Liquid Rocket Performance, Stability, and Compatibility

RALPH S. VALENTINE*
Atlantic Research Corporation, Alexandria, Va.

Ralph S. Valentine is Director of Engineering Research for the Atlantic Research Corporation, a Division of The Susquehanna Corporation. In this position, he directs research in technology areas associated with liquid rocket engines and propellants. Dr. Valentine has been involved with liquid rocket engine performance, stability and compatibility analysis since 1963 in his current function and in his previous assignment as Manager of Fluid Dynamics for the Aerojet Liquid Rocket Company. He received an MS degree in Chemical Engineering from the University of Illinois and BS and Ph.D. degrees from the University of Washington. Dr. Valentine has been a member of the Propellants and Combustion Technical Committee and is an Associate Fellow of the AIAA He has held elective offices in the Sacramento Section. He is listed in *Leaders in American Science, American Men of Science, Who's Who in the South and Southwest*, and the *International Scholars Directory*.

Nomenclature

A = empirical constant, in.$^{-1}$
A_c = chamber area, in.^{2}
A_t = thrust area, in.^{2}
C_d = orifice discharge coefficient
d = orifice diameter, in.
f = frequency, Hz
F = thrust, lbf
I_s = specific impulse lbf/lbm/sec
L^* = characteristic length, in.; V_c/A_t
L/D = orifice length to diamteer ratio
n = number of orifices
O/F = oxidizer to fuel weight ratio
ODE = one-dimensional equilibrium
P_a = ambient pressure, psia
P_c = chamber pressure, psia
V_c = chamber volume, in.^{3}
$\dot{w}$ = weight flow, lb/sec
ΔP = pressure drop, psi
ΔV = gas/droplet velocity differential, fps
γ = ratio of specific heats
ε = expansion area ratio

Introduction

SINCE the 1940's, when liquid rockets first became a significant factor in the aerospace industry, there has been continued effort to better understand their operation. The thermodynamic constraints upon specific impulse (I_s) performance were recognized early, but only recently have the kinetic factors been understood. Combustion instability evolved as an operational problem in the 1950's and has been the subject of extensive analysis and experimentation since that time. The problem of wall compatibility is ever present, but has received significant attention from the propulsion community only in recent years. The course of research in these areas has been relatively independent. In 1960, however, the concept of droplet vaporization was expounded[1] as a means of understanding liquid rocket engine operating characteristics. Subsequently, this approach has become the basis for tying together many of the previously unrelated component analyses into a general matrix for engine evaluation.

Most liquid rocket engines in use today which operate at thrust levels of 1000 lbf or greater were developed by methods which can be best described as trial and error. Original injector designs were often revised as many as 100 times, with each version requiring fabrication and testing to determine operating characteristics. When a concept evolved which exhibited acceptable I_s and stability characteristics, only then was wall compatibility considered in long duration firings. This development cycle was based upon applications of basic thermodynamic and aerodynamic theory and was guided by the best analytical techniques available at the time. The solution of problems was often single-faceted, however, and did not consider the interacting effects of the parameters which influenced performance, stability and compatibility. As a result, many systems flying today are far from optimized. They operate at nonoptimum mixture ratios, employ correctional devices to improve stability and have thrust chambers which are much heavier than would be necessary with a compatible injector design.

Recognition of this situation has led to a great deal of emphasis on the part of government agencies and industrial contractors upon providing improved tools for prediction and analysis of engine operating characteristics. Working groups were set by the Interagency Chemical Rocket Propulsion Group (ICRPG) to study the combustion stability problem and to identify best techniques for performance analysis. Considerable independent work has been accomplished to evaluate the effect of chamber environment on wall compatibility. The advent of sophisticated droplet atomization and vaporization analysis techniques has provided a common base for each technology area which has served to consolidate many of the previously unrelated efforts. It may be said that prediction and analysis of engine operating parameters has made the transition from an art to a science. The industry is currently capable of understanding the primary effects and mutual interactions between the factors influencing performance, stability and compatibility. This is the key to design and analysis methods which are within practical limits of accuracy.

The operational characteristics of a liquid rocket engine are determined by the choice of certain definable parameters as shown in Fig. 1. The definable parameters include the pro-

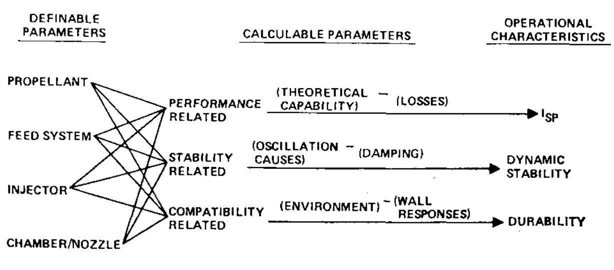

Fig. 1 Liquid rocket engine operation.

Received July 27, 1971; revision received January 19, 1972. The contributions of F. H. Reardon to the section on combustion stability are gratefully acknowledged.
Index category: Liquid Rocket Engines.
*Director of Engineering Research.

Table 1 Liquid rocket engine definable parameters

Propellant	*Feed System*	*Injector*	*Chamber/Nozzle*
Chemical composition	Line velocity	Injection velocity	Length
Viscosity	Line pressure drop	Atomization characteristics	L^*
Density	Line volume	F per element	P_c
Enthalpy	Line inlet temperature	Orifice diameter	Contraction ratio
Surface tension	Line length	Orifice L/D	Chamber L/D
Critical temperature	Channel velocity	Thrust per unit face area	Mass flow, primary and secondary
Specific heat	Surface finish	Primary mass flow	Gas residence time
γ of combustion products		Element type	Throat diameter
Heat of formation		Number of elements	Chamber diameter
Kinetic constants		Orifice length	Chamber shape
Heat of reaction		Injection area	Gas velocity
Sonic velocity		Number of fuel holes	Chamber upstream turning radius
Compressibility		Number of oxidizer holes	Throat turning radius
Heat of vaporization		Surface finish	Wall material
Monopropellant character		Orifice inlet geometry	Wall structure design
Hypergolicity		Orifice angle to flow	Mechanical damping devices
		O/F	Exit angle
			Nozzle contour

pellant properties and design constraints on the feed system, injector, and thrust chamber. From these, parameters related to performance, stability and compatibility may be calculated. Thus, deliverable I_s is the difference between thermodynamic capability of the engine system and the loss factors which prevent attainment of this potential. Dynamic stability characteristics are based on the balance between causes of combustion oscillation land the conditions which damp such oscillations. Durability of the system is the resultant of the chamber environment generated and the response of the wall material to this environment. All of these factors may be calculated or estimated by techniques which have been published and are in the industrial domain. The definable parameters of which knowledge is required are listed in Table 1. The parameters which may be derived from this input information are given in Table 2.

