CHAPTER 9

COMBUSTION OF LIQUID PROPELLANTS

The design, development, and operation of liquid rocket engines requires efficient stable burning of the propellants and the generation of a high-temperature, uniform gas that is the rocket's working fluid. In this chapter we treat the complex phenomena of the combustion processes in the combustion chamber of a liquid bipropellant thrust chamber. We describe in general terms the combustion behavior, the progress in analysis of combustion, the several types of combustion instability with its undesirable effects, and semiempirical remedies. The objective is to operate at very high combustion efficiencies and to prevent the occurrence of combustion instability. Thrust chambers should operate with stable combustion over a wide range of operating conditions. For a treatment of these subjects see Refs. 9–1 to 9–3.

The combustion of liquid propellants is very efficient in well-designed thrust chambers, precombustion chambers, or gas generators. Efficiencies of 95 to 99.5% are typical compared to turbojets or furnaces, which can range from 50 to 97%. This is due to the very high reaction rates at the high combustion temperatures and the thorough mixing of fuel and oxidizer reaction species by means of good injection distribution and gas turbulence. The losses are due to incomplete burning or inadequate mixing (nonuniform mixing ratio). For very small bipropellant thrust chambers, where the injector has very few injection orifices or elements, the combustion efficiency can be well below 95%.

9.1. COMBUSTION PROCESS

In describing the combustion processes, it is convenient and helpful to the understanding to divide the combustion chamber into a series of discrete zones, as shown in Fig. 9–1 for a typical configuration. It has a flat injector face with many small injection orifices for introducing both fuel and oxidizer liquids as many discrete individual streams, jets, or thin sprays or sheets. The relative thicknesses of these zones, their behavior, and their transitions are influenced by the specific propellant combination, the operating conditions (pressure, mixture ratio, etc.), the design of the injector, and chamber geometry. The boundaries between the zones shown in Fig. 9–1 are really not flat surfaces and do not display steady flow. They are undulating, dynamically movable, irregular boundaries with localized changes in velocity, temporary bulges, locally intense radiation emissions, or variable temperature. Table 9–1 shows the major interacting physical and chemical processes that occur in the chamber. This table is a modification of tables and data in Refs. 9–2 and 9–3.

The combustion behavior is propellant dependent. If the fuel were hydrogen that has been used to cool the thrust chamber, the hydrogen would be gaseous and fairly warm (60 to 500 K); there would be no liquid hydrogen droplets and no evaporation. With hypergolic propellants there is an initial chemical reaction in the liquid phase when a droplet of fuel impinges on a droplet of oxidizer. Experiments show that the contact can create local explosions and enough energy release to suddenly vaporize a thin layer of the fuel and the oxidizer locally at the droplet's contact face; there immediately follows a vapor chemical reaction and a blow-apart and breakup of the droplets, due to the explosion shock wave pressure (Refs. 9–4 and 9–5).

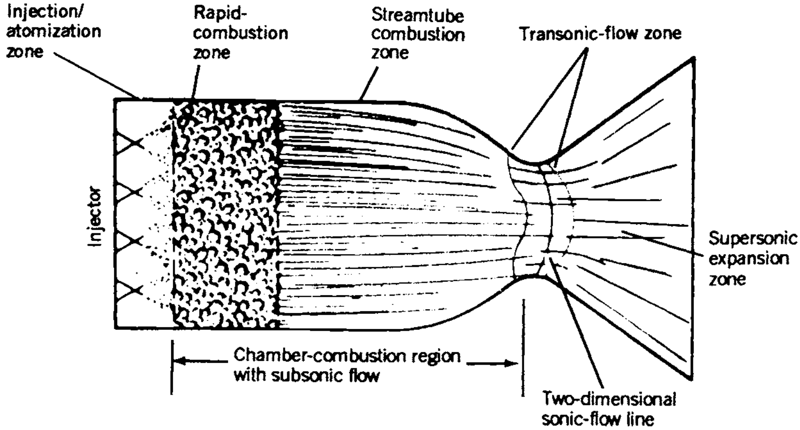

FIGURE 9–1. Division of combustion chamber into zones for analysis. (Reprinted with permission from Ref. 8–1, copyright by AIAA.)

TABLE 9–1. Physical and Chemical Processes in the Combustion of Liquid Propellants

Injection	Atomization	Vaporization
Liquid jets enter chamber at relatively low velocities Sometimes gas propellant is injected Partial evaporation of liquids Interaction of jets and high pressure gas	Impingement of jets or sheets Formation of liquid fans Formation of droplets Secondary breakup of drops Liquid mixing and some liquid–liquid chemical reaction Oscillations of jets or fans as they become unstable during breakup Vaporization begins and some vapor reactions occur	Droplet gasification and diffusion Further heat release from local chemical reactions Low gas velocities and some cross flow Heat absorbed by radiation and conduction from blowback of turbulent gases from the hot reaction zone Acceleration to higher velocities Vaporization rate influenced by pressure or temperature oscillations and acoustic waves

Mixing and Reaction	Expansion in Chamber
Turbulent mixing (three-dimensional) Multiple chemical reactions and major heat releases Interactions of turbulence with droplets and chemical reactions Temperature rise reduces densities Local mixture ratios, reaction rates, or velocities are not uniform across chamber and vary rapidly with time Some tangential and radial flows	Chemical kinetics causes attainment of final combustion temperature and final equilibrium reaction gas composition Gas dynamics displays turbulence and increasing axial gas velocities Formation of a boundary layer Acceleration to high chamber velocities Streamlined high-velocity axial flow with very little cross flow

Rapid Combustion Zone

In this zone intensive and rapid chemical reactions occur at increasingly higher temperature; any remaining liquid droplets are vaporized by convective heating and gas pockets of fuel-rich and fuel-lean gases are mixed. The mixing is aided by local turbulence and diffusion of the gas species.

The further breakdown of the propellant chemicals into intermediate fractions and smaller, simpler chemicals and the oxidation of fuel fractions occur rapidly in this zone. The rate of heat release increases greatly and this causes the specific volume of the gas mixture to increase and the local axial velocity to increase by a factor of 100 or more. The rapid expansion of the heated gases also forces a series of local transverse gas flows from hot high-burning-rate sites to colder low-burning-rate sites. The liquid droplets that may still persist in the upstream portion of this zone do not follow the gas flow quickly and are difficult to move in a transverse direction. Therefore, zones of fuel-rich or oxidizer-rich gases will persist according to the orifice spray pattern in the upstream injection zone. The gas composition and mixture ratio across the chamber section become more uniform as the gases move through this zone, but the mixture never becomes truly uniform. As the reaction product gases are accelerated, they become hotter (due to further heat releases) and the lateral velocities become relatively small compared to the increasing axial velocities.

