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CHARACTERIZATION OF OSCILLATIONS DURING PREMIX GAS TURBINE COMBUSTION

BREAK

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ABSTRACT

The use of premix combustion in stationary gas turbines can produce very low levels of NO_x emissions. This benefit is widely recognized, but turbine developers routinely encounter problems with combustion oscillations during the testing of new premix combustors. Because of the associated pressure fluctuations, combustion oscillations must be eliminated in a final combustor design. Eliminating these oscillations is often time-consuming and costly because there is no single approach to solve an oscillation problem.

Previous investigations of combustion stability have focused on rocket applications, industrial furnaces, and some aeroengine gas turbines. Comparatively little published data is available for premixed combustion at conditions typical of an industrial gas turbine.

In this paper, we report experimental observations of oscillations produced by a fuel nozzle typical of industrial gas turbines. Tests are conducted in a specially designed combustor, capable of providing the acoustic feedback needed to study oscillations. Tests results are presented for pressures up to 10 atmospheres, and with inlet air temperatures to 588 K (600 F) burning natural gas fuel.

Based on theoretical considerations, it is expected that oscillations can be characterized by a nozzle reference velocity, with operating pressure playing a smaller role. This expectation is compared to observed data, showing both the benefits and limitations of characterizing the combustor oscillating behavior in terms of a reference velocity rather than other engine operating parameters. This approach to characterizing oscillations is then used to evaluate how geometric changes to the fuel nozzle will affect the boundary between stable and oscillating combustion.

INTRODUCTION

Lean premix combustion is now accepted as a standard approach to reduce NO_x emissions from stationary gas turbine combustors. The lean-premix (LPM) combustor is designed to avoid high-temperature, stoichiometric combustion that produces NO_x by the thermal mechanism. With careful premixing of fuel and air, it is possible to achieve both excellent NO_x and CO emissions. Development programs from several turbine vendors include various versions of the LPM concept to achieve the lowest possible pollutant emissions (Alsup et al. 1995).

Although premixing fuel and air can produce very low NO_x emissions, practical application of LPM combustion is restricted by the problem of operating very close to the lean flammability limit. Slight changes in operating conditions can lead to sudden flame extinction, or to excessive CO emissions. This problem of flame *static stability* is often addressed by adding a smaller pilot flame to anchor the main premix flame. By operating the pilot flame closer to stoichiometric, the main flame can be maintained even during momentary upsets. The drawback is that the pilot contributes to NO_x emissions.

In addition to static stability, LPM combustors must achieve *dynamic stability*, meaning the combustion must not oscillate. It has been common experience that operation near the lean-extinction limit is often accompanied by oscillating combustion. Oscillation must be eliminated in a final combustor design because the associated pressure fluctuations can shorten the engine component lifetime. Keller (1995) discusses the significance of this problem, and points out that operation near the lean limit is especially prone to oscillation problems. Near the lean limit, minor variations in fuel/air ratio lead to appreciable variations in combustor reaction rate. When these variations in reaction rate couple to acoustic modes, significant pressure oscillations can occur with frequencies ranging from hundreds to thousands of Hertz.

Few studies of LPM combustion oscillation have been reported at representative gas turbine conditions. In practice, the task of studying (and eliminating) combustion oscillation in a gas turbine is complicated by the specific acoustic response of a given combustor design. Thus, no general trends have been proposed to describe the effect of usual gas turbine parameters such as inlet air temperature, operating pressure, and equivalence ratio. The effect of these parameters is not well understood. Consequently, tests of proposed oscillation remedies are complicated because it is difficult to specify an appropriate test matrix.

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In this paper, we report observations of oscillations from a premix combustor that is specially designed to study oscillations at gas turbine operating conditions. Test results are presented for pressures to 10 atmospheres, with inlet air temperatures to 589 K (600 F). Results are compared to expectations from a simple model for combustion oscillations.

BACKGROUND

Previous studies of combustion oscillations have focused on various technical applications. During the 1960s, considerable research was devoted to solving the problem of combustion oscillations in liquid rocket motors. An excellent review of this research was published in 1972 by Harrje and Reardon. Recent progress in liquid rocket combustion oscillations is described in the text edited by Yang and Anderson (1995). Routine problems with oscillating combustion during industrial burner development are discussed in the textbook by Putnam (1971). Schadow and Gutmark (1991) summarize a series of studies on the oscillations due to periodic vortex shedding in dump combustor applications. Candel (1992) reviews the oscillating mechanism in a number of burner applications.

Few previous studies have analyzed oscillations from premixed, swirl-stabilized flames that are typical of modem gas turbine combustors. Sivasegaram and Whitelaw (1991) studied oscillating combustion in swirl-stabilized flames, but the focus of the research was not on the discrete-tone oscillations that are typical of LPM combustors. Richards and Yip (1995) report preliminary data from a laboratory-scale burner that incorporates swirl-stabilization and a pilot flame similar to many gas turbine fuel nozzles. These authors showed that some oscillations could be described by the time-lag approach (described later). However, these lab-scale tests were limited to one atmosphere pressure and ambient inlet air temperature. A small number of articles report combustion oscillations at gas turbine conditions (Mehta et al. 1990a, 1990b), but focus on liquid-fueled aeroengine applications.

The exact features of specific burner configurations introduce a variety of mechanisms responsible for sustaining combustion oscillations. Variations in heat release can result from changes in mixing, changes in the supply of fuel and/or air, vortex shedding, or other mechanisms. In a practical application, it is often difficult to identify the mechanism responsible for variations in the heat release because several mechanisms can occur simultaneously, or may be dominant at different operating conditions. For example, pressure disturbances in a combustion chamber will change the instantaneous delivery of fuel and air or both, and all of these will affect the heat release. To date, there has been no general approach to account for the *combined* variation of these and other parameters. Feiler and Heidman (1967) proposed that stability behavior can be analyzed with a linear combination of the various mechanisms, but this method ignores the interaction between mechanisms.

No matter what the specific mechanism, all oscillations involve a feedback between variation in the local heat release rate Q(t) and the acoustic pressure field inside the combustor P(t). The well-known Rayleigh criterion (Rayleigh 1878) states that oscillations will be likely if the changes in heat release are in phase with the acoustic pressure disturbances. Conversely, oscillations are dampened if the heat release fluctuations are out of phase with pressure oscillations. It is necessary to integrate across the entire combustor volume to evaluate the cumulative effect of local fluctuations because P and Q are functions of both space and time. For clarity in this discussion, we consider just the time dependence.

The Rayleigh criterion has been the cornerstone for the development of many analyses of combustion oscillations. As early as 1956, Merk developed a linear analysis for premixed combustion oscillations that predicted stable and oscillating regions as a function of burner heat output. Merk proposed that heat release variation would result from changes in flame structure produced by acoustic pressure disturbances. The time delay between the pressure disturbance and the heat release variation determined the phase, and therefore whether the system was stable or likely to oscillate. In a similar fashion, characterization of oscillating combustion in liquid rockets was also recognized to depend upon a time delay (or time lag) between acoustic pressure and the subsequent variation in heat release rate (Crocco and Cheng, 1956). As explained by Harrje and Reardon (1972), the time lag and the gain of the system (denoted n) have been widely used to explain experimental data from rocket oscillations. Likewise, Putnam's approach to solving oscillations in industrial burners relies on determining the time lag between P(t) and Q(t); see Putnam (1971).

Based on these earlier studies, it is reasonable to expect that oscillations from LPM combustors can be characterized by a simple time lag approach. Figure 1 shows a schematic of the important processes for a specific case, where a sinusoidal pressure disturbance produces a sinusoidal variation in the air flow, but 180 degrees out of phase with the pressure. As shown, the time lag τ is estimated as the distance between the point of fuel injection and flame front divided by the average axial velocity:

$$\tau = (L+L')/U_{avg} \qquad (1)$$

where

L is the distance from the fuel injection point to the tip of the nozzle.

L' is the distance from the tip of the nozzle to the flame front.

U_{avg} is the average velocity of mixture in the fuel nozzle.

In this example, a positive pressure fluctuation in the combustor produces a momentary decrease in air flow. We assume that the fuel supply is choked, so that the fuel flow rate does not change as the pressure varies. Thus, the reduced air flow receives a proportionally higher amount of fuel, creating a fuel-rich pocket. This richer pocket will arrive at the flame front with a time lag given by (1). If this "extra" fuel produces an immediate increase in the heat release, oscillations will be most likely when the peak of the pressure fluctuation is in phase with the increased heat release; i.e., when the time lag t_2-t_1 is an integer multiple of the acoustic period. This criterion for oscillations can be stated as:

$$\text{(time lag)/(acoustic period)} = 1, 2, 3.... \qquad (2)$$

Noting that the acoustic period is the reciprocal of frequency f, and using (1), we have:

$$\text{(time lag)(frequency)} = 1, 2, 3... \text{ or,}$$
$$f\,(L+L')/U_{avg} = 1, 2, 3... \qquad (3)$$

This is a restatement of Rayleigh's criterion and again requires that the heat release and pressure fluctuations should be in phase to promote oscillations. In practice, it is not necessary that the heat release