The primary factors which influence performance, stability and compatibility are shown in Table 3. Exact knowledge of each of these would permit an exact prediction of engine operating characteristics. Each of these factors results from a complex combination of relationships and interactions between the various calculable parameters. These will be discussed in detail later.

Table 2 Liquid rocket engine calculable parameters

Performance	*Stability*
ODE performance	Initial drop size distribution
O/F distribution	Orifice discharge coefficients
Mass distribution	Hydraulic flip
Feed system ΔP	Drop shattering
Orifice inlet temperature	Drop stripping
Orifice ΔP	Mass distribution
Orifice discharge coefficients	Energy release profile
Hydraulic flip	System natural frequencies
Momentum angle	Wall energy absorption
Momentum ratio	Exhaust nozzle damping
Spray fan distribution	Condensed phase damping
Drop shattering	
Drop stripping	*Compatibility*
Differential velocity	Orifice discharge coefficients
Blow apart effects	Hydraulic flip
Drop size distribution	Mass distribution
Vaporization distance	Drop size distribution
Mixing length	Blow apart effects
Turbulence	Spray fan distribution
Fractional loss	Turbulence
Divergence	Radial velocity
Three-dimensional effects	Transverse velocity
Kinetic equilibrium	Heat transfer coefficient
Particle lag	Energy release profile
Stagnation pressure	Wall erosion potential
	Wall corrosion potential

Table 3 Rocket engine operating factors

Performance factors		
Theoretical capability	Thermodynamic potential	I_s
Degree of Achievement	Geometry loss	
	Boundary-layer loss	
	Kinetic loss	
	Two-phase flow loss	
	O/F distribution loss	
	Energy release loss	
Stability factors		
Corrective factor	Damping rate	Dynamic stability
Oscillation causative factors	Feed system coupling frequency	
	Chamber natural frequencies	
	Susceptibility to "pops"	
	Transient combustion phenomena	
Compatibility factors		
Environment	Thermal environment	Durability
	Chemical environment	
	Gas dynamic environment	
Ability to withstand environment	Wall material sensitivity	

I_s Performance

Rocket engine specific impulse is defined as the amount of net axial thrust generated per unit of propellant mass flow. The upper limit is generally accepted as the value obtained at chemical equilibrium with the reaction products expanded through a one-dimensional nozzle by an adiabatic reversible process. This is referred to as ODE performance, and represents the maximum attainable for a given O/F, P_c, ε, and P_a, or the thermodynamic potential of the system.

Achievement of this potential is affected by several loss mechanisms which degrade the performance efficiency. Some of these are inherent to the system and cannot be completely eliminated in a practical engine. In this category are geometric, boundary layer, particle lag, and kinetic losses. Others can be controlled to an extent by system design and in some cases eliminated completely. These include thermal, combustion efficiency and nonuniform propellant distribution losses.

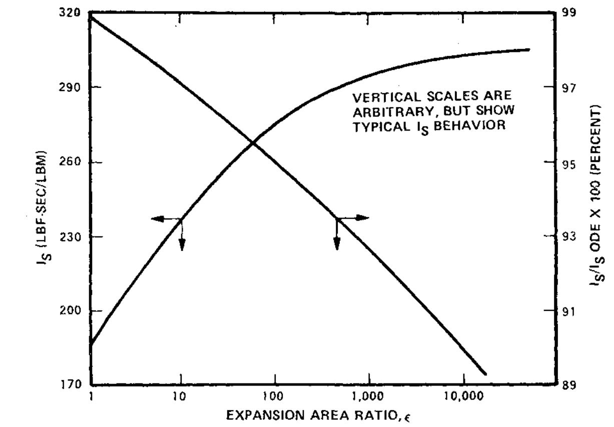

Fig. 2 Typical performance of a liquid rocket engine.

The relative importance of these loss mechanisms depends upon the system operating conditions. A low P_c engine might suffer more I_s loss from kinetic effects than from other losses. A very low thrust engine might exhibit a predominant boundary-layer loss. An imperfect injector or very short combustion length could make poor combustion efficiency the most significant loss mechanism.

A very high ε engine would exhibit a low performance relative to ODE due to increased boundary layer and kinetic losses even though I_s increases with ε. Typical engine performance with ε is shown in Fig. 2. There is no relationship between the magnitude of the various losses which may be considered as generally applicable.

Performance predictions are made by calculation of the values of the various losses based upon the engine design and operating conditions. The methods available to do this will be discussed later. It test data are available, performance analysis permits apportionment of the observed difference between ODE and measured I_s into the component losses. Without test data, the greatest limitation to performance analysis has been the uncertainty in estimating combustion efficiency. However, even when combustion efficiency has been established experimentally, serious errors have been observed in data extrapolation. The probable reason for these errors lies in interaction effects which have only recently been recognized and taken into account in performance calculations.

One such interaction is the effect of the level of combustion efficiency in the chamber upon the nozzle expansion process.

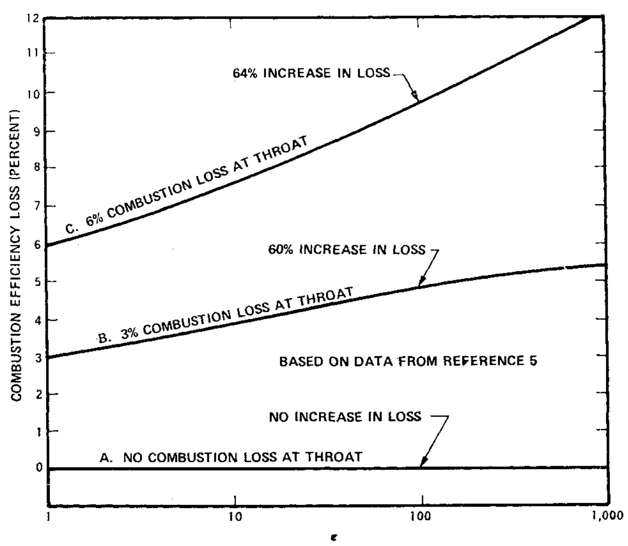

Fig. 3 Typical effect of ε on apparent combustion loss.