The combustion process is not a steady flow process. Some people believe that the combustion is locally so intense that it approches localized explosions that create a series of shock waves. When observing any one specific location in the chamber, one finds that there are rapid fluctuations in pressure, temperature, density, mixture ratio, and radiation emissions with time.

Injection/Atomization Zone

Two different liquids are injected with storable propellants and with liquid oxygen/hydrocarbon combinations. They are injected through orifices at velocities typically between 7 and 60 m/sec or about 20 to 200 ft/sec. The injector design has a profound influence on the combustion behavior and some seemingly minor design changes can have a major effect on instability. The pattern, sizes, number, distribution, and types of orifices influence the combustion behavior, as do the pressure drop, manifold geometry, or surface roughness in the injection orifice walls. The individual jets, streams, or sheets break up into droplets by impingement of one jet with another (or with a surface), by the inherent instabilities of liquid sprays, or by the interaction with gases at a different velocity and temperature. In this first zone the liquids are atomized into a large number of small droplets (see Refs. 9–3 and 9–6). Heat is transferred to the droplets by radiation from the very hot rapid combustion zone and by convection from moderately hot gases in the first zone. The droplets evaporate and create local regions rich either in fuel vapor or oxidizer vapor.

This first zone is heterogeneous; it contains liquids and vaporized propellant as well as some burning hot gases. With the liquid being located at discrete sites, there are large gradients in all directions with respect to fuel and oxidizer mass fluxes, mixture ratio, size and dispersion of droplets, or properties of the gaseous medium. Chemical reactions occur in this zone, but the rate of heat generation is relatively low, in part because the liquids and the gases are still relatively cold and in part because vaporization near the droplets causes fuel-rich and fuel-lean regions which do not burn as quickly. Some hot gases from the combustion zone are recirculated back from the rapid combustion zone, and they can create local gas velocities that flow across the injector face. The hot gases, which can flow in unsteady vortexes or turbulence patterns, are essential to the initial evaporation of the liquids.

The injection, atomization and vaporization processes are different if one of the propellants is a gas. For example, this occurs in liquid oxygen with gaseous hydrogen propellant in thrust chambers or precombustion chambers, where liquid hydrogen has absorbed heat from cooling jackets and has been gasified. Hydrogen gas has no droplets and does not evaporate. The gas usually has a much higher injection velocity (above 120 m/sec) than the liquid propellant. This causes shear forces to be imposed on the liquid jets, with more rapid droplet formation and gasification. The preferred injector design for gaseous hydrogen and liquid oxygen is different from the individual jet streams used with storable propellants, as shown in Chapter 8.

Stream Tube Combustion Zone

In this zone oxidation reactions continue, but at a lower rate, and some additional heat is released. However, chemical reactions continue because the mixture tends to be driven toward an equilibrium composition. Since axial velocities are high (200 to 600 m/sec) the transverse convective flow velocities become relatively small. Streamlines are formed and there is relatively little turbulent mixing across streamline boundaries. Locally the flow velocity and the pressure fluctuate somewhat. The residence time in this zone is very short compared to the residence time in the other two zones. The streamline type, inviscid flow, and the chemical reactions toward achieving chemical equilibrium presist not only throughout the remainder of the combustion chamber, but are also extended into the nozzle.

Actually, the major processes do not take place strictly sequentially, but several seem to occur simultaneously in several parts of the chamber. The flame front is not a simple plane surface across the combustion chamber. There is turbulence in the gas flow in all parts of the combustion chamber.

The residence time of the propellant material in the combustion chamber is very short, usually less than 10 milliseconds. Combustion in a liquid rocket engine is very dynamic, with the volumetric heat release being approximately 370 MJ/m^3-sec, which is much higher than in turbojets. Further, the higher temperature in a rocket causes chemical reaction rates to be several times faster (increasing exponentially with temperature) than in turbojet.

9.2. ANALYSIS AND SIMULATION

For the purpose of analysing the combustion process and its instabilities, it has been convenient to divide the acoustical characteristics into linear and nonlinear behavior. A number of computer simulations with linear analyses have been developed over the last 45 years and have been used to understand the combustion process with liquid propellant combustion devices and to predict combustion oscillation frequencies. The nonlinear behavior (for example, why does a disturbance cause an apparently stable combustion to suddenly become unstable?) is not well understood and not properly simulated. Mathematical simulations require a number of assumptions and simplifications to permit feasible solutions (see Refs. 9–1, 9–3, 9–6, and 9–7). Good models exist for relatively simple phenomena such as droplets of a propellant vaporizing and burning in a gaseous atmosphere or the steady-state flow of gases with heat release from chemical reactions. The thermochemical equilibrium principles mentioned in Chapter 5 also apply here. Some programs who consider some turbulence and film cooling effects.

The following phenomena are usually ignored or greatly simplified: cross flows; nonsymmetrical gradients; unsteadiness of the flow; time variations in the local temperature, local velocity, or local gas composition; thermochemical reactions at local off-design mixture ratios and at different kinetic rates; enhancement of vaporization by acoustic fields (see Ref. 9-8); uncertainties in the spatial as well as the size distribution of droplets from sprays; or drag forces on droplets. It requires skilled, experienced personnel to use, interpret, and modify the more complex programs so that meaningful results and conclusions can be obtained. The outputs of these computer programs can give valuable help and confirmation about the particular design and are useful guides in interpreting actual test results, but by themselves they are not sufficient to determine the designs, select specific injector patterns, or predict the ocurrence of combustion instabilities.

All the existing computer programs known to the authors are suitable for steady-state flow conditions, usually at a predetermined average mixture ratio and chamber pressure. However, during the starting, thrust change, and stopping transients, the mixture ratio and the pressure change drastically. The analysis of these transient conditions is more difficult.

The combustion is strongly influenced by the injector design. The following are some of the injection parameters which influence combustion behavior: injector spray or jet pattern; their impingement; hole sizes or hole distribution; droplet evaporation; injection pressure drop; mixture ratio; pressure or temperature gradients near the injector; chamber/injector geometry; initial propellant temperature, and liquid injection pressure drop. Attempts to analyze these effects have met with only partial success.