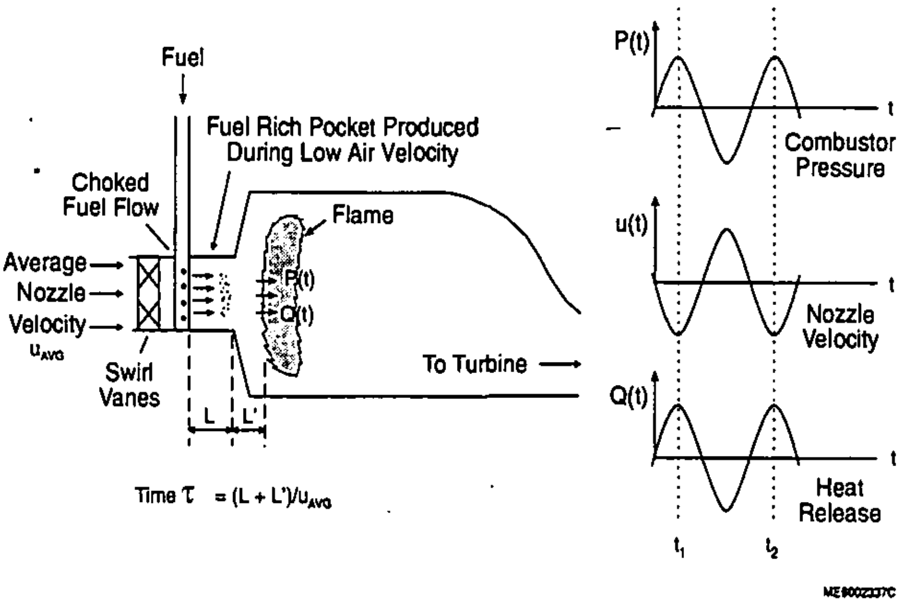


FIGURE 1. SCHEMATIC OF PROCESSES IN AN EXAMPLE OSCILLATION. IN THIS EXAMPLE, THE PRESSURE P(t) CHANGES THE AIR FLOW. THE AIR VELOCITY U(t) IS 180 DEGREES OUT OF PHASE WITH THE COMBUSTOR PRESSURE, AND PRODUCES A FUEL-RICH POCKET AT TIME t_1. THIS POCKET ARRIVES AT THE FLAME TO INCREASE Q(t) AT TIME t_2.

and pressure be exactly in phase to drive oscillations. Some driving will occur for heat release fluctuations that lead or lag the pressure by as much as 1/4 of the acoustic cycle (Putnam 1971). Thus, there is a range of plus/minus 0.25 around each integer in the series (3) where driving is *possible*, but driving is greatest for the integer values where the pressure and heat release are exactly in phase.

The criterion (3) is specific to the example where positive pressure produces an immediate decrease in air flow, and assumes that the fuel-rich pocket produces an immediate increase in reaction rate when it arrives at the flame front. In practice, other mechanisms for variable heat release can complicate the criterion for oscillations such that the series expressed by (2) will have numeric values other than 1, 2, 3..., etc. Richards and Yip (1995) show that a similar criterion can be developed which accounts for the fuel system impedance, or can also describe oscillations apparently linked to transport of tangential velocity from the fuel nozzle swirl vane. Putnam (1971) analyzed industrial burners and shows how to account for the geometry of the flame front in determining the numeric series.

Although the numeric series (3) will be different for different situations, the idea that oscillations can be characterized using the average nozzle velocity provides a convenient starting point to analyze data from gas turbine tests. In a gas turbine, the fuel nozzle mass flow, pressure, and inlet air temperature are all variables. This complicates evaluation of nozzle stability because tests must be conducted over a wide range of parameters. In contrast, if equation (3) is valid, tests can be conducted over a range of nozzle velocities, rather than independent values of mass flow, pressure, or inlet air temperature.

The proposed approach is complicated by the flame standoff distance L'. Clearly, as the equivalence ratio rises, or as the inlet air temperature is increased, the premixed flame speed will rise, and the flame standoff will decrease. The geometry of the flame will introduce further complication in determining L'. Although these complications are recognized, in this paper we will evaluate the stability behavior of a fuel nozzle based on the simple concept of a fixed flame standoff. The experimental description is provided in the next section.

EXPERIMENTAL DESCRIPTION

A cutaway view of the experimental combustor is shown in Figure 2. Combustion facilities at the Federal Energy Technology Center (FETC) can supply up to 1.4 kg/s (3 lbm/s) of unvitiated inlet air at temperatures up to 840 K (1050 F) and pressures up to 30 atmospheres. The experiment described here is limited to pressures up to 10 atmospheres by certain components in the pressure vessel design. The facilities include gas sampling capabilities for major species such as NO_x, CO, and UHC. (See Halow et al. 1994 for a complete description of the facility capabilities.)

Preheated combustion air enters the combustor inlet plenum and is diffused by a series of three perforated plates (not shown) located at the left end of the plenum. Air passes through the premixing fuel nozzle and into the combustion zone. The combustion zone is enclosed by a water-cooled liner with 19.8 cm (7.80 in) inside diameter. The water-cooled liner is fitted with a removable refractory plug. The plug inside diameter of 10.4 cm (4.10 in) forms an exhaust "neck" that provides an acoustic response. The plug length is 22.9 cm

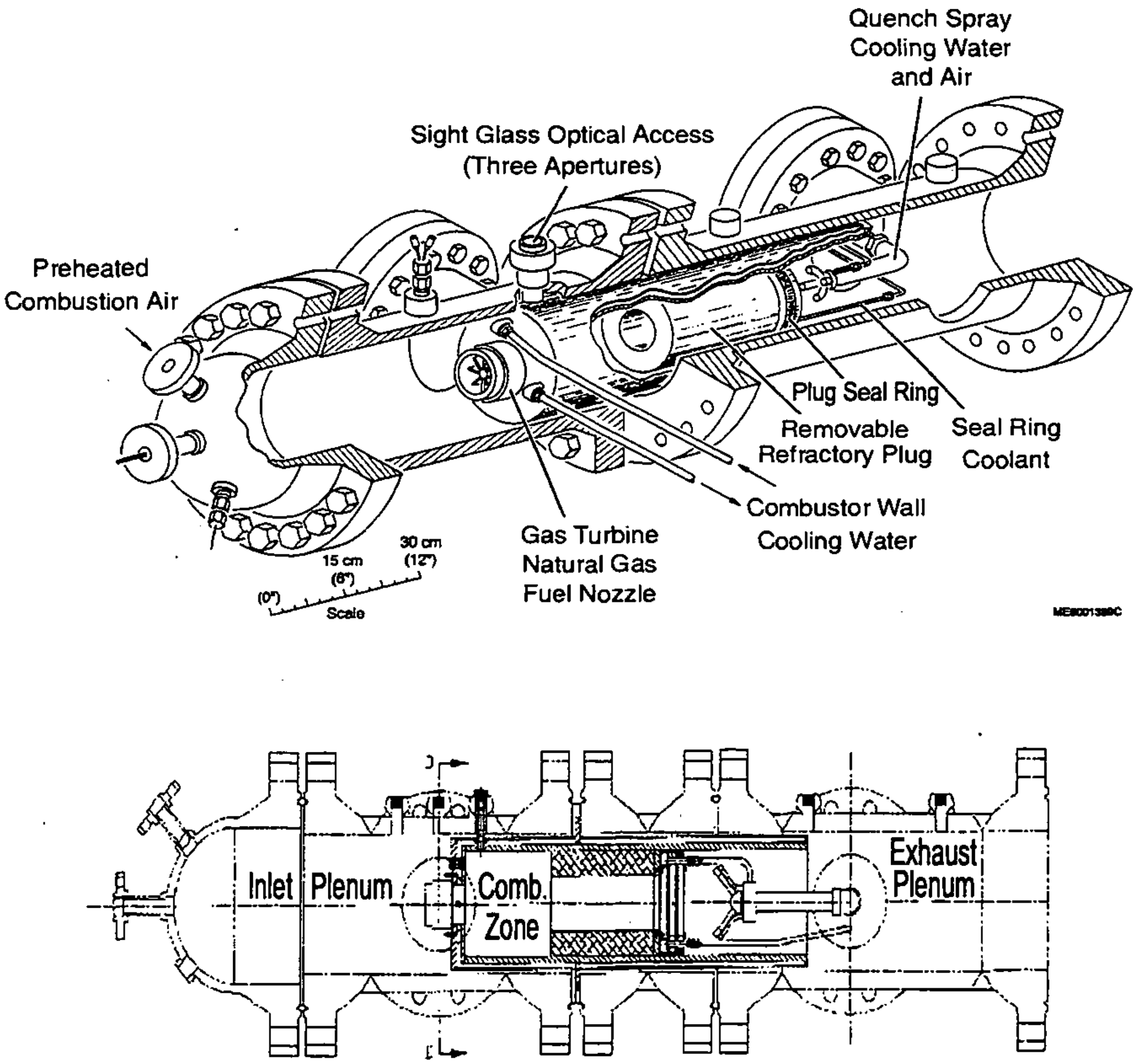


FIGURE 2. CUTAWAY AND CROSS-SECTION VIEW OF THE EXPERIMENTAL COMBUSTOR.

(9.00 in). The combination of the combustion zone volume and the mass of gas in the exhaust neck approximates a classic acoustic Helmholtz resonator. As described by Richards et al. (1996), the water-cooled walls and the refractory plug form a "hard" acoustic boundary. The use of hard acoustic boundaries (as opposed to an air-cooled liner) provides appreciable acoustic feedback, and therefore promotes the occurrence of oscillations near the natural frequencies of the systems. The rationale for the acoustic design of this test combustor is described in Richards et al. (1996). After the refractory plug, cooling water is sprayed into the exhaust stream to quench the combustion products that exhaust through a back-pressure control valve.

The combustor dynamic pressure was recorded with a Kistler Model 206 pressure transducer mounted *outside* the pressure vessel. This avoids the need to locate the transducer in a high-temperature environment. To prevent signal distortion by the connecting tube, we used the transducer connection described by Mahan and Karchmer (1991) and Englund and Richards (1984). The transducer was mounted on a small side branch of 4.8 mm (0.188 in) ID tube connected to the combustor interior. The side branch was located 61 cm

(24 in) from the combustor interior, with 10 m (33 ft) of terminating tubing attached beyond the point of the transducer side-branch. In other words, the transducer was mounted on a small side-branch of a very long tube. The long tube approximates an infinite waveguide, so that pressure signals are transmitted down the waveguide but dissipated by the length of tubing beyond the side-branch. Mahan and Karchmer (1991) show that this approach to signal measurement produces little signal distortion, and typically less than 1 dB attenuation for frequencies below 1000 Hz.

The oscillating heat release was recorded using a fiber optic probe to view the emission from the electronically excited hydroxyl radical (OH*). For this radical, the radiative decay to ground state occurs approximately one microsecond after formation during combustion reactions (Gaydon 1974). Because the excited state exists only momentarily during the combustion reaction, radiative emission (310 nm) is a measure of the combustion heat release. Many studies have shown that OH* emission is proportional to heat release; see Mehta et al. (1981), Keller and Saito (1987), Keller et al. (1990), and Samaniego et al. (1995) for more details. Here, we are concerned only with the phase of the heat release relative to the pressure, so no attempt is made to calibrate the OH* signal to the actual heat release. The fiber optic probe used here was made from a sapphire rod (1.6 mm) (0.063 in) inserted through the pressure vessel in a standard compression fitting using graphite ferrules. (See Richards et al. 1996 for details of the probe construction.)