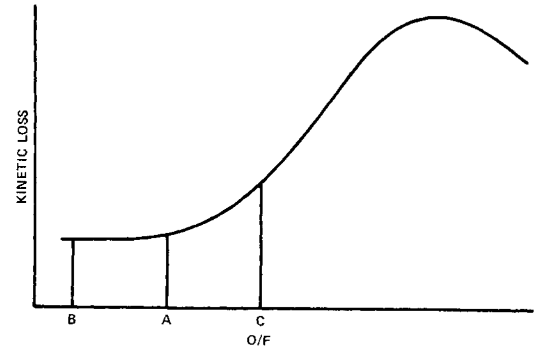

Fig. 4 Effect of O/F on kinetic loss.

This effect was recognized[3,4] when a comparison of high and low ε test data on an engine system showed an apparent reduction in combustion efficiency at the higher ε. Several mechanisms have been proposed for this effect.[2,5,6] The magnitude of the loss is a function of the propellant system, P_c, ε, and O/F. A typical loss from this effect is shown in Fig. 3. Note in Fig. 3 that the loss at high ε is a function of the combustion efficiency at the throat, increasing in both absolute value and percent of ODE as ε increases. It has been established[7] that this loss in I_s cannot be recovered even if combustion is completed in the nozzle.

Another obvious interaction is the effect of nonuniform O/F upon any loss that is O/F dependent. For example, kinetic losses often vary with O/F as shown in Fig. 4. It is apparent that an engine operating at point A would suffer less loss than an engine with 50% mass flow at point B and 50% at point C, even though both engines have the same over all O/F. Other interactions of significance[2] include the effects of incomplete combustion, nonuniform O/F and nonequilibrium flow on boundary-layer loss and the effects of incomplete combustion and three-dimensional flow upon kinetic loss.

Factors Influencing Performance Losses

Each of the loss mechanisms mentioned above is influenced to some degree by the properties of the propellants and the design of the thrust chamber. Although thermodynamic and kinetic phenomena determine the performance potential of the system, the degree of achievement is controlled by aerodynamic effects.[8] The physical processes which occur when liquid propellants are injected into a rocket combustion chamber are atomization, vaporization, mixing, reaction and expansion. In most cases, vaporization of one propellant is limiting, the other processes occurring relatively fast by comparison.

Mixing in the combustion chamber will not resolve nonuniform O/F at the injector unless the combustor is very long and narrow or turbulence is generated mechanically. Gas phase mixing is very rapid within a stream tube of a size similar to the element spacing. Mixing between such stream tubes is a function of turbulence intensity,[9] combustor length and element spacing. Typical values of rocket chamber turbulence intensity[10] are such that gas phase mixing is generally 97–98% complete in a length equal to 4–10 times the element spacing. Thus, for an injector with $\frac{1}{2}$-in. element spacing, essentially complete combustion at the local O/F would occur within 2–5 in. after droplet vaporization was complete. Over-all nonuniform O/F distribution will, however, still be a significant factor in absolute performance. Both the shape of the I_s vs O/F curve and absolute level of I_s will vary as a function of O/F distribution during excursions of over-all O/F from the nominal value. The effect is compounded by nozzle interactions.[11]

Droplet atomization and vaporization are the processes which generally determine combustion efficiency and have a secondary effect on kinetic and boundary-layer losses.

Atomization is controlled primarily by the injector design while vaporization rates depend upon the chamber design and propellant properties. The optimum injector design for performance is that which atomizes both propellants to the degree that they will vaporize at the same rate in the chamber. Vaporization of both should be completed at the same distance downstream from the injector. The role of the chamber is to provide a sufficiently large relative velocity (ΔV) between gas and droplets to vaporize completely in the available length. The ΔV profile is determined to a great extent by contraction ratio, A_c/A_t.

Many other factors also influence atomization and vaporization. Self-atomization occurs in the orifice effluent stream in proportion to the orifice ΔP and the propellant physical properties. Sheets atomize to a greater extent than jets for similar flow rates. Initial drop size is dependent on aerodynamic effects near the injector face, in particular ΔV (Ref. 12). Large drops may be shattered by deformation, or small elements of propellant may be stripped from the surface.[13] Transient oscillations can significantly reduce drop size.[14] Secondary atomization occurs when like or unlike streams impinge by transfer of momentum which converts kinetic energy into the generation of surface area. Hypergolicity of the propellants or approach to critical pressure affects atomization results. Induced turbulence or mechanical flow restrictors also increase droplet breakup.[15]

Vaporization starts as soon as the surface of the drop reaches the boiling point. At first, most of the heat transferred to the drop goes to sensible heat either through conduction or circulation within the drop.[16,17] At pressures approaching critical, no steady-state vaporization occurs as no definite boiling point exists.[18] Heat-transfer rates to the drops are a function of ΔV, which has a steady component due to axial flow an an unsteady component resulting from random transverse flows or combustion oscillations. As the drop progresses down the chamber, secondary shattering may occur as ΔV increases and the maximum stable drop size decreases.

The distribution of propellant in the spray fan from an impinging element determines whether or not mixing will be a determinant of performance. Element type, orientation and spacing determines the O/F distribution within a stream tube. I_s variations of several percent have been noted solely by reorienting identical elements to change the degree of overlap of the fuel-rich and oxidizer-rich portions of the spray fan. Criteria for design of several types of elements to provide optimum mixing have been developed.[19] These criteria do not apply to hypergolic propellants which exhibit the stream separation, or blow-apart, phenomenon.[20] It has also been hypothesized that streams of similar stagnation pressure may result in an unstable spray fan with random fluctuations in concentration.[21]

The blow-apart phenomenon results from surface reactions in hypergolic streams at the impingement point. The gas generated from these reactions expands and separates the streams, which then reform in a cyclic process.[22] Specific blow-apart characteristics are a function of P_c and stream size.[23] Smaller streams show less tendency to separate. Impinging sheets are less likely to blow-apart than jets of the same flow rate.[24] The atomization characteristics of hypergolic propellants differ significantly from nonhypergolics as a result of this process.