Computational fluid dynamics (CFD) is a relatively new analytical tool that can provide a comprehensive description of complex fluid dynamic and thermodynamic behavior. It allows for a time history of all parameters and can even include some nonlinear effects. Numerical approaches are used to evaluate sets of equations and models that represent the behavior of the fluid. For complex geometries the information has been tracked with up to 250,000 discrete locations and can include changes in gas composition, thermodynamic conditions, equilibrium reactions, phase changes, viscous or nonviscous flow, one-, two-, or three-dimensional flow, and steady-state or transient conditions. It has been applied to resonance cavities in injectors or chambers and to the flow of burning gases through turbines. A comprehensive rocket combustion model using CFD is not yet available, but could become useful in the future.

9.3. COMBUSTION INSTABILITY

If the process of rocket combustion is not controlled (by proper design), then combustion instabilities can occur which can very quickly cause excessive pressure vibration forces (which may break engine parts) or excessive heat transfer (which may melt thrust chamber parts). The aim is to prevent occurrence of this instability and to maintain reliable operation (see Ref. 9–8). Although much progress has been made in understanding and avoiding combustion instability, new rocket engines can still be plagued by it.

Table 9–2 lists the principal types of combustion vibrations encountered in liquid rocket thrust chambers (see Refs. 9–3 and 9–9). Admittedly, combustion in a liquid rocket is never perfectly smooth; some fluctuations of pressure, temperature, and velocity are always present. When these fluctuations interact with the natural frequencies of the propellant feed system (with and without vehicle structure) or the chamber acoustics, periodic superimposed oscillations, recognized as instability, occur. In normal rocket practice *smooth combustion* occurs when pressure fluctuations during steady operation do not exceed about ±5% of the mean chamber pressure. Combustion that gives greater pressure fluctuations at a chamber wall location which occur at completely random intervals is called *rough combustion.* Unstable combustion, or *combustion instability*, displays organized oscillations occurring at well-defined intervals with a pressure peak that may be maintained, may increase, or may die out. These periodic peaks, representing fairly large concentrations

TABLE 9–2. Principal Types of Combustion Instability

Type and Word Description	Frequency Range (Hz)	Cause Relationship
Low frequency, called chugging or feed system instability	10–400	Linked with pressure interactions between propellant feed system, if not the entire vehicle, and combustion chamber
Intermediate frequency, called acoustic,[a] buzzing, or entropy waves	400–1000	Linked with mechanical vibrations of propulsion structure, injector manifold, flow eddies, fuel/oxidizer ratio fluctuations, and propellant feed system resonances
High frequency, called screaming, screeching, or squealing	Above 1000	Linked with combustion process forces (pressure waves) and chamber acoustical resonance properties

[a]Use of the word *acoustical* stems from the fact the frequency of the oscillations is related to combustion chamber dimensions and velocity of sound in the combustion gas.

of vibratory energy, can be easily recognized against the random-noise background (see Fig. 9–2).

Chugging, the first type of combustion instability listed in Table 9–2, stems mostly from the elastic nature of the feed systems and structures of vehicles or the imposition of propulsion forces upon the vehicle. Chugging of an engine or thrust chamber assembly can occur in a test facility, especially with low chamber pressure engines (100 to 500 psia), because of propellant pump cavitation, gas entrapment in propellant flow, tank pressurization control fluctuations, and vibration of engine supports and propellant lines. It can be caused by resonances in the engine feed system (such as an oscillating bellows inducing a periodic flow fluctuation) or a coupling of structural and feed system frequencies.

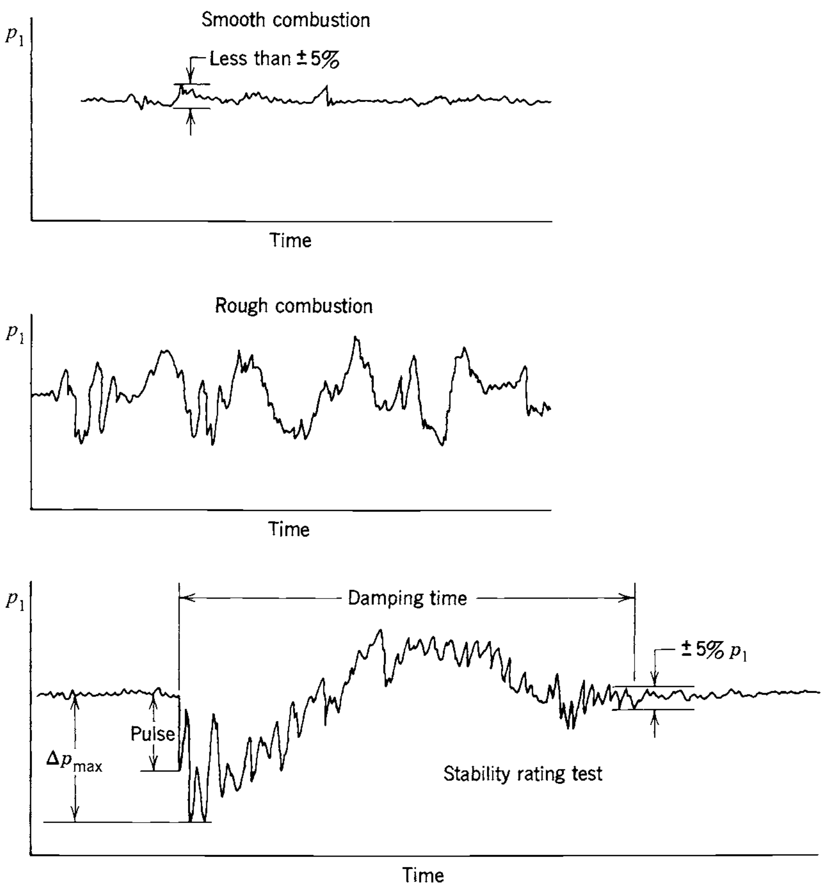

FIGURE 9–2. Typical oscillogrpah traces of chamber pressure p_1 with time for different combustion events.

When both the vehicle structure and the propellant liquid in the feed system have about the same natural frequency, then force coupling can occur, not only to maintain, but also to strongly amplify oscillations. Propellant flow rate disturbances, usually at 10 to 50 Hz, give rise to low-frequency longitudinal combustion instability, producing a longitudinal motion of vibration in the vehicle. This vehicle flight instability phenomenon has been called *pogo instability* since it is similar to pogo jumping stick motion. Pogo instabilities can occur in the propellant feed lines of large vehcles such as space launch vehicles or ballistic missiles.