The probe was located 2.54 cm (1 in) downstream of the fuel nozzle exit plane, viewing across the combustor diameter in a conic region with a 15-degree included angle. This viewing angle does not record emissions from the entire flame region, but was taken as a representative measure of the fluctuating heat release near the fuel nozzle. The ends of the sapphire rod were polished to allow coupling to a UV-transmissive fiber optic cable. The cable was terminated at the focal point of a UV transmissive lens, which allowed beam expansion to 25.4 mm (1 in). An interference filter (310 nm) and photomultiplier tube (PMT) are used to record the signal from the fiber optic probe. The PMT amplifier response was essentially linear over the range of frequencies of interest (less than 2 kHz).

Both the OH* and pressure signals were recorded using a TEAC RD-135T digital tape recorder with a sampling rate of 12 kHz. Signals were analyzed for frequency content, correlation, and root-mean-square (RMS) values using standard signal analysis software available on the TEAC recorder, or on a desktop personal computer.

The fuel nozzle was designed to quickly investigate geometric changes in the nozzle design. The assembled nozzle is shown in Figure 3. The nozzle consists of a series of rings stacked on three threaded rods as shown. The nozzle centerbody is cantilevered from a support ring attached to the end of the threaded rods. Up to the maximum centerbody length, the threaded rods can be fitted with various spacer rings to study the effect of different nozzle geometries. Pilot fuel is injected through the centerbody on the nozzle axis. Pilot fuel was used for ignition only, and not during the combustion tests reported here. Combustion air passes through the annular region between the centerbody and surrounding rings. Eleven straight-blade vanes generate swirl in the nozzle annulus and provide mechanical support for the centerbody. The vanes are set at 45 degrees to the nozzle axis, and are 0.32 cm (0.13 in) thick. The main fuel is injected by eight fuel spokes manufactured from standard 0.32 cm (1/8 in) stainless steel tubing. Fuel is distributed to each spoke by an annular passage inside the ring. Each fuel spoke has six 0.066 cm (0.022 in) holes perpendicular to the axis of the nozzle. These holes were sized to insure that the fuel supply would be choked at the point of fuel injection during most operating conditions. Thus, oscillations produced by this fuel nozzle are not the result of fuel system acoustics.

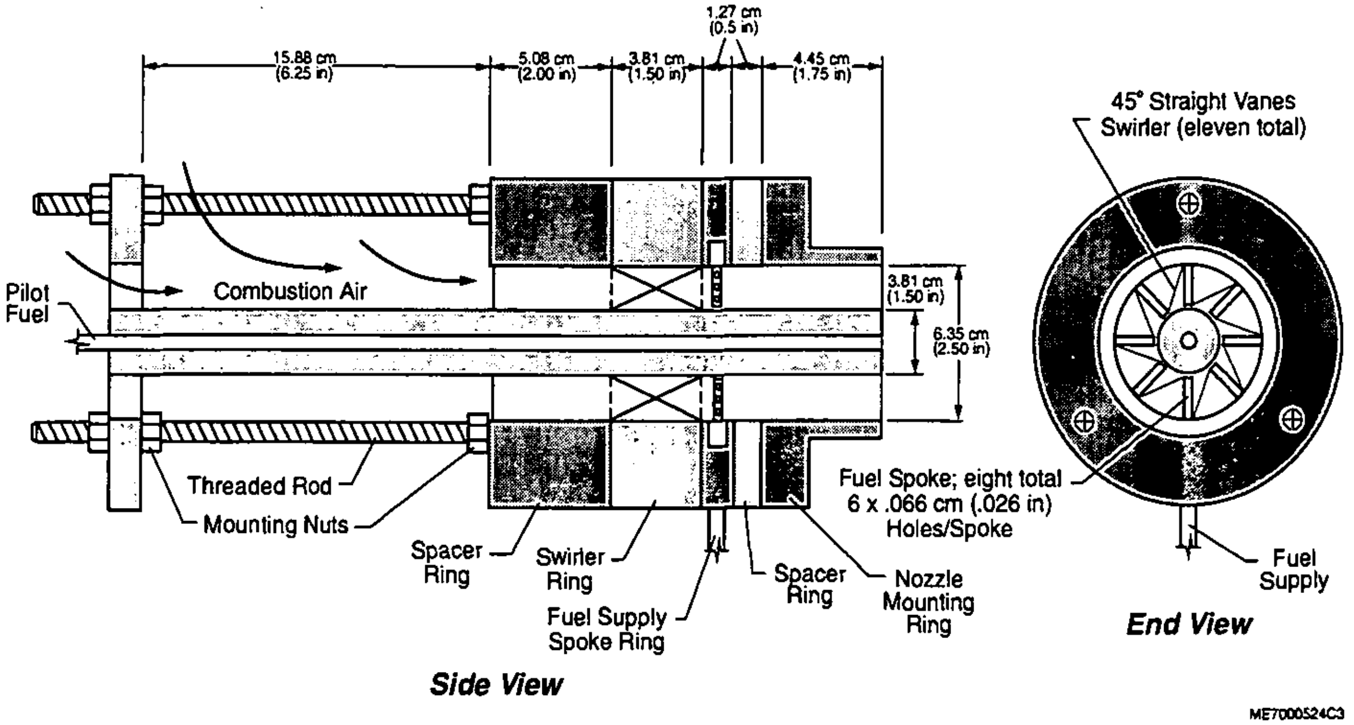
We point out that practical fuel nozzles are typically not choked, and therefore fuel system acoustics may be a consideration in combustion oscillations. However, for development purposes, it is possible to choke the fuel supply and to characterize other oscillating mechanisms as done here. Comparison to the *unchoked* nozzle behavior will then help identify the contribution of fuel system acoustics versus other mechanisms. As explained earlier, the combined effect of various mechanisms may complicate data interpretation, and this complication was avoided by choking the fuel supply.

The fuel (natural gas) was supplied from the local gas company, and pumped to 4.1 MPa (600 psig) prior to metering and injection to the fuel nozzle. The high fuel pressure allowed choking the fuel nozzle even at relatively low fuel flow rates, but still allowing operation at higher flow rates. The (nominal) fuel composition was 94% CH_4, 4% C_2H_6, 1% C_3H_8, and 1% inerts and higher hydrocarbons. Equivalence ratios reported here were established with a precision of better than plus/minus 0.01 at each operating condition. Fuel and air flow rates are metered by standard orifice runs. The absolute accuracy of all flow loops was better than 97% compared to flow-proving performed with sonic nozzle flow meters.

Shih et al. (1996) have recently shown that the level of premixing is an important consideration in characterizing combustion oscillations. We recognized that tests conducted with inadequate premixing would not be representative of LPM combustor design. To ensure that the nozzle described in Figure 3 produced a uniform flow of premixed fuel and air, we conducted a series of cold-flow studies using CO_2 as a tracer gas. Using the same nozzle velocity as in the hot-flow tests, CO_2 was injected at the same volume flow rate as fuel used in the actual hot flow tests. A sampling probe made from 3.2 mm (0.125 in) tubing was then used to traverse the exit plane of the nozzle. The measured CO_2 concentration was essentially flat across the nozzle face, indicating that premixing was complete. We mention that an initial nozzle design incorporated vanes which were half the length shown in Figure 3. The cold-flow premixing studies showed that the shorter vanes did not produce a uniform velocity downstream of the vanes, and the resulting CO_2 concentration was not uniform. In contrast, the longer vanes used here produced very uniform premixing.

Before studying the oscillating characteristic of the fuel nozzle, a limited number of combustion test points were sampled for CO and NO_x emissions. These data are presented as further validation that the fuel nozzle was indeed providing excellent premixing. Emissions were measured in the exhaust stack, after the combustion gases are quenched by the spray cooling water. Stack temperature established by quenching was maintained at less than 477 K (400 F) to minimize any post-quench reactions. The sampling probe (not shown in Figure 2) was located more than 3 m (10 ft) from the quench point, but prior to the backpressure control valve. The sampling probe is a 6.4 mm (0.25 in) tube placed across the diameter of the exhaust stack, with multiple holes drilled along the length of the probe to ensure that the extracted sample is a spatial average of the stack gases. Measurements are reported on a dry basis corrected to 15% oxygen after assuring mass closure from the flow loops versus the measured CO_2 concentration. Figure 4 is a plot of NO_x and CO emissions as a function of equivalence ratio. The reported emissions are typical of premixed burner designs (Lovett and Mick 1995), and as expected,