Hydraulic effects in the injector orifices or manifold can also effect achievable performance. The magnitude and location of ΔP in the manifold flow passages determines O/F distribution. Heat soakback from the injector face is a further complicating factor, as propellant temperature will vary with element location. Further, since fuel and oxidizer often enter the injector from opposite sides, a situation may exist where the coldest fuel is injected with the warmest oxidizer. This will affect atomization characteristics in a manner that is very difficult to define when other hydraulic perturbations are superimposed.

Surface finish of the injector orifices influences hydraulic behavior as do entry geometry, orientation to the flow, L/D and ΔP. A 10–20% reduction in C_d was noted between orifices with a 5 μin. ground surface compared to a 50 μin. EDM surface.[25] Stream effluent geometry and direction are influenced by manifold velocity and the angle of the orifice to the flow.[26] Streams from a round orifice vary from round jets to bushy sprays and flattened sheets. Stream direction has been observed to deviate by 20° or more from the orifice axis. Thus, orifices designed to impinge at a given point may miss each other completely. Hydraulic flip within the orifice is also commonly observed if orifice L/D is in the range of 3–8. Below this range consistent sharp edge orifice flow is usually obtained and above $L/D = 8$, attached tube flow generally occurs. When hydraulic flip occurs, C_d variations of 10–30% are observed, leading to large local O/F variations. It is clear that these hydraulic effects make generalizations with respect to atomization very risky indeed.

Performance Analysis Techniques

Until 1968, most performance analysis was accomplished by summing the results of independent calculations of boundary, geometry, kinetic, multiphase flow and thermal losses. The methods used were generally based upon use of an optimum nozzle contour,[27] a given chamber pressure, propellant system and over-all O/F. No method existed for calculation of combustion efficiency based upon design and operational variables. Combustion efficiency was either estimated by gross rules of thumb, or determined from measured c^* or by difference between measured I_s plus known losses from ODE.

Little uniformity existed within the industry. Over 70 different techniques have been reported for boundary-layer calculations alone. Kinetic rates as reported in the literature varied over a range which often was wide enough to account of several percent I_s. Claims made in proposals submitted to government agencies were difficult to confirm because of the lack of uniform calculational methodology.

Concern over nonstandardized performance analysis led to the formation of the Performance Standardization Working Group (PSWG) of the Interagency Chemical Rocket Propulsion Group (ICRPG) in 1965. The mission of this group was to survey the available techniques for evaluating I_s and recommend the best combination of analysis methods.[28]

The PSWG met its initial goal in 1968 with the publication of the ICRPG Liquid Propellant Thrust Chamber Performance Evaluation Manual.[2] This document presents an over-all matrix for performance analysis which not only uses the best calculation techniques, but takes into account interactions related to O/F distribution and incomplete combustion. All of the required methodology[2,28] is available on request from the CPIA.

The method recommended by the PSWG does not include a method for independent evaluation of combustion efficiency from basic design and operational data. It is used primarily to extrapolate test data to other operating conditions and to evaluate the distribution of losses from experimental results. Interaction effects resulting from incomplete combustion are estimated by thermodynamic and kinetic calculations based upon reduced enthalpy of the reaction products at the throat.[2,5] An improved technique has recently been suggested which determines combustion efficiency and nozzle interactions independently by droplet vaporization analysis.[29] This technique fills the need for a method of calculating combustion efficiency which does not depend upon test data and which can take into account specific injector/chamber design variables. The limitation to use of this method is in determination of initial drop size distribution.

When extensive firing data are available on similar engines, statistical methods may be used to predict I_s of future engines of the same design. Multiple covariance computer methods can be used to determine the primary effects of variations in

O/F, P_c, propellant temperature and thrust level. No understanding is required of the effects of interactions or thermodynamic processes. The independent variables are merely related to c^*, C_f, and I_s by statistical analysis. This method is limited in that it only applies to engines exactly the same as those fired to obtain the original data. Usually, 20 to 100 test firings are necessary to reduce the confidence limits on predictions to within acceptable bounds. Thus, statistical methods are attractive only for systems which are available in production quantities.

The accuracy of measured performance parameters is a primary determinant of the quality of performance analysis results. The PSWG recognized that accurate analytical methods would be of little value if the input data were in question. This led to publication by the PSWG of a document outlining best practices for measurement of engine parameters.[30] Of course, there is always some measurement uncertainty even with best practices used. This uncertainty derives from two sources: fixed bias errors and random variations which affect precision. Techniques for evaluation of the uncertainty in measurements have also been published by the PSWG.[31,32]

Stability

The combustion process in a liquid rocket thrust chamber is inherently unsteady. Hydraulic factors can lead to nonsteady flow within an orifice.[26] Under certain conditions, an element with steady flow will exhibit an unstable spray fan.[21] Even in stable operation, at steady state, pockets of propellants are constantly being formed, reacted and reformed, so that the radial and axial distributions of energy release are undefinable except in time-averaged terms. The amplitude of the random fluctuations of any combustion or flow quantity varies within the chamber depending upon the local conditions. For example, there is no single place where a truly representative pressure measurement may be taken, either for its mean absolute value, or its fluctuation intensity. If the measured pressure fluctuations are quite small, the combustion is considered smooth. If they exceed about 5% of the mean P_c, it is classified as rough.[33] The difference is only a matter of degree, as there is no fundamental change in combustion behavior regardless of the measured level of random pressure variations. Indeed, the stable combustion process may be characterized as a series of random events which fall within certain bounds. They are self-correcting about the mean, as a higher pressure reduces the flow of propellants to the chamber which in turn lowers P_c. A single, large amplitude fluctuation sometimes occurs, and is referred to as a pressure spike. However, if pressure fluctuations interact with the natural frequencies of the feed system or the acoustic properties of the chamber, periodic oscillations may occur at frequencies characteristic of the system. Once initiated, these oscillations may be damped, maintained or amplified by the combustion process. The occurence of sustained oscillations is generally termed combustion instability. Random fluctuations may be superimposed upon the periodic oscillations as illustrated in Fig. 5. Two or more modes of periodic oscillations may occur simultaneously as well. Absence of periodic escillations is referred to as stable operation regardless of the level of random pressure fluctuations.