Avoiding objectionable engine–vehicle coupled oscillation is best accomplished at the time of initial design of the vehicle, as contrasted to applying "fixes" later as has been the case with rocket engines for the Thor, Atlas, and Titan vehicles. Analytical methods exist for understanding the vibration modes and damping tendencies of major vehicle components, including the propellant tanks, tank pressurization systems, propellant flow lines, engines, and basic vehicle structure. Figure 9–3, a simplified spring–mass model of a typical two-stage vehicle, indicates the complexity of the analytical problem. Fortunately, the vibrational characteristics of the assembly can be affected substantially by designing damping into the major components or subassemblies. Techniques for damping pogo instability include the use of energy-absorption devices in fluid flow lines, perforated tank liners, special tank supports, and properly designed engine, interstage, and payload support structures (see Refs. 9–10 and 9–11).

A partially gas-filled pogo accumulator has been an effective damping device; it is attached to the main propellant feed line. Such an accumulator is used in the oxidizer feed line of the Space Shuttle main engine (SSME) betwen the two oxidizer turbopumps; it can be seen in Figs. 6–1 and 6–12. The SSME fuel line does not need such a damping device, because the fuel has a relatively very low density and a lower mass flow.

The dynamic characteristics of a propellant pump can also have an influence on the pogo-type vibrations, as examined in Ref. 9–12. The pogo frequency will change as propellant is consumed and the remaining mass of propellant in the vehicle changes. The bending or flexing of pipes, joints or bellows, or long tanks also has an influence.

Buzzing, the intermediate type of instability, seldom represents pressure perturbations greater than 5% of the mean in the combustion chamber and usually is not accompanied by large vibratory energy. It often is more noisy and annoying than damaging, although the occurrence of buzzing may initiate high-frequency instability. Often it is characteristic of coupling between the combustion process and flow in a portion of the propellant feed system. Initiation is thought to be from the combustion process. Acoustic resonance of the combustion chamber with a critical portion of the propellant flow system, sometimes originating in a pump, promotes continuation of the phenomenon. This type of instability is more prevalent in medium-size engines (2000 to 250,000 N thrust or about 500 to 60,000 lbf) than in large engines.

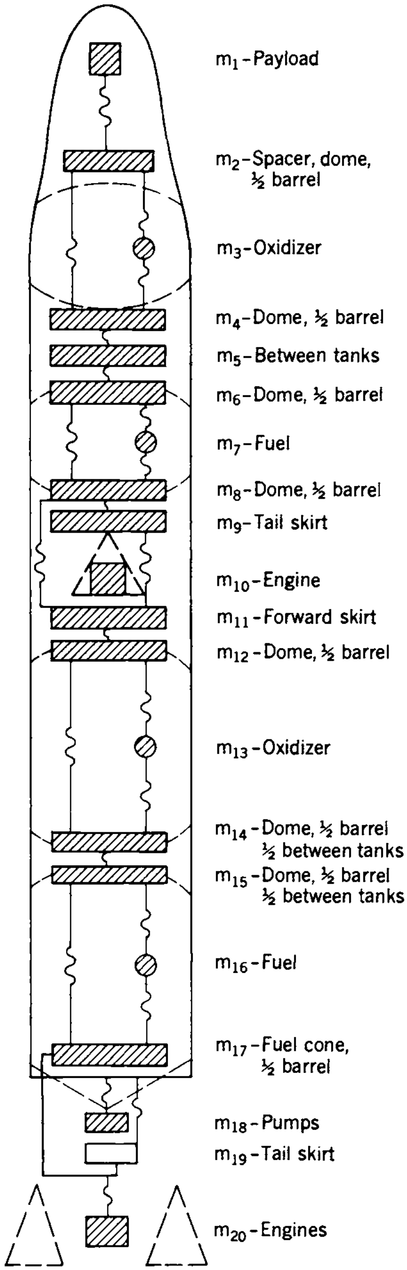

FIGURE 9–3. Typical two-stage vehicle spring–mass model used in analysis of pogo vibration in the veritcal direction.

The third type, *screeching* or *screaming*, has high frequency and is most perplexing and most common in the development of new engines. Both liquid and solid propellant rockets commonly experience high-frequency instablity during their development phase. Since energy content increases with frequency, this type is the most destructive, capable of destroying an engine in much less than 1 sec. Once encountered, it is the type for which it is most difficult to prove that the incorporated "fixes" or improvements render the engine "stable" under all launch and flight conditions. It can be treated as a phenomenon isolated to the combustion chamber and not generally influenced by feed system or structure.

High-frequency instability occurs in at least two modes, *longitudinal* and *transverse*. The *longitudinal mode* (sometimes called *organ pipe mode* propagates along axial planes of the combustion chamber and the pressure waves are reflected at the injector face and the converging nozzle cone. The *transverse modes* propagate along planes perpendicular to the chamber axis and can be broken down into *tangential* and *radial* modes. Transverse mode instability predominates in large liquid rockets, particularly in the vicinity of the injector. Figure 9–4 shows the distribution of pressure at various time intervals in a cylindrical combustion chamber (cross section) encountering transverse mode instability. Two kinds of wave form have been observed for tangential vibrations. One can be considered a *standing* wave that remains fixed in position while its pressure amplitude fluctuates. The second is a *spinning* or *traveling* tangential wave which has associated with it a rotation of the whole vibratory system. This waveform can be visualized as one in which the amplitude remains constant while the wave rotates. Combinations of transverse and longitudinal modes can also occur and their frequency can also be estimated.

Energy that drives screeching is believed to be predominantly from acoustically stimulated variations in droplet vaporization and/or mixing, local detonations, and acoustic changes in combustion rates. Thus, with favorable acoustic properties, high-frequency combustion instability, once triggered, can rapidly drive itself into a destructive mode. Invariable, a distinct boundary layer seems to disappear and heat transfer rates increase by an order of magnitude, much as with detonation, causing metal melting and wall burn-throughs, sometimes within less than 1 sec. The tangential modes appear to be the most damaging, heat transfer rates during instability often increasing 4 to 10 times. Often the instantaneous pressure peaks are about twice as high as with stable operation.