Fuel Nozzle



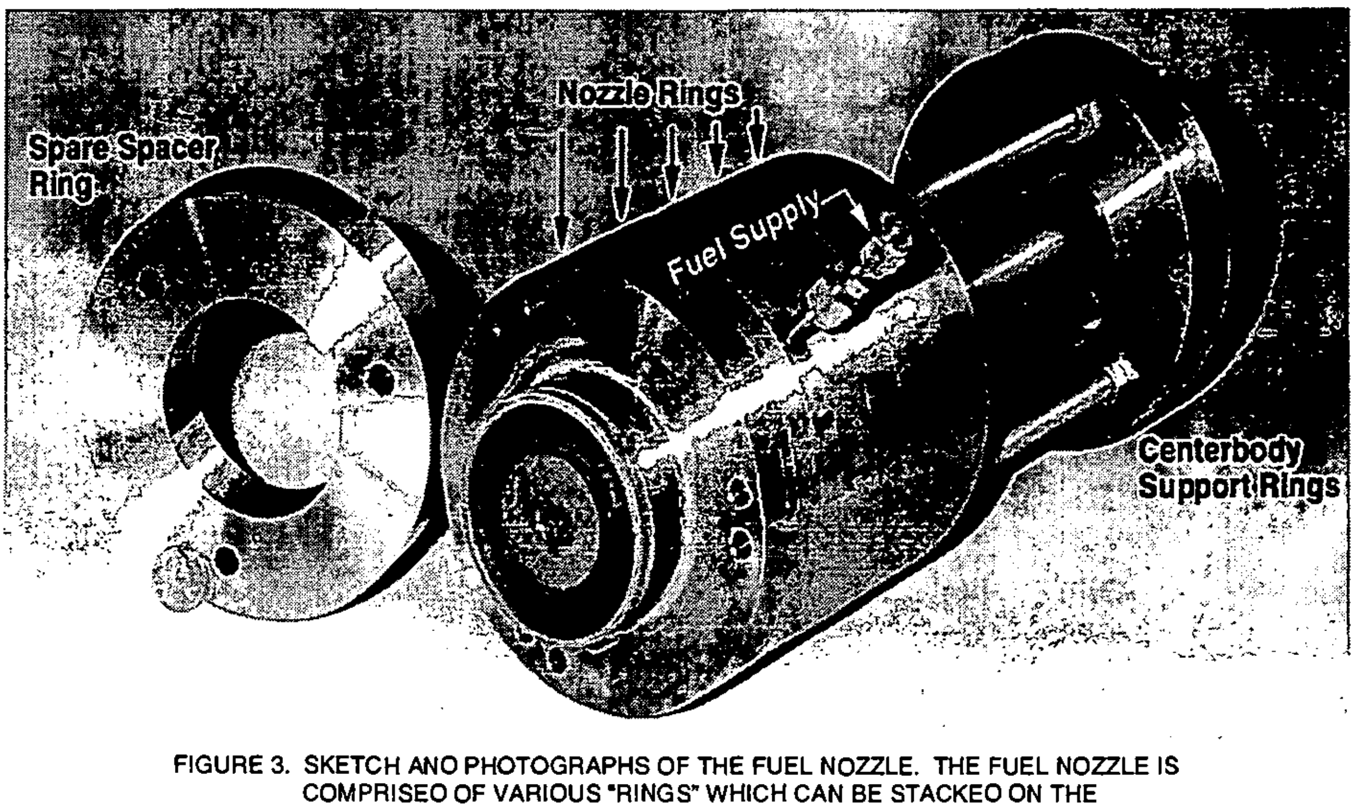


FIGURE 3. SKETCH ANO PHOTOGRAPHS OF THE FUEL NOZZLE. THE FUEL NOZZLE IS COMPRISEO OF VARIOUS "RINGS" WHICH CAN BE STACKEO ON THE THREADEO RODS TO PROOUCE OIFFERENT NOZZLE GEOMETRIES. ELEVEN STRAIGHT-BLADE VANES (SET AT 45 OEGREES) PROVIOE SWIRL.

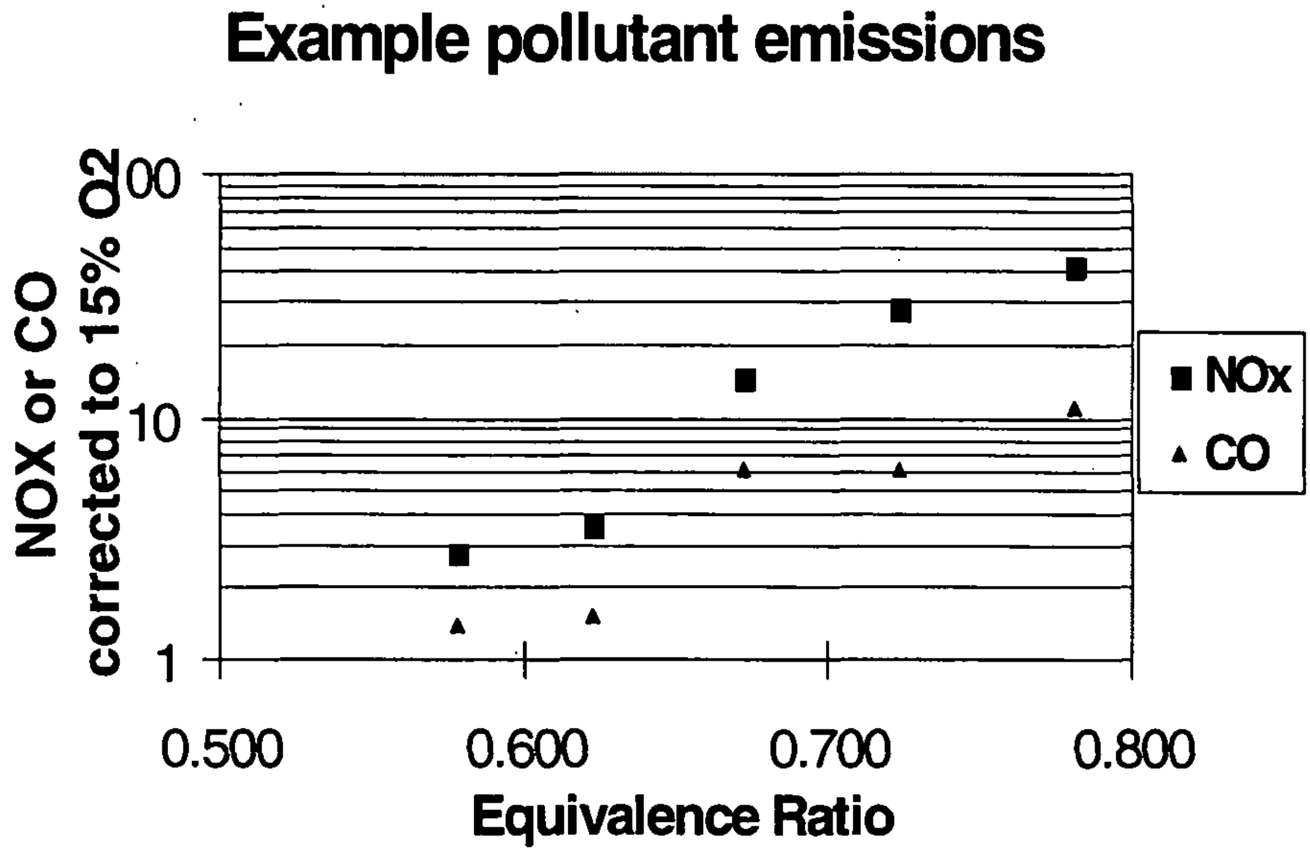


FIGURE 4. NO_X AND CO EMISSIONS (PPMV, CORRECTED TO 15% O_2, DRY BASIS) AS A FUNCTION OF EQUIVALENCE RATIO. EXPERIMENTAL CONDITIONS ARE 5 ATMOSPHERES PRESSURE, 533 K (500 F) INLET AIR TEMPERATURE, 40 M/S REFERENCE VELOCITY.

flame blowoff occurred for an equivalence ratio near 0.52. These combustion data affirm the excellent premixing observed in cold-flow studies described above.

EXPERIMENTAL RESULTS

As explained above, oscillating characteristics of the nozzle were recorded as a function of the nozzle reference velocity. The range of various test parameters is shown below. Because of the desire to choke the fuel supply at even relatively low fuel flows, some higher-flow operating conditions were not accessible due to limits on the maximum fuel supply pressure. Likewise, some low air-flow conditions were inaccessible due to the minimum flow requirements of the combustion air preheater. Aside from these limitations, tests were conducted at the following conditions:

Inlet Air Temperature: 533, 589 K (500, 600 F)

Pressure: 5, 7.5, 10 atmospheres

Equivalence Ratio: 0.59, 0.63, 0.67, 0.71, 0.77

Nozzle Velocity: 30, 40, 50, 60 m/s (98, 131, 164, 197 ft/s)

Over this range of parameters, oscillations with frequencies ranging from approximately 190 Hz to 240 Hz were observed, depending upon operating conditions. Figure 5 is an example of the time history of one oscillation. The nozzle velocity is 30 m/s, with other conditions indicated on the figure. The figure shows both the history of the oscillating pressure and the heat-release fluctuation as measured by the flame OH* emission. The pressure signal was phase-shifted to account for the remote transducer location by simply accounting for the speed of sound traveling in the connecting tube.[1] In accordance with Rayleigh's criterion, the pressure and OH* signals are approximately in phase, driving significant oscillations. The "dual-peak" signal in the OH* time history shows that the heat-release fluctuation includes some complex features that we did not investigate further for this paper, but are the subject of current analysis. The spectra corresponding to both pressure and OH* signals show the dominant frequency of 220 Hz, and the harmonic at 440 Hz.

Figure 6 is a plot of the same combustor signals as in Figure 5, but with a higher nozzle velocity (50 m/s). Here, the combustor is essentially steady. Notice the spectra show weak acoustic peaks at

[1] The infinite waveguide transmits the signal from the combustor to the transducer side-branch. The delay for this transmission is simply the length to the side branch, divided by the speed of sound in the tube. We estimated the speed of sound at the average temperature of the combustion air supply because the waveguide tube is bathed by the (heated) air supply over most of its length. Based on these assumptions, the delay is 1.4 ms.