A combustor which is capable of sustaining periodic oscillations does not necessarily exhibit such instability at all times. It may operate in a metastable condition, only suffering instability when a combination of random events triggers it. An artificial disturbance (produced, for example, by an explosive charge) is often used to test the stability characteristics of an engine, permitting observation of the response to a sudden local release of energy. If the engine fails to develop oscillations or the oscillations are damped within a sufficiently short time, the engine is considered dynamically stable.

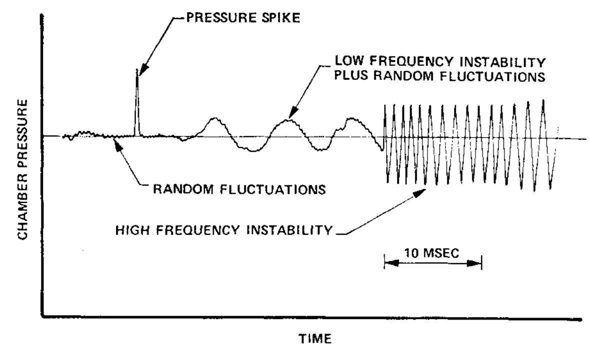

Fig. 5 Typical combustion oscillations in a rocket engine.

Periodic combustion oscillations have been categorized according to the parts of the engine system which are interacting.[33] Frequencies in the range of 10–200 Hz ("low-frequency" instability) result from coupling between the combustion process and the propellant feed system. "High-frequency" instability (over 1000 Hz except in very large chambers) is associated with the acoustic properties of the chamber volume. Intermediate frequencies are believed to involve hydraulic or thermal phenomena within the injector flow passages, or mechanical vibrations in the engine structure. Sizable combustion oscillations, either random or periodic, are normally considered undesirable in that they may lead to increased thermal stress in engine components and thus reduce engine life.

By analogy to the classical acoustic modes of a cylindrical cavity, cases of high-frequency instability are classified as longitudinal, radial, or tangential. Combined modes also occur. The tangential modes are most destructive; they often exhibit oscillation amplitudes (peak-to-peak) equal to the mean chamber pressure and increases in wall heat transfer up to an order of magnitude higher than normal. Exposure to such an oscillation for as long as 0.3 sec will usually result in destruction of the chamber.

Factors Influencing Stability

The random pressure fluctuations that occur in every rocket thrust chamber yield results similar to those of periodic oscillations, though far less destructive. The transverse flows generated do have an impact upon heat transfer rates, impingement of orifice streams, droplet atomization characteristics, and vaporization rates. In addition, these random perturbations may occasionally trigger instability. Pressure spikes, random fluctuations of greater than normal amplitude often approaching the magnitude of P_c itself, have been observed to trigger instabilities.[26] This is not surprising since they act very much like an explosive charge, suddenly releasing a large amount of energy in a local area of the chamber.

Local perturbations in the combustion process are the cause of random pressure fluctuations. These combustion perturbations are related to propellant properties and injector design features. Each element of the injector functions essentially independently[34] as a vehicle for atomizing and mixing the propellants. The degree of mixing achieved is a function of the orifice entrance conditions and mechanical characteristics which vary from element to element. There is only a limited correlation between events occurring at different locations within the chanber. The effects of random pressure fluctuations can be minimized by axisymmetric propellant flow, uniform O/F and mass distribution and small elements. However, these fluctuations can never be eliminated in a practical rocket engine.

Low and intermediate frequency instabilities require coupling between the combustion process and a flow process or mechanical system external to the combustion chamber.

Thus, the natural frequencies of the feed system or of structural components under dynamic conditions determine whether or not this type of instability can occur. The combustion process can be decoupled from the flow system by increasing the ΔP across the injector orifices. If the ΔP is on the order of $\frac{1}{2}P_c$, low frequency instability will rarely occur. The use of suppression devices or impedance matching allows ΔP values lower than $\frac{1}{2}P_c$ to be used while still maintaining stability. Alternatively, the natural frequency of the feed system can be changed by altering line lengths or manifold volumes or by installing energy absorbing devices such as Helmholtz resonators or quarter-wave tubes.[35] The natural frequency of a mechanical component can be varied by changing connection points or adding stiffening braces. Changes can also be made within the chamber to decrease the range of susceptibility to low or intermediate frequency oscillations. An increase in L^* or injection orifice L/D usually improves stability.[36] For gaseous H_2 systems, lower propellant density decreases the tendency toward instability.[37,38]

High-frequency instability is usually independent of factors external to the combustion chamber, since it results from an interaction between the combustion process and the chamber acoustic resonance properties. Thus, it is influenced by propellant characteristics as well as chamber geometry. The important propellant properties are those relating to the dynamic response of the combustion process to disturbances within the chamber. This response has been represented by a characteristic time and a sensitivity factor,[34] which are dependent on propellant volatility, hypergolicity, degree of atomization, and the operating chamber pressure and mixture ratio.[39] The chamber design not only determines the characteristic acoustic frequencies, but also strongly influences the gas/droplet ΔV, which controls vaporization rates.

The most sensitive stability zone is where ΔV is lowest, usually within a few inches of the injector face.[40] A typical vaporization profile is shown in Fig. 6. The vaporization rate per unit of droplet surface area increases with ΔV while the total droplet surface area decreases. Thus, the vaporization rate and gas velocity reach a maximum rate of increase at point A on Fig. 6. The liquid velocity starts at the injection velocity and decelerates until $\Delta V = 0$ at point *B*. Beyond point B, the gases are accelerating the droplets, but a lag remains until the droplets are completely vaporized at point *C*.

Factors which determine the location of the $\Delta V = 0$ point are[41–43]: drop size distribution; transient heating rate of drops; propellant volatility; gas velocity; mass flux distribution; P_c; and O/F distribution. The closer the $\Delta V = 0$ point is to the injector, the less stable.[44] The sensitive region moves closer to the face if the following things occur[14,42–47]: decrease in orifice size, injection velocity, A_c/A_t; increase in propellant temperatures; transverse flows; more uniform mass and O/F distributions.