One possible source of triggering high-frequency instability is a rocket combustion phenomenon called *popping*. Popping is an undesirable random high-amplitude pressure disturbance that occurs during steady-state operation of a rocket engine with hypergolic propellants. It is a possible source for initiation of high-frequency instability. "Pops" exhibit some of the characteristics of a detonation wave. The rise time of the pressure is a few microseconds and the pressure ratio across the wave can be as high as 7:1. The elimination of popping

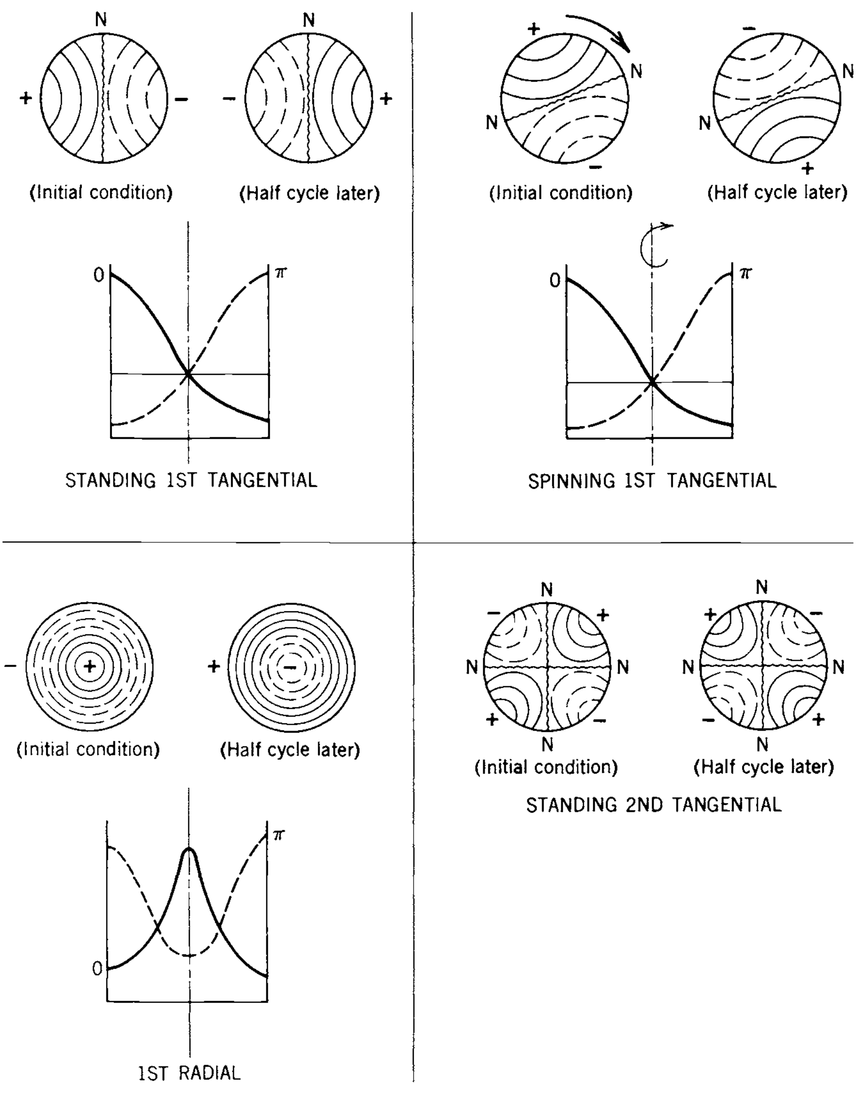

FIGURE 9–4. Simplified representation of transverse pressure oscillation modes at two time intervals in a cylindrical combustion chamber. The solid line curves indicate pressures greater than the normal or mean operating pressure and the dashed lines indicate lower pressures. The N–N lines show the node locations for these wave modes.

is usually achieved by redesign of the injector rather than by the application of baffles or absorbers.

Some combustion instabilities can be induced by *pulsations in the liquid flow* originating in turbopumps. Unsteady liquid flow can be caused by irregular cavitation at the leading edge of the inducer impellers or the main pump impellers. Also, when an impeller's trailing edge passes a rib or stationary vane of the volute, a small pressure perturbation always occurs in the liquid flow that travels downstream to the injector. These two types of pressure fluctuation can be greatly amplified if they coincide with the natural frequency of combustion vibrations in the chamber.

The estimated natural frequencies can be determined from the wavelength l, or the distance traveled per cycle, and the acoustic velocity a (see Eq. 3–10). The frequency, or number of cycles per second, is

$$\text{frequency} = a/l = (1/l)\sqrt{kTR'/\mathfrak{M}} \tag{9–1}$$

where k is the specific heat ratio, R' the universal gas constant, $\mathfrak{M}$ the estimated molecular weight of the hot chamber gases, and T the local average absolute temperature. The length of wave travel depends on the vibrational mode, as shown in Fig. 9–4. Smaller chambers give higher frequencies.

Table 9–3 shows a list of estimated vibration frequencies for the Vulcain HM 60 rocket thrust chamber; it operates with liquid hydrogen and liquid oxygen propellants at a vacuum thrust of 1008 kN, a nominal chamber pressure of 10 MPa, and a nominal mixture ratio of 5.6 (see Ref. 9–13). The data in the table are based on acoustic measurements at ambient conditions with corrections for an appropriate sonic velocity correlation; since the chamber has a shallow conical shape and no discrete converging nozzle section, the purely longitudinal vibration modes would be weak; in fact, no pure longitudinal modes were detected.

Figure 9–5 shows a series of time-sequenced diagrams of frequency–pressure–amplitude measurements taken in the oxygen injector manifold of the Vulcain HM 60 engine during the first 8 sec of a static thrust chamber test while operating at off-nominal design conditions. Chugging can be seen at low

TABLE 9–3. Estimated Acoustic Hot Gas Frequencies for Nominal Chamber Operating Conditions for the Vulcain HM-60 Thrust Chamber

Mode[a]	(L, T, R)	Frequency (Hz)	Mode[a]	(L, T, R)	Frequency (Hz)
T1	(0, 1, 0)	2424	L1T3	(1, 3, 0)	6303
L1T1	(1, 1, 0)	3579	T4	(0, 4, 0)	6719
T2	(0, 2, 0)	3856	L2R1	(2, 0, 1)	7088
R1	(0, 0, 1)	4849	T5	(0, 5, 0)	8035
L1T2	(1, 2, 0)	4987	TR21	(0, 2, 1)	8335
T3	(0, 3, 0)	5264	R2	(0, 0, 2)	8774
L1R1	(1, 0, 1)	5934			

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[a]Modes are classified as L (longitudinal), T (tangential), or R (radial) and the number refers to the first, second, or third natural frequency.