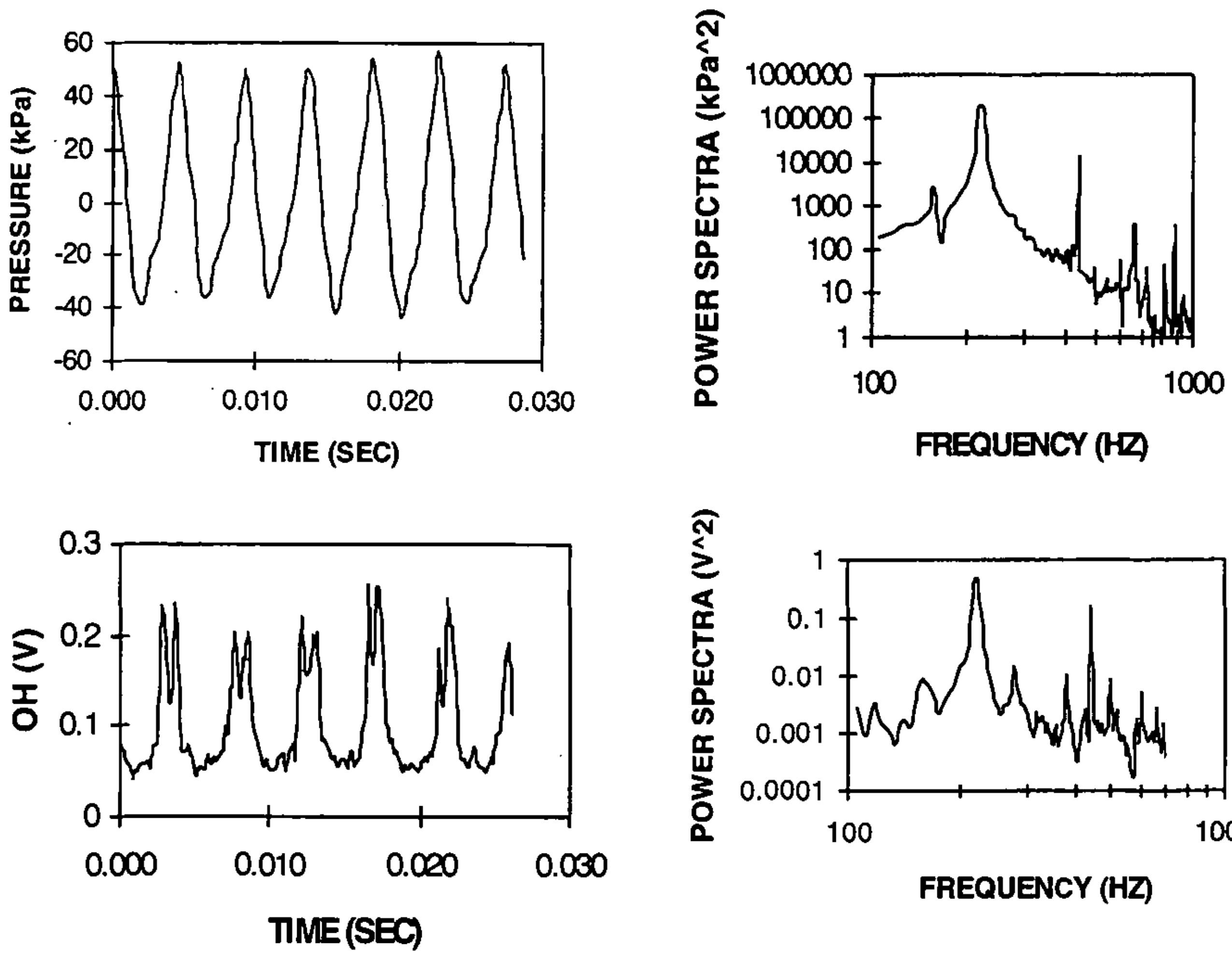


FIGURE 5. EXAMPLE OF OSCILLATION. TESTS CONDITIONS: 7.5 ATMOSPHERES, 533 K INLET TEMPERATURE, EQUIVALENCE RATIO 0.77, (AVERAGE) NOZZLE VELOCITY 30 M/S. THE OH* SIGNAL REPRESENTS THE OSCILLATING HEAT RELEASE. THE SPECTRA SHOW THAT THE DOMINANT FREQUENCY OF OSCILLATION IS 220 HZ AT THIS CONDITION. OTHER OPERATING CONDITIONS PRODUCED APPRECIABLE RMS SIGNALS (I.E., GREATER THAN 0.5% OF THE OPERATING PRESSURE) BETWEEN 180 AND 250 HZ.

150, 180, 215, 225, 269, and 454 Hz. This small signal is the result of excitation of the natural acoustic modes by the air moving through the combustor. A cross-correlation between the pressure and the OH* signal confirmed that the signals were only weakly correlated at this condition, indicating that the combustion was not driving the weak acoustic signal. Compared to Figure 5, these results demonstrate the significant effect of nozzle velocity on stability.

Pressure and OH* signals were recorded over the range of operating conditions. Figures 7 and 8 show the measured root-mean-square (RMS) pressure as a function of nozzle velocity and equivalence ratio. The vertical scale is the normalized RMS pressure expressed as a percent of the operating pressure. Figure 7 shows results for an inlet temperature of 533 K (500 F) at pressures 5, 7.5, and 10 atmospheres. Data points which could not be reached due to limits on the fuel supply pressure are shown as having zero RMS, with an 'x' placed through the data point. Comparing the graphs, note that significant oscillations are confined to nozzle velocities of 30 and 40 m/s, and are most pronounced at the highest equivalence ratios. These same general trends are seen in the data for the 589 K (600 F) inlet temperature shown in Figure 8. In Figure 8, the data points that were not accessible due to the preheater minimum flow are also marked with an 'x'.

We point out that the higher inlet temperature cases do show a modest reduction in RMS pressure along the 40 m/s velocity. In a study of the effect of inlet air temperature, Janus et al. (1996) have shown that changes in inlet temperature should move the stability

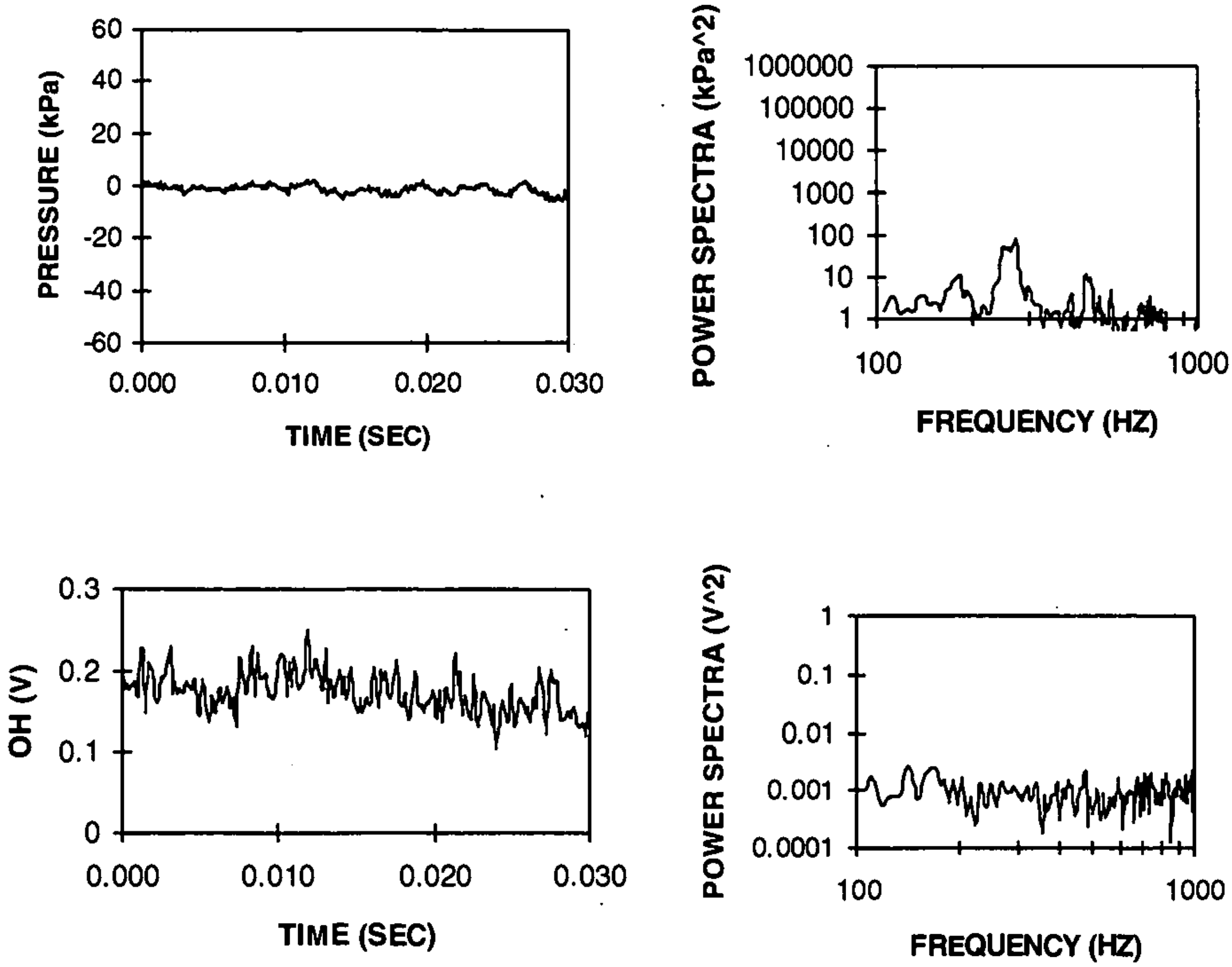


FIGURE 6. COMBUSTOR SIGNALS AT 50 M/S NOZZLE VELOCITY. OTHER CONDITIONS ARE THE SAME AS IN FIGURE 5. COMBUSTION IS STEADY AT THESE CONDITIONS. SPECTRAL PEAKS IN THE PRESSURE SIGNAL ARE DUE TO ACOUSTIC EXCITATION OF RESONANT FREQUENCIES.

boundary to lower values of nozzle velocity. This behavior is approximately evident when comparing Figures 7 and 8, but not very distinct. More data, at a finer spacing of velocity, would be needed to clearly capture this behavior.

Aside from the similarity in the general shape of the instability regions, Figures 7 and 8 also show that the *normalized* RMS pressure is approximately similar at different operating pressures. For example, if we compare the data in Figure 7 at 40 m/s, 0.77 equivalence ratio, the RMS pressure is a constant 5% for operating pressures of 5, 7.5, and 10 atmospheres. Thus, the *actual* RMS scales directly with the operating pressure. This observation is consistent with results from other studies. Based on theoretical considerations, Narayanaswami and Richards (1996) have shown that RMS pressure in an oscillating (pulse) combustor will scale directly with pressure as long as the mixing and chemical reaction rate in the combustor are not appreciably affected by operating pressure. In an experimental study of pressurized pulse combustion, Gemmen et al. (1995) showed that the normalized RMS pressure was only a weak function of operating pressure, and the observed dependence was explained by modest changes in the chemical reaction rate.