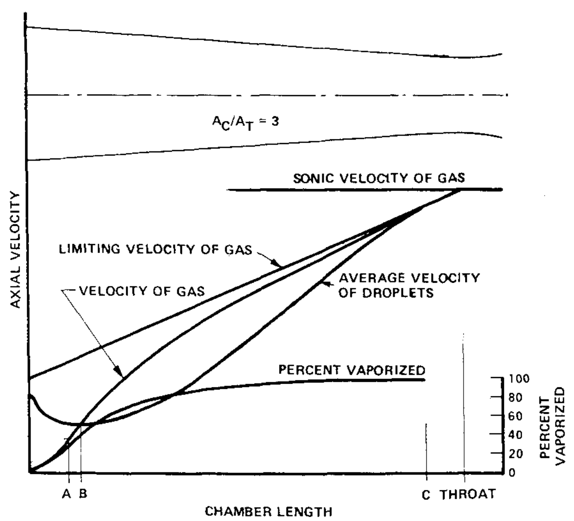

Fig. 6 Typical vaporization profile for a conical chamber with $A_c/A_t = 3$.

As the $\Delta V = 0$ point approaches the injector face, the energy release per unit combustor volume near the injector increases, providing the driving force for a sustained instability. Transverse waves near the injector face may exhibit a pressure ratio of 20/1 (Ref. 41). When transverse flows occur, liquid may be stripped from the local droplets adding energy to the waves and supporting the instability.[13,47] Since stripping or shattering is a function of drop size, there may be a critical size which minimizes stability.[46] However, at high ΔV, the atomization rate is less sensitive to pressure oscillation.[48]

For combustors operating above the critical pressure of the propellants, the driving mechanisms for instability may be quite different. That is, the heating-up of the propellants becomes much more important than at low pressures,[49] and there are no sharp density change across a phase boundary. Thus, there are similarities to operation with gaseous propellants. For similar geometry, gaseous systems are generally more stable[50] although not completely free of combustion instability problems. It has been found that increasing propellant density[37,38] and operation at off-stoichiometric mixture ratios[51,52] increase the tendency to unstable operation of gas rockets. Coupling between feed system and combustion chamber oscillations appears to be a significant factor in both low- and high-frequency types of gaseous system instability.[37]

The size of the disturbance required to initiate instability varies widely, ranging from the level of the unavoidable, random fluctuations always present in the combustion system up to pressure amplitudes of the order of the mean chamber pressure. Because of the complex interactions involved in combustion instability, there are no simple rules-of-thumb for predicting the size of the triggering disturbance for a given system. In fact, this perturbation amplitude is commonly used to measure the resistance of rocket engines to combustion instability (see Chap. 10 of Ref. 53).

Stability Analysis Techniques

The development of theories for analysis of high-frequency instability has been actively pursued for 30 yr. The total effort devoted to stability analysis probably exceeds that applied to performance and compatibility theory by an order of magnitude. Unfortunately, only a small fraction of this analysis effort has been applied in engine development programs. Most emphasis has centered on two widely accepted approaches, the Sensitive Time Lag Theory[34] and the Droplet Vaporization Model.[54] Another, more empirical approach has been developed which has practical utility but is not based upon a comprehensive theory.[55,56] The application of these and other theories to the definition of design criteria to minimize instability in engine development programs has been the objective of the ICRPG Liquid Propellant Combustion Instability Working Group.

A comprehensive compilation of the various analytical theories and experimental results has been prepared recently.[53] It contains contributions by many eminent specialists in the stability field, and represents the most complete treatise available on stability theory and practice. An earlier, less detailed general introduction to stability theory, analysis methods and fixes is given in Ref. 33.

The concept of a combustion time lag was first proposed in 1941 by von Kármán. Initial application was to low-frequency instability,[57,58] with the time lag regarded as a constant. The concept was extended to high frequency oscillation problems in the early 1950s by regarding the combustion time lag to be sensitive to fluctuations in the state of the gases in the chamber.[34] The combustion time lag is a mathematical artifice introduced to simplify the analytical description of the unsteady combustion process. That is, the continuous evolution of products from a given mass of reactants is replaced by a step-function change from reactants to products. The sensitivity of the rates of the combustion processes is similarly simplified by assuming that the total time lag can be represented as the sum of a constant "insensitive time lag" and a variable "sensitive time lag," the latter being only about 10% of the total time lag. An "interaction index" is also defined; it measures the degree of sensitivity, that is, the amount of variation of the sensitive time lag for a unit pressure perturbation. It has been found that the combustion responses calculated from this very simple model are in good agreement with more sophisticated formulations.[59]

The sensitive time lag and interaction index have been determined by experiment to be functions of propellant combination, injection pattern, chamber pressure, and mixture ratio (see Sec. 6.3 of Ref. 53). For a given combustion chamber configuration, stability boundaries can be determined in terms of the sensitive time lag and interaction index, as shown in Fig. 7. At the stability boundary, the energy produced by the unsteady combustion process can be completely dissipated by the aerodynamic processes in the chamber. In the region of stability (below the curves in Fig. 7), more energy can be absorbed than is generated, so that oscillations will be damped. In the region of instability (above the curves) the energy generation exceeds the absorption; hence, the oscillation amplitude will increase. Comparison of the calculated stability boundaries for a given combustion chamber with the values of the interaction index and time lag for the given injector allows the stability of the injector-chamber combination to be predicted.

The sensitive time lag theory has been most completely developed under the simplifying assumption that perturbations in chamber conditions are small enough that nonlinear terms can be neglected. Both longitudinal and transverse modes have been considered, as well as the effects of thermodynamic state and gas displacement on the combustion response. In recent years, a significant amount of work has been done on nonlinear effects. A summary of theoretical and practical aspects of this approach has recently been published.[39] A design guide is also available.[60]

In 1962, a stability analysis technique based on quasi-steady droplet vaporization was presented.[54] The energy required to support oscillating chamber conditions must come from the droplet burning process. In the original formulation, the burning rate was taken to be identical to the vaporization rate, which depends at any instant on the gas density and relative velocity between droplet and gas. Thus under oscillating conditions both chamber pressure and relative velocity fluctuations (ΔV) relative to their steady-state values, determine the energy contributed to the oscillation.

