburner injector has a 0.5 in. diameter, 1.5 in. long pre-combustion chamber where an impinging injector provides the propellants at a target mixture ratio of about 30. Further downstream, the preburner has a circular cross-section with a diameter of 1.0 in. Additional oxygen is introduced radially inwards through four holes to mix with the near-stoichiometric core from the pre-combustion chamber. The products (overall O/F~170) then negotiate a 90° bend before entering three rectangular tangential entry slots that impart swirl to the main injector flow.

DIAGNOSTICS

The goal of these initial experiments was to evaluate the combustion performance and stability characteristics of the injector configurations. The performance of the injector designs were evaluated through c* efficiency calculations. The stability characteristics of the injector configurations were



(a) (b)

Figure 4. Representative rocket main chamber and oxidizer preburner pressure versus time traces for (a) flush and (b) recessed injector configurations.

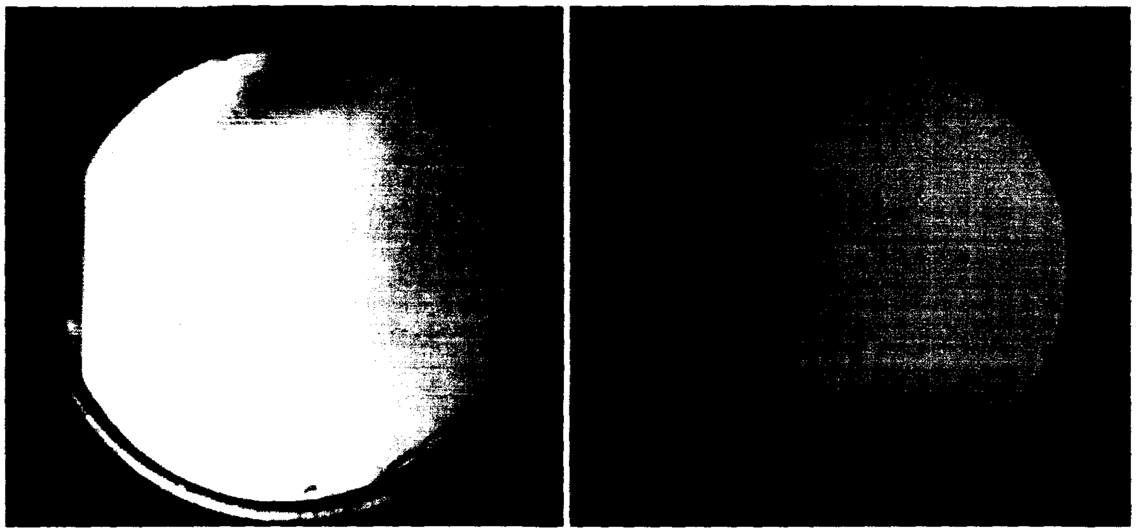
assessed through spectrum analysis of high frequency pressure transducer measurements sampled at 50 kHz. Note that the pressure transducer measurement locations are not indicated in Figure 1. For any given rocket firing, three high frequency piezoelectric pressure transducers manufactured by PCB were mounted along the length of the main combustion chamber. Through the course of the experiments, high frequency pressure measurements were made at axial locations of 1.0, 4.0, 7.5 and 10.5 in. from the injector face. A limited number of experiments were also conducted with a window section providing optical access to the near injector flowfield in the main chamber. 35 mm photography was employed as the qualitative diagnostic for obtaining visible and UV emission images of the near injector flowfield.

RESULTS AND DISCUSSION

Representative pressure versus time traces for rocket firings are presented in Figure 4 for both the flush and recessed injector configurations. For the firings, a two-step procedure was utilized for getting to the main stage pressure. In the first step, the preburner was ignited at the full target flowrate conditions with GH_2 flow through the RP-1 passage. Once this stage was successfully achieved, the GH_2 flow was turned off and the full target RP-1 flow was turned on. This two-step transition procedure allows the main chamber and oxidizer-rich preburner to achieve the main target conditions safely. The steady state portion of the firing was kept to a short time (~1.0 s) since longer firings resulted in window section and/or nozzle throat damage.

The c* efficiencies for both injector configurations were calculated to be between 1.0 and 1.03 for all firings. Note that c* efficiencies higher than 1.0 can be realized due to inaccuracies in propellant mass flow rate measurements and the nozzle throat diameter measurement. In any case, combustion performance for both configurations is high. The measurements also do not indicate any systematic performance superiority of one configuration over the other.

Representative visible and UV emission images of the near injector flowfield are presented in Figure 5 for the flush injector configuration. The visible image clearly shows that the combustion zone rapidly increases radially with axial distance from the injector face. This behavior is characteristic of swirl injector designs, where the tangential momentum of the central swirling flow distributes the propellants in the radial direction. The image also shows indications that there are individual flames for each of the individual RP-1 jets. Note that the dark region evident in the downstream part of the image is due to soot deposition on the window. The corresponding UV image shown in Figure 5 also highlights the effect of