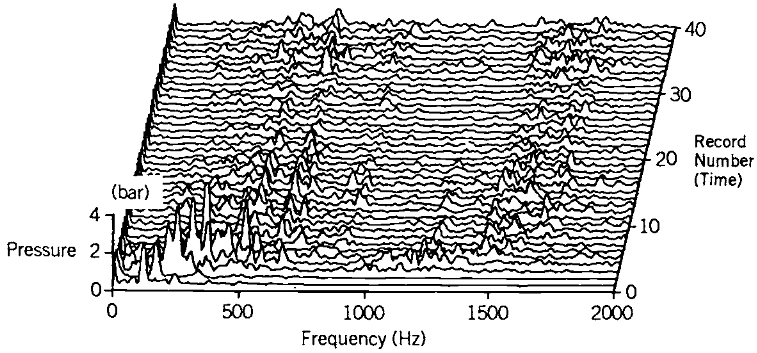

FIGURE 9–5. Graphical representation of a series of 40 superimposed frequency–amplitude diagrams taken 0.200 sec apart during the start phase (for the first 8 sec) of the Vulcain HM 60 thrust chamber. In this static hot-firing test the thrust chamber was operating at 109 bar chamber pressure and an oxidizer-to-fuel mass flow mixture ratio of 6.6. (Copied with permission from Ref. 9–13).

frequency (up to 500 Hz) during the first few seconds and a natural frequency around 1500 Hz is attributed to the natural resonance frequency of the oxygen injector dome structure where the high-frequency pressure transducer was-mounted. The continued oscillations observed at about 500 and 600 Hz are probably resonances associated with the feed system.

Rating Techniques

Semi-empirical techniques exist for artificially disturbing combustion in a rocket thrust chamber during test operation and evaluating its resistance to instability (see Ref. 9–14). These include: (1) nondirectional "bombs" placed within the combustion chamber; (2) oriented explosive pulses from a "pulse gun" directed through the chamber sidewall; and (3) directed flows of inert gas through the sidewall into the chamber. Often heavy prototype thrust chambers are used because they are less expensive and more resistant to damage than flight-weight engines. Other techniques used less widely but which are important, especially for small engines, include: (1) momentary operation at "off-mixture ratio;" (2) introduction of "slugs" of inert gas into a propellant line; and (3) a purposeful "hard start" achieved by introducing a quantity of unreacted propellant at the beginning of the operation.

The objective of these rating techniques is to measure and demonstrate the ability of an engine system to return quickly to normal operation and stable combustion after the combustion process has intentionally been disturbed or perturbed.

All techniques are intended to introduce shock waves into the combustion chamber or to otherwise perturb the combustion process, affording opportunity for measuring recovery time for a predetermined overpressure disturbance, assuming stable combustion resumes. Important to the magnitude and mode of the instability are the type of explosive charge selected, the size of the charge, the location and direction of the charge, and the duration of the exciting pulse. The bottom curve in Fig. 9–2 characterizes the recover of stable operation after a combustion chamber was "bombed." The time interval to recover and the magnitude of explosive or perturbation pressure are then used to rate the resistance of the engine to instability.

The nondirectional bomb method and the explosive pulse-gun method are the two techniques in common use. The bomb that can be used in large flight-weight thrust chambers without modification consists of six 250 grains of explosive powder (PETN,RDX,etc.) encased in a Teflon, nylon, or micarta case. Detonation of the bomb is achieved either electrically or thermally. Although the pulse gun requires modification of a combustion chamber, this technique affords directional control, which is important to tangential modes of high-frequency instability and allows several data points to be observed in a single test run by installing several pulse guns on one combustion chamber. Charges most frequently used are 10, 15, 20, 40, and 80 grains of pistol powder. Pulse guns can be fired in sequence, introducing successive pressure perturbations (approximately 150 msec apart), each of increasing intensity, into the combustion chamber.

Control of Instabilities

The control of instabilities is an important task during the design and development of a rocket engine. The designer usually relies on prior experience with similar engines and tests on new experimental engines. He also has available analytical tools with which to simulate and evaluate the combustion process. The design selection has to be proven in actual experiments to be free of instabilities over a wide range of transient and steady-state operating conditions. Some of the experiments can be accomplished on a subscale rocket thrust chamber that has a similar injector, but most tests have to be done on a full-scale engine.

The design features to control instabilities are different for the three types described in Table 9–2. Chugging is usually avoided if there is no resonance in the propellant feed system and its coupling with the elastic vehicle structure. Increased injection pressure drop and the addition of artificial damping devices in the propellant feed lines have been used successfully. Chugging and acoustical instabilities sometimes relate to the natural frequency of a particular feed system component that is free to oscillate, such as a loop of piping that can vibrate or a bellows whose oscillations cause a pumping effect.

With the choice of the propellant combination usually fixed early in the planning of a new engine, the designer can alter combustion feedback (depressing the driving mechanism) by altering injector details, (such as changing the injector hole pattern, hole sizes or by increasing the injection pressure drop), or alternatively by increasing acoustical damping within the combustion chamber. Of the two methods, the second has been favored in recent years because it is very effective, it is better understood, and theory fits. This leads to the application of *injector face baffles, discrete acoustic energy absorption cavities*, and *combustion chamber liners* or *changes in injector design*, often by using a trial and error approach.

Injector face baffles (see Fig. 9–6) were a widely accepted design practice in the 1960s for overcoming or preventing high-frequency instability. Baffle design is predicated on the assumption that the most severe instability, oscillations, along witht he driving source, are located in or near the injector–atomization zone at the injector end of the combustion chamber. The baffles minimize influential coupling and amplification of gas dynamic forces within the chamber. Obviously, baffles must be strong, have excellent resistance to combustion temperatures (they are usually cooled by propellant), and must protrude into the chamber enough to be effective, yet not so far as to act like an individual combustion chamber with its own acoustical characteristics. The number of baffle compartments is always odd. An even number of compartments enhances the standing modes of instability, with the baffles acting as nodal lines separating regions of relatively high and low pressure. The design and development of baffles remains highly empirical. Generally, baffles are designed to minimize acoustical frequencies below 4000 Hz, since experience has shown damaging instability is rare at frequencies above 4000 Hz.