Thus, useful estimates of the expected RMS pressure at various operating pressures can be directly scaled from data at other operating pressures. For example, small-amplitude oscillations encountered during low-cost atmospheric pressure tests should not be dismissed during combustor development, because the actual RMS may be unacceptably large during pressurized testing. It is emphasized again that this conclusion depends on the specific mixing and reaction response to changes in operating pressure, and may need to be tested experimentally in specific cases. Nevertheless, the data in Figures 7 and 8 show that in this nozzle (which is typical of LPM combustor

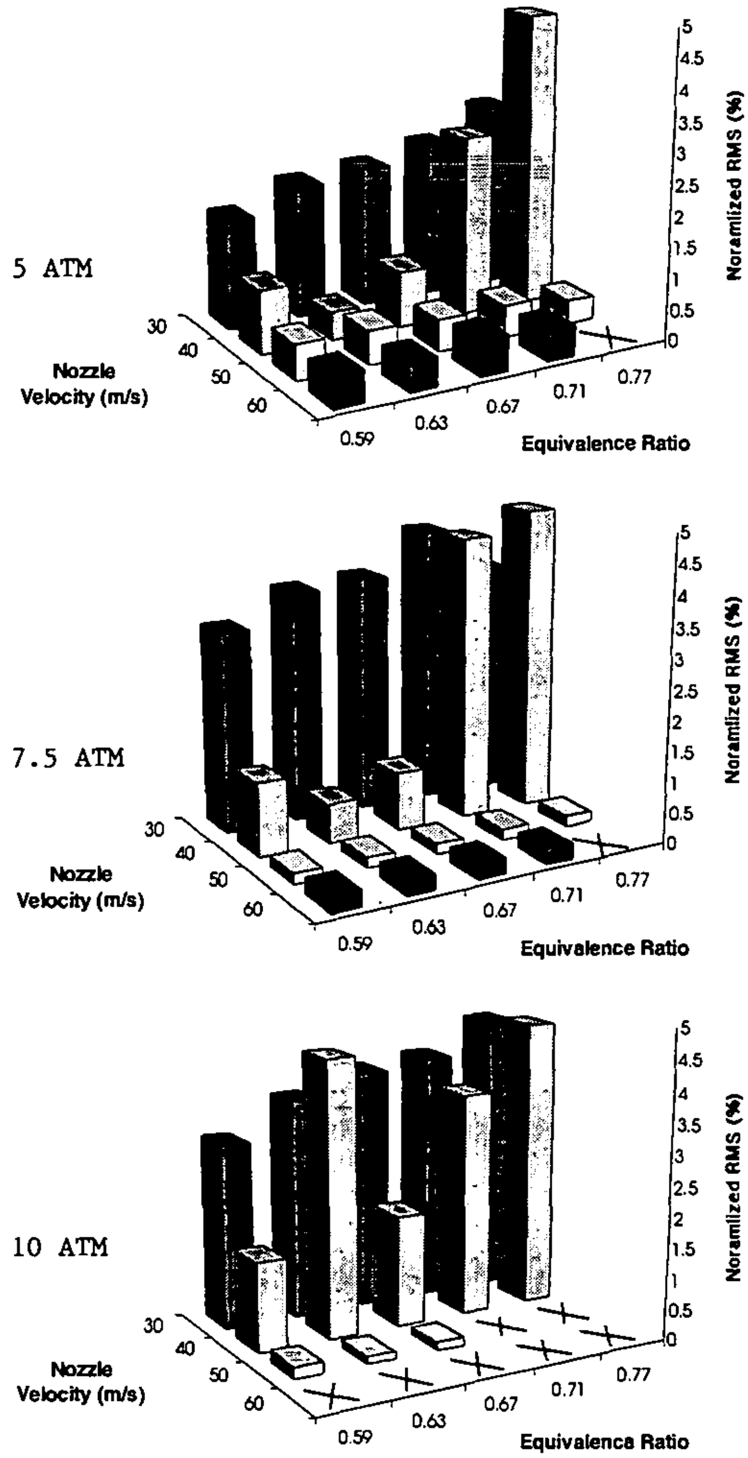


FIGURE 7. ROOT-MEAN-SQUARE (RMS) PRESSURE AS A FUNCTION OF NOZZLE VELOCITY ANO EQUIVALENCE RATIO. THE RMS PRESSURE IS EXPRESSED AS A PERCENTAGE OF THE OPERATING PRESSURE. INLET AIR TEMPERATURE: 533 K. CONOITIONS MARKEO WITH 'X' WERE NOT ACCESSIBLE.

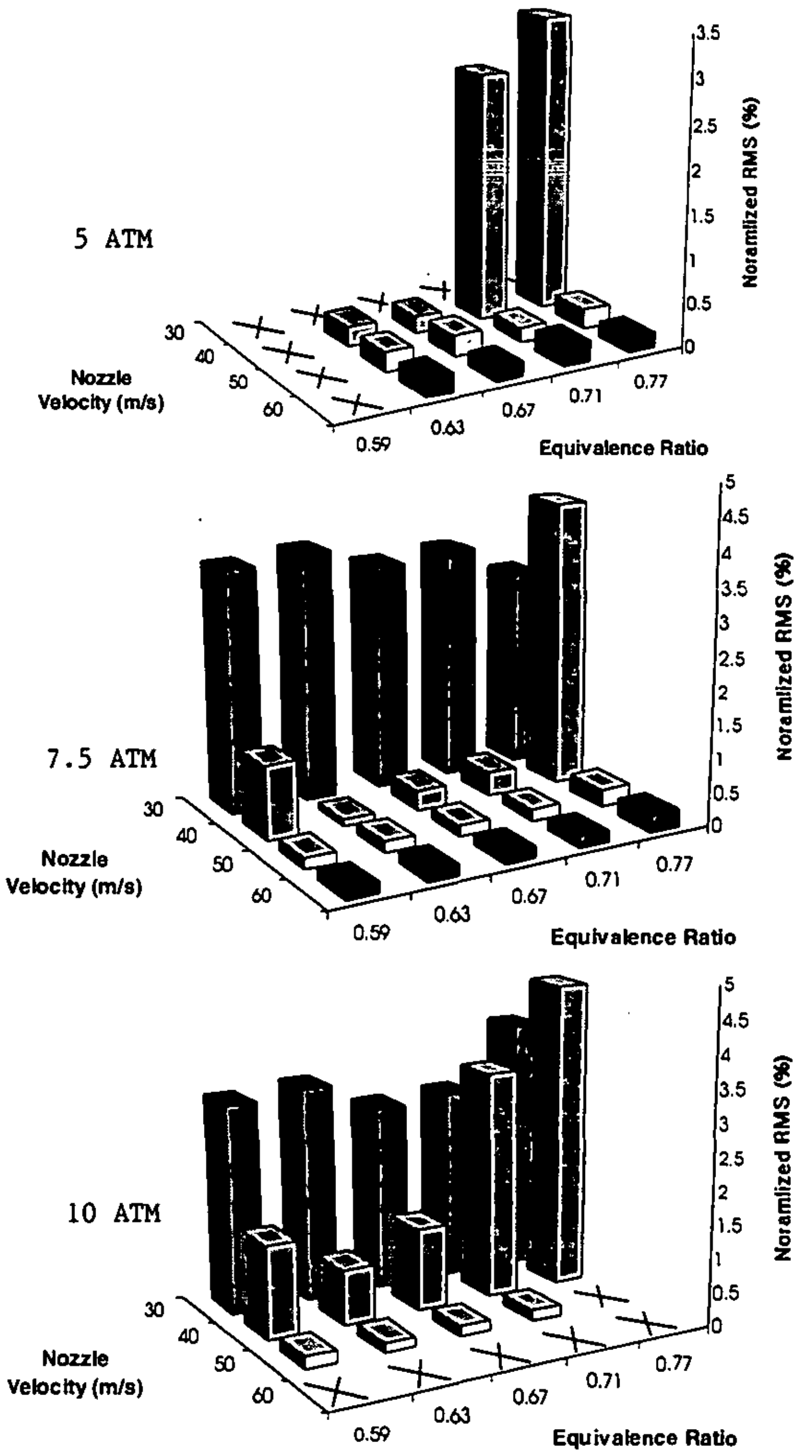


FIGURE 8. ROOT-MEAN-SQUARE (RMS) PRESSURE AS A FUNCTION OF NOZZLE VELOCITY AND EQUIVALENCE RATIO. THE RMS PRESSURE IS EXPRESSED AS A PERCENTAGE OF THE OPERATING PRESSURE. INLET AIR TEMPERATURE: 589 K. CONDITIONS MARKED WITH 'X' WERE NOT ACCESSIBLE.

designs) the RMS can be approximately estimated by scaling from various pressures.

The shape of the instability region for all plots in Figures 7 and 8 shows a surprisingly consistent series of trends, with a few interesting exceptions. It is particularly instructive to compare the behavior of the RMS versus equivalence ratio at a nozzle velocity of 40 m/s. Generally, the RMS reduces to a minimum at equivalence ratio 0.63, but a notable exception occurs for the 10 atmosphere 533 K case (Figure 7). Here, the RMS is very large, almost 5%. Why is this point anomalous? A careful analysis of the time history at this operating condition revealed that the limit cycle oscillation was unstable, and would intermittently alternate between low and high RMS values. A plot of the combustor pressure during several of these transitions is shown in Figure 9. Notice that the oscillation amplitude cycles at a very low frequency, so that the measured RMS plotted in Figure 7 is really not a meaningful characterization of the oscillation. This behavior was observed during several tests occurring on different days at this operating point, confirming that the results were not a peculiar experimental artifact occurring during a single test.

This irregular oscillation apparently exists right at the boundary between steady and oscillating combustion. Knoop et al. (1996) have recently reported control of so-called hysteresis boundaries in step-stabilized flame oscillations. These hysteresis boundaries are difficult to characterize because the oscillating amplitude may depend on the history of the combustor operation, so that slight disturbances can either stabilize or activate oscillations. Although it is not clear that conditions associated with Figure 9 constitute a formal hysteresis boundary,[2] the irregular behavior of the combustor provides a caution that unwitting operation near the oscillation boundary can lead to significant problems. Very slight changes in nozzle velocity (due to machining errors in new engines, or combustor seal wear in older engines) can move the combustor from stable to oscillating combustion. These types of boundaries should be identified in new combustor testing, with appropriate nozzle changes to move the nozzle velocity away from the irregular condition. Plots of combustor dynamics, as shown in Figures 7 and 8, are useful for this purpose.