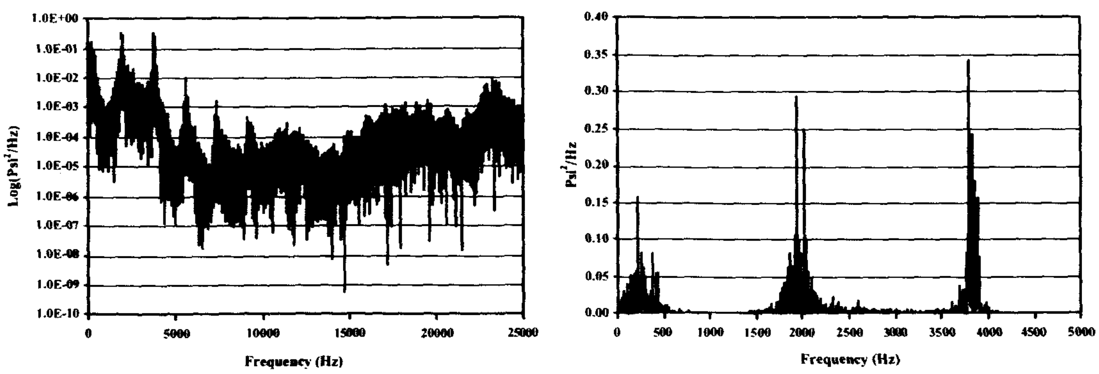


(a) (b)

Figure 5. Representative (a) visible and (b) UV light emission images of the near injector face region for the flush injector configuration. The individual RP-1 jets can be clearly seen in the UV image. The diameter of the window is approximately 1.9 in.

swirl on the radially expanding combustion zone. In this image, the individual RP-1 jets emanating from the injector face can be clearly seen.

The high frequency pressure transducer measurements were analyzed for energy content as a function of frequency. The results of this endeavor are discussed next. A representative power spectrum from a firing for the recessed injector configuration is shown in Figure 6 for measurements at the 1 in. axial location from the injector face. The full power spectrum is shown on the left (0-25kHz) on a log-linear plot, whereas the 0-5 kHz range is shown on a linear-linear plot on the right. The full spectrum



(a) (b)

Figure 6. Frequency spectrum (psi^2/Hz) of measured chamber pressure oscillations for the recessed injector is plotted versus frequency (Hz). The pressure signal was sampled at 50 kHz at a location 1 in. from the injector face. (a) Full frequency range with spectrum on a log-linear plot, and (b) frequency spectrum highlighting the 0-5000 Hz frequency range.

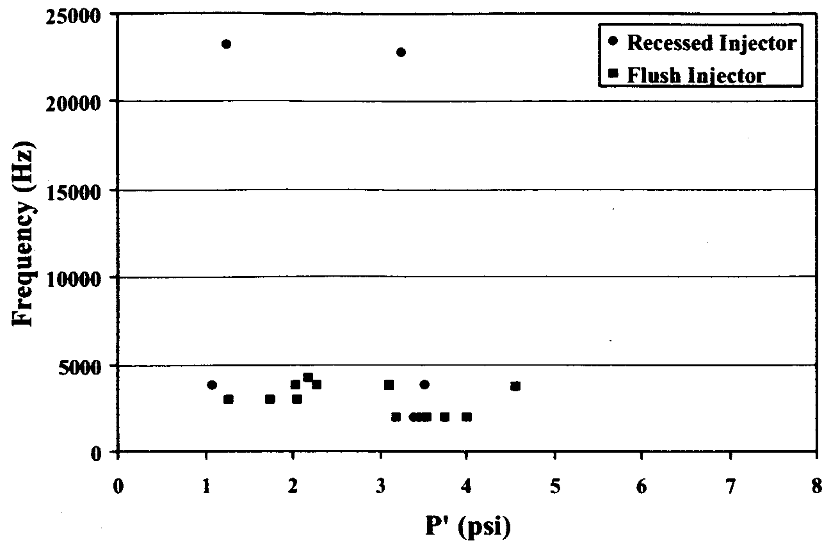


Figure 7. Comparison of RMS pressures for all runs for both flush and recessed injector configurations. Measurements shown are from 2 and 4 firings each for the recessed and flush injector configurations, respectively. The high-frequency pressure transducer was located 1 in. from the injector face.

shows peaks at approximately 1950 Hz and its multiples. The 1950 Hz frequency corresponds closely to the calculated first longitudinal mode (1L) of the main chamber. The energy content of the second longitudinal mode (2L) at approximately 3800 Hz is similar to that of the first longitudinal mode as is highlighted in plot (b) of Figure 6. Higher longitudinal modes (up to 5L) with progressively diminishing energy content are also distinguishable in the full spectrum.

The pressure amplitudes (P') for "energetic" modes are plotted versus frequency in Figure 7. In the plot, results for both the flush (4 firings) and recessed (2 firings) injector configurations are shown for the 1 in. axial location from the injector face. These results show that the 1L and 2L modes of the chamber have the highest pressure amplitudes. For both injector configurations, the 1L mode has pressure amplitudes between 3 and 4 psi. The 2L mode pressure amplitude shows significantly more scatter from run to run. The recessed injector also shows a mode at the very high frequency of around 23 kHz which is not present in the power spectrums for the flush injector configuration. This mode is also measured only for the 1 in. axial location and is not evident in the further downstream power spectrum measurements. This frequency corresponds to a longitudinal quarter wave in the recessed injector's mixing cup.

SUMMARY/FUTURE WORK

The experimental results discussed here indicate that both the flush and recessed swirl injectors are high performing and relatively stable designs. The power spectrums of the high frequency pressure measurements indicate that combustion excites the first and second longitudinal modes, albeit the pressure amplitudes are less than 1% of the chamber pressure.

The initial set of experiments described in this paper has yielded a high performing injector design for the vitiated hot GO_2/RP-1 propellant combination. Future experiments will focus on design and testing of a linearly arranged multi-element rectangular combustion chamber for studying the effects of transverse combustion instabilities. CFD analysis of this design indicates that chamber transverse modes can be induced by oscillating the flow of one or two element of the linear array of elements [15].
The synergy between the complementary CFD and experimental studies is expected to provide improved understanding of the combustion instability phenomenon.

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