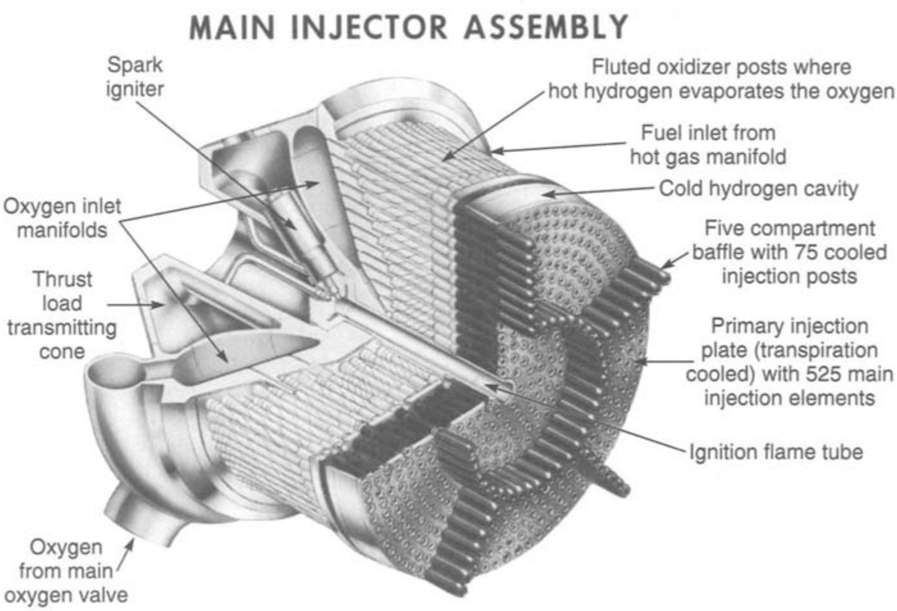

FIGURE 9–6. Main injector assembly of the Space Shuttle main engine showing baffle with five outer compartments. (Courtesy of The Boeing Company, Rocketdyne Propulsion and Power.)

Various mechanisms of *energy absorption* or *vibration damping* exist in a thrust chamber. Damping by well friction in combustion chambers is not significant. The exhaust nozzle produces the main damping of longitudinal mode oscillations; the reflection of waves from the convergent nozzle entrance departs from that of an ideal closed end. The principal damping source affecting propagation in the transverse plane is combustion itself. The great volumetric change in going from liquid to burned gases and the momentum imparted to a particle (solid or liquid) both constitute damping phenomena in that they take energy from high instantaneous local pressures. Unfortunately, the combustion process can generate a great deal more pressure oscillation energy than is absorbed by its inherent damping mechanism.

Acoustical absorbers are applied usually as discrete *cavities* along or in the wall of the combustion chamber near the injector end. Both act as a series of Helmholtz resonators that remove energy from the vibratory system which otherwise would maintain the pressure oscillations. Figure 9–7 shows the application of discrete cavities (interrupted slots) at the "corner" of the injector face. The corner location usually minimizes the fabrication problems, and it is the one location in a combustion chamber where a pressure antinode exists for all

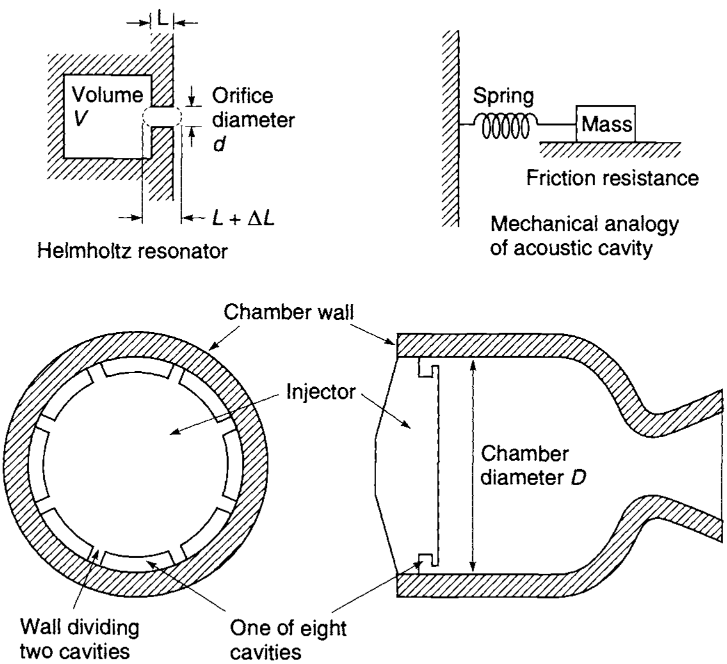

FIGURE 9–7. Diagram of acoustic energy absorber cavities at the periphery of an injector. In this thrust chamber the cavity restriction is a slot (in the shape of sections of a circular arc) and not a hole. Details of the chamber cooling channels, injector holes, or internal feed passages are not shown.

resonant modes of vibration, including longitudinal, tangential, radial, and combinations of these. Velocity oscillations are minimal at this point, which favors absorber effectiveness. Transverse modes of instability are best damped by locating absorbers at the corner location. Figure 9–7 also shows a Helmholtz resonator cavity and its working principles in simple form. Taking one resonator element, the mass of gas in the orifice with the volume of gas behind it forms an oscillatory system analogous to the spring–mass system shown (see Ref. 9–15). Even though Helmholtz resonator theory is well understood, problems exist in applying the theory to conditions of high pressure, temperature, chamber flow, and sound energy levels present when screech occurs, end in properly tuning the cavities to the estimated frequencies.

Absorption cavities designed as Helmholtz resonators placed in or near the injector face offer relatively high absorption bandwidth and energy absorbed per cycle. The Helmholtz resonator (an enclosed cavity with a small passage entry) dissipates energy twice each cycle (jets are formed upon inflow and outflow). Modern design practice favors acoustic absorbers over baffles. The storable propellant rocket engine shown in Fig. 8–2 has acoustic absorption cavities in the chamber wall at a location next to the injector.