Oscillations produced by a simple time lag should obey equation (3). As already stated, we cannot precisely determine the flame standoff distance L' and we cannot accurately account for the change in L' as operating conditions change. Recognizing these limitations, Figure 10 is a plot of equation (3) for all the data presented in Figures 7 and 8. We estimated that the flame standoff was constant and equal to 2.5 cm (1.0 in), and simply plotted all the RMS data as a function of the nozzle velocity and observed peak in the frequency spectra. The figure shows that significant oscillations were indeed confined to a small range of the product (time lag)x(frequency). When this product is in the range 0.45 to 0.70, oscillations were observed. Outside this range, the RMS pressure was very small.

Figure 10 suggests that the oscillations observed here are indeed described by a simple time lag. When the time lag was the correct fraction of the acoustic period, oscillations occurred. The time lag was changed by studying the effect of the nozzle velocity on stability. The time lag can also be changed by increasing L, the length of the fuel nozzle. According to (3), if we increase L, then oscillations given by a specific constant will occur at a higher velocity. For example, we showed in Figure 10 that the stability boundary occurred when the product of frequency and transport time was 0.45. If we denote our original nozzle geometry with subscript "a", and a second geometry with subscript "b", then:

$$f_a\,(L_a + L')/U_{avg,a} = 0.45, \tag{4}$$

and for a second nozzle geometry "b" we also have:

$$f_b(L_b + L')/U_{avg,b} = 0.45 \tag{5}$$

If we assume that the oscillating frequency is not changed appreciably between the two nozzles, then along the stability boundary:

$$U_{avg,b} = U_{avg,a}\,(L_b + L')/(L_a + L') \tag{6}$$

Thus, if we lengthen the nozzle, we should see the boundary between stable and oscillating combustion move to a higher velocity.

We tested this hypothesis experimentally. We added a spacer ring to our original fuel nozzle geometry (Figure 3) so that L was extended by 1.9 cm (0.75 in). Again estimating L' = 2.5 cm (1.0 in), equation (6) predicts that the stability boundary at 40 m/s would move to 52 m/s. To confirm this prediction, we repeated some of the tests shown in Figure 7.

Figure 11 is a plot of the same operating conditions as in Figure 7, at 7.5 atmospheres pressure. Figure 11 shows that the oscillating boundary has moved from approximately 40 to 50 m/s nozzle velocity, as predicted by equation (6). Also note that some of the data at 30 m/s has a small amplitude. For this nozzle length, the product of frequency and transport time at 30 m/s was 0.72. From Figure 10, this is again the boundary between stable and oscillating combustion, and we would expect that oscillations should be weak, or irregular.

For practical application, these results show the value of describing the oscillations with a time lag model. Reasonable predictions of the effect of changing nozzle geometry can be made after gathering basic information on the oscillating behavior of the fuel nozzle. If plots such as Figures 7 and 8 can be gathered from an engine or test stand, it may be possible to move the oscillating region out of the operating map by changing the fuel nozzle length or nozzle velocity.

We caution that many subtleties can complicate this approach. The flame stand-off (L') can change abruptly with relatively minor changes in nozzle geometry or operating conditions. Prior work on the flame standoff distance (Ferguson and Keck 1979) may be qualitatively helpful, but exact predictions of L' are complicated by the recognition that the flame is not a flat surface. The acoustic response of the combustor may include several modes so that various frequency-time lag products need to be considered in defining the stability boundaries. We have also artificially choked the fuel supply in this test combustor, eliminating fuel-feed variation as a possible mechanism for variation in heat release. Recognizing these limitations, the time lag approach provides a good starting point to understand more complex mechanisms and to propose modifications to solve instability problems.

SUMMARY AND FUTURE WORK

In this study, we used a simple time lag model to experimentally characterize combustion oscillations produced by a premix fuel nozzle. The combustor was specially designed to provide an acoustic environment suitable for investigating these oscillations. The fuel nozzle

[2] A hysteresis is identified as a subcritical Hopf bifurcation point. See Strogatz (1994) for a discussion of these issues.

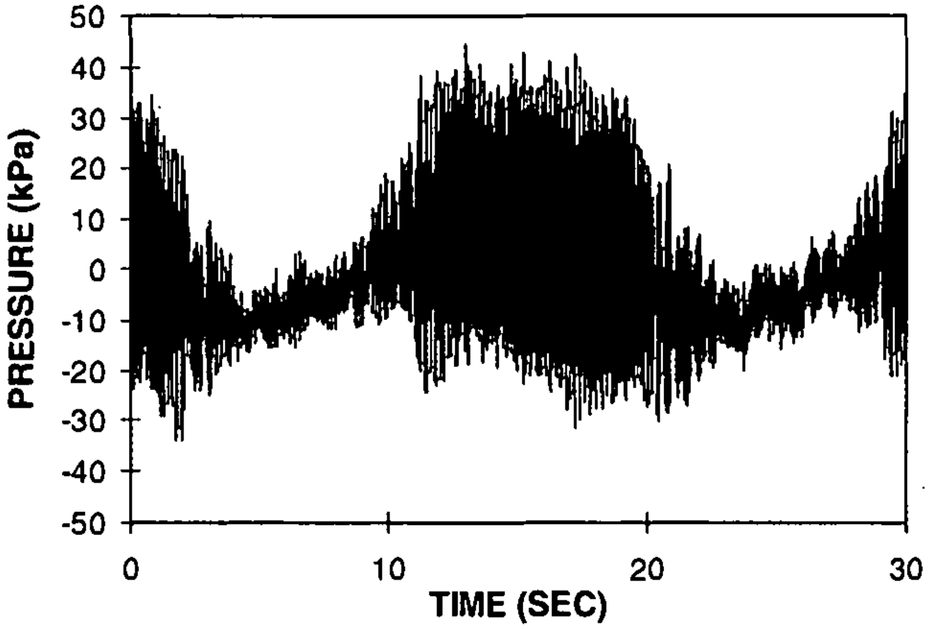


FIGURE 9. TIME HISTORY OF OSCILLATING PRESSURE AT ANOMALOUS CONDITION (40 M/S REFERENCE VELOCITY, 10 ATM, 533 K INLET TEMPERATURE, EQUIVALENCE RATIO 0.63). NOTE THAT THE TIME SCALE IS 30 SECONDS. THE AMPLITUDE OF THE OSCILLATION VARIES SUBSTANTIALLY DURING THIS 30-SECOND PERIOD.

uses a modular design to allow rapid investigation of the effect of nozzle geometry.

The time lag model suggests that the nozzle velocity should play a determining role in oscillations. Thus, oscillation data were gathered as a function of nozzle velocity, for pressures of 5, 7.5, and 10 atmospheres, with inlet air temperatures of 533 K and 589 K (500 F and 600 F). Results showed that the conditions which produced oscillations changed only slightly with pressure and inlet air temperature. For many cases, the oscillation amplitude scaled directly with pressure. This suggests that small oscillations encountered during low-pressure testing may scale to unacceptably large oscillations during higher-pressure operation. At the boundary between stable and oscillating combustion, we observed some intermittent behavior where the oscillation amplitude would rise and fall over several seconds. This irregular behavior emphasizes the need for engineers to identify stability boundaries in new combustors and then ensure that the regular operating conditions are well-removed from the oscillating boundary. Using the approach described here, we were able to characterize the stability boundary from the time lag model and then propose how changes in the nozzle geometry would affect the boundary between oscillating and steady combustion. An experimental test of this approach proved successful for this fuel nozzle and combustor.

We close by noting that the experimental approach used here was tested on just two fuel nozzle geometries. Although the fuel nozzle and operating conditions were representative of current LPM combustor design, more testing is needed to confirm the value of this approach in a wider range of nozzle geometries, with multiple combustor acoustic modes, or with multiple mechanisms driving the variation in heat release. We are presently conducting additional tests using variations on the fuel nozzle geometry (Figure 3) to evaluate these issues.

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REFERENCES

Alsup, C. T., C. M. Zeh, and S. Blazewicz, eds. (1995). *Proceedings of the Advanced Turbine Systems Program Review Meeting.* 2 Vols. DOE/METC-96/1023. NTIS/DE96000561 and NTIS/DE96000562. Springfield, Va.: National Technical Information Service.

Candel, S. M. (1992). Combustion Instabilities Coupled by Pressure Waves and Their Active Control. *The Twenty-Fourth (International) Symposium on Combustion*, Pittsburgh, PA: The Combustion Institute, pp. 1277-1296.

Crocco, L., and S. I. Cheng (1956). Theory of Combustion Instability in Liquid Propellant Rocket Motors. AGARD Monograph, No. 8, Butterworths, London.

Englund, D. R., and W. B. Richards (1984). The Infinite Line Probe. *Proceedings of the 30th International Instrumentation Symposium*, May 7-10, Denver, CO. Available from the Instrument Society of America, Research Triangle Park, North Carolina, pp. 115-124.