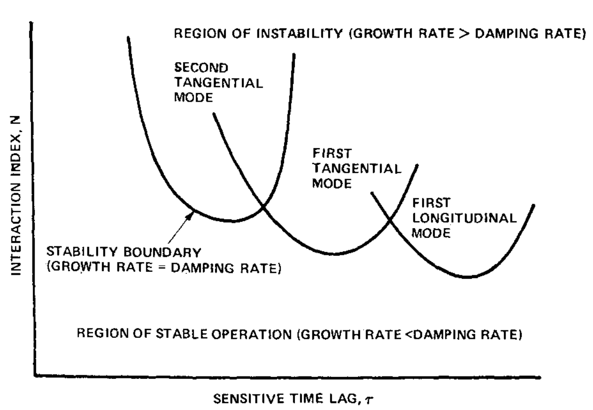

Fig. 7 Sensitive time lag stability map.

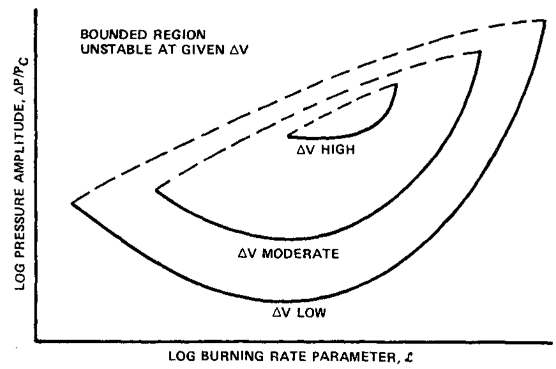

Fig. 8 Typical stability map based on vaporization.

Because this approach is based on numerical solution of the conservation equations, nearly all studies have been restricted to one space dimension. That is, the analysis has generally been applied to an annular region (within the combustion chamber) the axial and radial dimensions of which are small compared to the chamber diameter. Typically, the stability is determined as a function of the axial and radial location of this annular volume, in effect assuming that the various regions of the chamber act independently with regard to combustion oscillations.

The model shows that the most sensitive zone is where the steady-state ΔV is minimum, usually near the injector face. A typical stability map based on vaporization theory is shown in Fig. 8. The region inside the stability boundary represents a condition where a pressure oscillation will diverge to an equilibrium value, represented by the upper dotted line. As ΔV increases, the region of instability recedes. The curves also indicate that minimum stability occurs at a certain value of burning rate. This suggests that there is a particular drop size which will result in minimum stability.[46]

Several improvements and extensions of the basic model have been published. The computational techniques have been simplified, and the monopropellant character of amine propellants taken into account.[42] Experimental and analytical studies have been performed to correlate the effects of design variables with stability results.[44] The effects of droplet drag, propellant spray distribution, and independent addition of heat and mass have been evaluated.[61] Limited studies using two space dimensions have been performed, showing qualitative agreement with the one-dimensional analyses (see Sec. 4.3.3 of Ref. 53). The model is limited to liquid propellants as a result of dependence upon vaporization rates. Extension to consider supercritical chamber pressures and transient response has been accomplished, however.[62] The primary limitation to use of the model is lack of knowledge of droplet size distribution in a given system. Even so, it has become a widely accepted tool for stability predictions.

A simplified approach for determining the potential for instability has been suggested.[55] This method has generally been found to agree with the Sensitive Time Lag and Droplet Vaporization results.[56] It is based upon an analysis of the unsteady burning of a single droplet in an oscillating pressure field. A combustion gain parameter is defined

$$N_s = A(nd^3)fP_c/w \tag{1}$$

such that the droplet burning rate oscillation is in phase with the pressure oscillation of $5.6 < N_s < 30$. Hence, when N_s falls within that range the combustor is potentially unstable. In Eq. (1), the constant A is an empirical parameter, which must be estimated from existing instability test experience. However, it has been found that the value of A, although somewhat dependent on propellant combination, is between 7 and 10, and thus does not introduce a large uncertainty into the predicted stability behavior.

Stability Fixes

The accuracy of stability predictions is limited by the random nature of the injection and combustion processes and the primarily meta-stable nature of the instability phenomenon (i.e., a finite disturbance is required to initiate it). Stability analysis can, however, discern the probability of sensitivity to the various modes of instability. Knowledge of the most probable mode or modes provides direction for making the physical changes necessary to reduce sensitivity to the critical frequencies. These changes must a) reduce the energy input to the combustion wave by altering the combustion time lag or burning rate, b) increase the ability of the system to absorb and dissipate the energy, or c) change the resonant frequencies of the chamber. Stability fixes are either chemical, aerodynamic or mechanical in nature. The first two operate by changing the energy release profile in the chamber. The latter type affects either the resonant frequencies or the energy absorption rate.

Chemical additives offer the advantage that they can be applied without changing the hardware design. Addition of 10.9% of Hybaline A_{14} was found to suppress instability in a LO_2/RP engine.[63] Ammonium Nitrate has been used to stabilize N_2O_4/N_2H_4 engines.[64] The suggested mechanism was that the additive in concentrations of 5–10% increases droplet shattering, thus changing the energy release profile. The damping rate was found to increase by 100–200%. Less than 1% additive had no effect. Aluminum particles were found to stabilize an experimental O_2/H_2 combustor.[65] One-half percent aided stability over a wide range, but less than 0.22% tended to destabilize the system. In another study,[66] the threshold for stability improvement using aluminum particles was 20% concentration. Near the threshold concentration, reduction in particle size from 6 μ to 3 μ was destabilizing. O/F increase raised the threshold level, but no effect of P_c variation was noted up to 500 psig.

Fixes which adjust the aerodynamic environment in the chamber require hardware changes. Addition of a tangential flow has been found to amplify or damp transverse oscillations depending upon the phase relationship to the combustion wave.[67] Radially nonuniform mass injection distributions have been used to stabilize large engines.[68] Varying the injector distribution changes combustion profile, which can reduce the amplitude of the pressure response to a disturbance. Changes in injector or element design, when permissible, can have the greatest effect upon stability margin.[33] Variations in orifice size, location, impingement angle or in element type will result in significant changes in energy release profile. As noted previously, changes which tend to distribute the energy release axially and move the $\Delta V = 0$ point away from the injector will tend to stabilize the system. An increase in chamber A_c/A_t will reduce gas velocity, thus slowing the vaporization rate. The energy release will then be distributed over a longer axial distance, tending to improve stability.