The resonance frequency f of a Helmholtz cavity can be estimated as

$$f = \frac{a}{2\pi}\sqrt{\frac{A}{VL}} \tag{9–2}$$

Here a is the local acoustic velocity, A is the restrictor area, $A = (\pi/4)d^2$, and other symbols are as shownin Fig. 9–7. The ΔL is an empirical factor between 0.05 and 0.9 to allow for additional oscillating gas mass. It varies with the L/d ratio and the edge condition of the restricted orifice (sharp edge, rounded, chamfered). Resonators in thrust chambers are tuned or designed to perform their maximum damping at predicted frequencies.

Small changes in injector geometry or design can cause an unstable combustion to become stable and vice versa. New injectors, therefore, use the design and geometry of proven, stable prior designs with the same propellants. For example, the individual pattern of concentric tube injector elements used with gaseous hydrogen and liquid oxygen (shown in Fig. 8–3) are likely to be more stable, if the hydrogen gas is relatively warm and the injection velocity of the hydrogen is at least 10 times larger than that of the liquid oxygen.

In summary, the designer needs to (1) use data from prior successful engines and simulation programs to establish key design features and estimate the likely resonances, (2) design the feed system and structure to avoid these resonances, (3) use a robust injector design that will provide good mixing and dispersion of propellants and be resistant to disturbances, and (4) if needed, include tuned damping devices (cavities) to overcome acoustic oscillations. To validate that a particular thrust chamber is stable, it is necessary to test it over the range of likely operating conditions without encountering instability. An analysis is needed to determine the maximum and minimum likely propellant temperatures, maximum and minimum probable chamber pressures, and the highest and lowest mixture ratios, using a propellant budget as shown in Section 10.3. These limits then establish the variations of test conditions for this test series. Because of our improved understanding, the amount of testing needed to prove stability has been greatly reduced.

PROBLEMS

1. For a particular liquid propellant thrust chamber the following data are given:

Chamber presure	68 MPa
Chamber shape	cylindrical
Internal chamber diameter	0.270 m
Length of cylindrical section	0.500 m
Nozzle convergent section angle	45°
Throat diameter and radius of wall curvature	0.050 m
Injector face	Flat
Average chamber gas temperature	2800 K
Average chamber gas molecular weight	20 kg/kg-mol
Specific heat ratio	1.20

   Assume the gas composition and temperature to be uniform in the cylindrical chamber section. State any other assumptions that may be needed. Determine the approximate resonance frequencies in the first longitudinal mode, radial mode, and tangential mode.

2. In Problem 1, explain how these three frequencies will change with combustion temperature, chamber pressure, chamber length, chamber diameter, and throat diameter.

3. Why does heat transfer increase during combustion instability?

4. Prepare a list of steps for undertaking a series of tests to validate the stability of a new pressure-fed liquid bipropellant rocket engine.

5. Estimate the resonant frequency of a set of each of nine cavities similar to Fig. 9–7. Here the chamber diameter $D = 0.200$ m, the slot width is 1.0 mm, and the width and height of the cavity are each 20.0 mm. The walls separating the individual cavities are 10.0 mm thick. Assume $L = 4.00$ mm, $\Delta L = 3.00$ mm, and $a = 1050$ m/sec.
   Answer: approximately 3138 cycles/sec.

REFERENCES

9–1. R. D. Sutton, W. S. Hines, and L. P. Combs, "Development and Application of a Comprehensive Analysis of Liquid Rocket Combustion," *AIAA Journal*, Vol. 10, No. 2, Feburary 1972, pp. 194–203.

9–2. K. K. Kuo, *Principles of Combustion*, John Wiley & Sons, New York, 1986.

9–3. V. Yang and W. Anderson (Eds.) *Liquid Rocket Engine Combustion Instability*, Vol. 169 of *Progress in Astronautics and Aeronautics*, AIAA, 1995, in particular Chapter 1, F.E.C. Culick and V. Yang, "Overview of Combustion Instabilities in Liquid Propellant Rocket Engines."

9–4. B. R. Lawver, "Photographic Observations of Reactive Stream Impingement," *Journal of Spacecraft and Rockets*, Vol. 17, No. 2, March–April 1980, pp. 134–139.

9–5. M. Tanaka and W. Daimon, "ExplosionPhenomena from Contact of Hypergolic Liquids," *Journal of Propulsion and Power*, Vol. 1, No. 4, 1984, pp. 314–316.

9–6. P. Y. Liang, R. J. Jensen, and Y. M. Chang, "Numerical Analysis of the SSME Preburner Injector Atomization and Combustion Process," *Journal of Propulsion and Power*, Vol. 3, No. 6, November–December 1987, pp. 508–513.

9–7. M. Habiballah, D. Lourme, and F. Pit, "PHEDRE—Numerical Model for Combustion Stability Studies Applied to the Ariane Viking Engine," *Journal of Propulsion and Power*, Vol. 7, No. 3, May–June 1991, pp. 322–329.

9–8. R. I. Sujith, G. A. Waldherr, J. I. Jagoda and B. T. Zinn, "Experimental Investigation of the Evaporation of Droplets in Axial Acoustic Fields," *Journal of Propulsion and Power*, AIAA, Vol. 16, No. 2, March–April 2000, pp. 278–285.

9–9. D. T. Hartje (Ed.), "Liquid Propellant Rocket Combustion Instability," *NASA SP-194*, U.S. Government Printing Office, No. 3300–0450, 1972.

9–10. B. W. Oppenheim and S. Rubin," Advanced Pogo Analysis for Liquid Rockets," *Journal of Spacecraft and Rockets*, Vol. 30, No. 3, May–June 1993.

9–11. G. About et al., "A New Approach of POGO Phenomenon Three-Dimensional Studies on the Ariane 4 Launcher," *Acta Astronautica*, Vol. 15, Nos. 6 and 7, 1987, pp. 321–330.

9–12. T Shimura and K. Kamijo, "Dynamic Response of the LE-5 Rocket Engine Liquid Oxygen Pump," *Journal of Spacecraft and Rockets*, Vol. 22, No. 7, March–April 1985.

9–13. E. Kirner, W. Oechslein, D. Thelemann and D. Wolf, "Development Status of the Vulcain (HM 60) Thrust Chamber." *AIAA Paper 90;2255*, July 1990.

9–14. F. H. Reardon, "Combustion Stability Specification and Verification Procedure," *CPIA Publication 247*, October 1973.

9–15. T. L. Acker and C. E. Mitchell, "Combustion Zone–Acoustic Cavity Interactions in Rocket Combustors," *Journal of Propulsion and Power*, Vol. 10, No 2, March–April 1994, pp. 235–243.