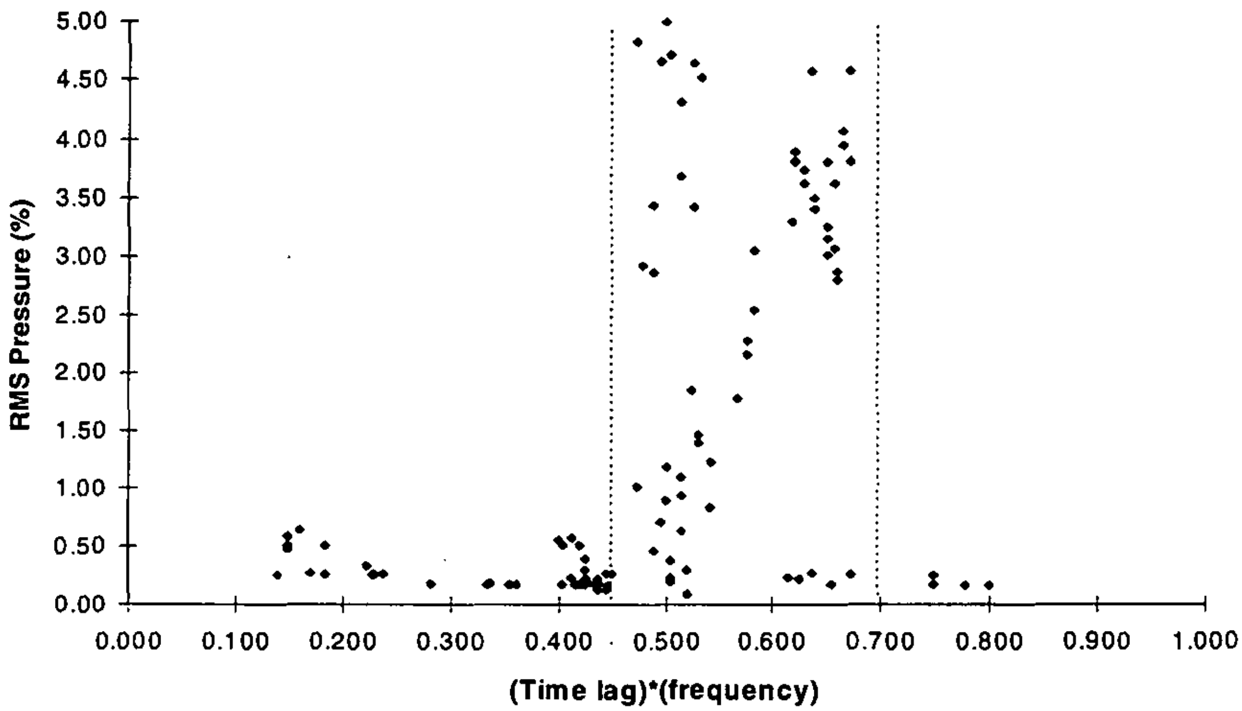

FIGURE 10. PLOT OF RMS PRESSURE VERSUS THE PRODUCT OF TIME LAG AND FREQUENCY. ALL OF THE DATA SHOWN IN FIGURES 7 AND 8 ARE INCLUDED.

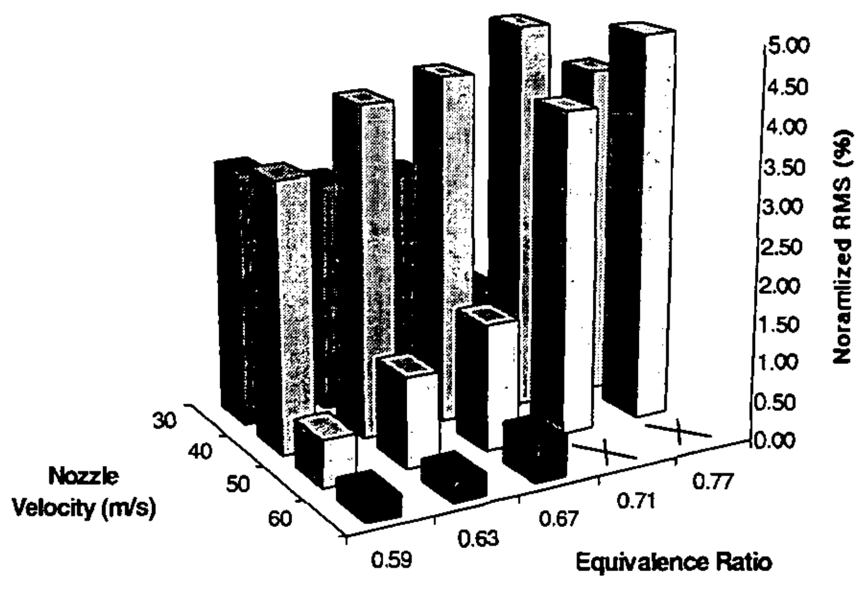

FIGURE 11. RMS PRESSURE FROM LONGER NOZZLE. THE RMS PRESSURE IS EXPRESSED AS A PERCENTAGE OF THE OPERATING PRESSURE. OPERATING CONDITIONS COMPARE DIRECTLY TO DATA FROM FIGURE 7 AT 7.5 ATMOSPHERE PRESSURE. NOTICE THE CHANGE IN THE LOCATION OF THE STABILITY BOUNDARY: OSCILLATIONS ARE ACTIVATED AT 50 M/S IN THIS FIGURE, BUT ARE ABSENT AT 50 M/S IN FIGURE 7.

Feiler, C. E., and M. F. Heidman (1967). Dynamic Response of Gaseous-Hydrogen Flow System and Its Application to High-Frequency Combustion Instability. NASA TN D-4040. Springfield, Va.: National Technical Information Service.

Ferguson, C. R., and J. C. Keck (1979). Stand-off Distances on a Flat Flame Burner. *Combustion and Flame*, Vol. 34, pp. 85-98.

Gaydon, A. G. (1974). The Spectroscopy of Flames. Chapman and Hall, London.

Gemmen, R. S., M. C. Janus, G. A. Richards, T. S. Norton, and W. A. Rogers (1995). Achieving Improved Cycle Efficiency Via Pressure Gain Combustors. ASME Paper 95-GT-63. Presented at the IGTI/ASME Gas Turbine and Aeroengine Congress and Exposition, June 5-8, Houston, TX.

Halow, J. S., D. J. Maloney, and G. A. Richards (1994). METC Combustion Research Facility. *Proceedings of the Advanced Turbine Systems Annual Program Review Meeting*, DOE/OR-2025.

Harrje, D. T., and F. H. Reardon (1972). Liquid Propellant Rocket Combustion Instability. NASA SP-194.

Janus, M. C., G. A. Richards, M. J. Yip, and E. H. Robey (1996). The Effect of Ambient Conditions and Fuel Composition On Combustion Stability. Presented at the IGTI/ASME Gas Turbine and Aeroengine Congress and Exposition, June 2-5, Orlando, FL.

Keller, J. J. (1995). Thermoacoustic Oscillations in Combustion Chambers of Gas Turbines. *AIAA Journal*, Vol. 33, No. 12, pp. 2280-2287.

Keller, J. O., and K. Saito (1987). Measurement of the Combusting Flow in a Pulse Combustor. *Combustion Science and Technology*, Vol. 53, pp. 137-163.

Keller, J. O., T. T. Bramlette, C. K. Westbrook, and J. E. Dec (1990). Pulse Combustion: The Quantification of Characteristic Times. *Combustion and Flame*, Vol. 79, pp. 151-161.

Knoop, P., F. E. C. Culick, and E. E. Zukoski (1996). Extension of the Stability of Motions in a Combustion Chamber by Non-Linear Active Control Based on Hysteresis. To appear, *The Twenty-Sixth (International) Symposium on Combustion*. Pittsburgh, PA: The combustion Institute. Also available: Springfield, Va.: National Technical Information Service, DOE/MC/29061-96/C0701.

Lovett, J. A., and W. J. Mick (1995). Development of a Swirl and Bluff-Body Stabilized Burner for Low-NOx, Lean-Premixed Combustion, ASME Paper 95-GT-166. Presented at the IGTI/ASME Gas Turbine and Aeroengine Congress and Exposition, June 5-8, Houston, TX.

Mahan, J. R., and A. Karchmer (1991). Combustion and Core Noise. in *Aeroacoustics of Flight Vehicles: Theory and Practice. Volume I: Noise Sources, NASA Langley Research Center*. Available from AIAA Technical Library.

Mehta, G. K., M. K. Ramachandra, and W. C. Strahle (1981). Correlations Between Light Emission, Acoustic Emission, and Ion Density in Premixed Turbulent Flames. *The Eighteenth (International) Symposium on Combustion*, Pittsburgh, PA: The Combustion Institute, pp. 1051-1059.

Mehta, J., P. Mungur, W. Dodds, D. Bahr, and S. Clouser (1990a). Fuel Effects on Gas Turbine Combustor Dynamics, AIAA Paper No. 90-1957. Presented at the 26th Joint Propulsion Conference, July 16, Orlando, FL.

Mehta, J., P. Mungur, W. Dodds, D. Bahr, and S. Clouser (1990b). Thermoacoustics of Unsteady Combustion, AIAA Paper No. 90-3928. Presented at the 13th Aeroacoustics Conference, October 22-24, Tallahassee, FL.

Merk, H. J. (1956). An Analysis of Unstable Combustion of Premixed Gases. *Sixth Symposium (International) on Combustion*, Pittsburgh, PA: The Combustion Institute, pp. 500-512.

Narayanaswami, L., and G. A. Richards (1996). Pressure-Gain Combustion: Part 1 - Model Development. *ASME Journal of Engineering for Gas Turbines and Power*, Vol. 118, pp. 461-468.

Putnam, A. A. (1971). *Combustion Driven Oscillations in Industry*. American Elsevier Publishers, New York, NY.

Rayleigh (1878). The Explanation of Certain Acoustical Phenomena. *Royal Institute Proceedings*, Vol. VIII, pp. 536-542.

Richards, G. A., R. S. Gemmen, and M. J. Yip (1996). A Test Device for Premixed Gas Turbine Combustion Oscillations, DOE/METC-96/1027 (DE96004367), Springfield, Va.: National Technical Information Service.

Richards, G. A., and M. J. Yip (1995). Oscillating Combustion from a Premix Fuel Nozzle. Presented at the Combustion Institute/American Flame Research Committee Meeting, April 23-26, San Antonio, TX.

Samaniego, J. M., N. Egolfopoulos, and C. T. Bowman (1995). CO_2 Chemiluminescence in Premixed Flames. *Combustion Science and Technology*, Vol. 109, pp. 183-203.

Schadow, K. C., and E. Gutmark (1991). Combustion Instability Related to Vortex Shedding in Dump Combustors and Their Passive Control. *Progress in Energy and Combustion Science*, Vol. 18, pp. 195-205.

Shih, W.-P., J. Lee, and D. A. Santavicca (1996). Stability and Emission Characteristics of a Lean Premixed Gas Turbine Combustor. To appear, *The Twenty-Sixth (International) Symposium on Combustion*. Pittsburgh, PA: The Combustion Institute.

Sivasegaram, S., and J. H. Whitelaw (1991). The Influence of Swirl on Oscillations in Ducted Premixed Flames. *Combustion and Flame*, Vol. 85, pp. 195-205.

Strogatz, S. H. (1994). Nonlinear Dynamics and Chaos. Addison Wesley Publishing, pp. 215-252.

Yang, V., and W. Anderson, eds. (1995). *Liquid Rocket Combustion Instability*, The American Institute of Aeronautics and Astronautics, Cambridge, MA.