Mechanical fixes can generally be applied with minimum change to the geometry of the injector or chamber. Two devices which have been very effective in controlling stability are baffles and acoustic resonators. Baffles (Fig. 9) are effective against radial or tangential instability modes. They effectively form compartments within the sensitive zone near the injector face. These compartments exhibit different resonant frequencies than the unbaffled chamber, thus preventing a sustained oscillation.

The theory and practice of acoustic resonators (Fig. 9) has received a great deal of recent attention. Resonators may be placed in either the injector face of the chamber and provide effective damping. A general discussion of the theory of acoustic liners and their application is given in Ref. 33. In another investigation,[69] a description of acoustic liner theory is presented with a discussion of the results of use of these devices. Evaluation of the placement of cavities and analysis

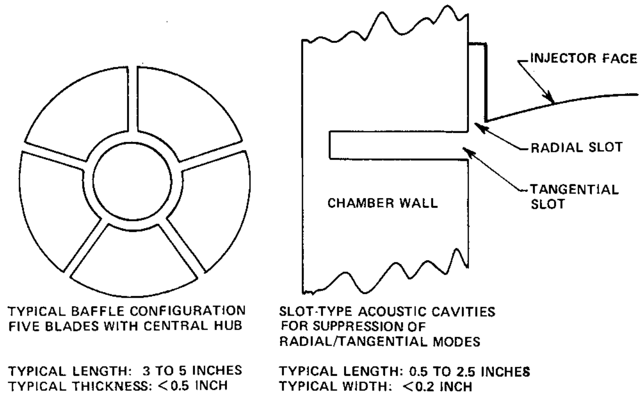

Fig. 9 Baffles and acoustic resonators.

of admittance characteristics has also been accomplished.[70] It has been shown that baffles can be replaced by acoustic liners and result in a dynamically stable system.[71]

Wall Compatibility

Wall Compatibility is a term which has relative rather than absolute meaning. Any combustion chamber which is capable of meeting its design operating duration is considered compatible. All exposed surfaces must meet the durability requirement, so the injector face, baffles, or any other device which extends into the chamber have to be adequately protected.

The durability of the thrust chamber represents a balance between the destructive forces in the combustion volume and the ability of the wall materials to withstand these forces. Three components of the chamber environment may be identified: thermal, chemical, and gas dynamic. The thermal forces are determined by the temperature of the gases near the wall and the heat flux into the wall. Chemical damage is related to the reactivity of the wall material with the propellants or reaction products in contact with it. Gas dynamic forces can cause severe erosion of the wall and may be the dominant factor in compatibility in some cases.[72]

There are interactions between the three primary components of chamber environment. The rate of chemical reactions is significant at typical combustion temperatures, but may be negligible a few hundred degrees lower. The chemical and thermal wall environment is often controlled by containing a hot core flow within a cooler boundary flow of fuel-rich gases. It is also obvious that gas dynamic forces would be insignificant in the absence of high temperatures. The erosive nature of radial or tangential flows is not a result of the physical forces, but rather an increase in heat transfer coefficients, reduction of boundary-layer thickness and transport of adverse chemical species to the wall. The importance of controlling transverse flows is based upon the need to minimize these interacting effects.

The relative importance of the three environmental components is determined by the type of chamber, which is in turn influenced by mission requirements. Engines for long duration service requiring relatively consistent operation may require geometric stability over the entire operating life. These systems would require active cooling with no throat or wall erosion permissible. If, for example, regenerative cooling is applied, the primary concern for compatibility would be the thermal design. If the entire wall is maintained at a temperature below the point where structural strength diminishes, and if the burn-out margins are satisfactory, the system will be compatible. Chemical reaction is unlikely to occur between the metal wall and typical oxidizers at this itemperature level. Steady state transverse flows are not important if the thermal conditions are met.

Conversely, a graphite chamber or component might be totally insensitive to the temperature level, but subject to chemical attack, particularly if oxygen is present, at temperatures as low as 1200°F. An ablative chamber is subject to damage from all three sources, with rates of attack generally increasing as ablation occurs and the wall becomes roughened. When surface recession takes place, radial outflow of main core gases occurs to a greater extent. This results in a reduction in boundary-layer thickness and possible chemical attack as well.

Factors Influencing Compatibility

Development of liquid rocket engines has consistently tended toward higher energy propellant systems. Combustion temperatures over 6000°F are common, much higher than can be withstood by conventional structure materials. Consequently, the problem of compatibility has required more attention in recent years.

The choice of propellant system is thus a significant factor influencing the achievement of compatibility. The thermodynamic potential of the system determines the driving force for thermal damage. Kinetic factors can become important as well. The pyrolysis reactions of ablative materials vary with temperature and yield different products.[73]

The geometry of the chamber also affects compatibility. Sudden inward turning of gases can result in radial outflows downstream of the turn, and near stagnation at the point of turning. The chamber A_c/A_t determines gas velocity in the zone where energy release is highest, near the injector. Wall roughness in ablative chambers may become self-accelerating, as the gases will tend to stay attached to the wall. If a depression occurs, gases flowing into it will be turned in the radial direction and may stagnate at the base of the depression. The increase in temperature and gas dynamic force will cause the depression to grow rapidly.

Distribution of propellants near the injector face and vaporization rates will influence this energy release profile. Distribution of heat, and unreacted propellants has been shown to be directly relatable to erosion.[74] Elements should be designed to provide similar vaporization rates for each propellant. Stream separation with hypergolic propellants can result in local O/F distribution which affects compatibility. The degree of separation depends upon element orientation as well as the pressure and temperature of the injected propellants.[23,75] Figure 10 illustrates some typical spray fan distributions.

Design of the injector probably influences compatibility more than any other single factor. Uneven mass flux or energy release distribution results in transverse flows. The force of lateral gas movement can cause misimpingement of element jets, sometimes permitting oxidizer to spray directly on the wall or in other chemically sensitive areas.[26] It is highly probable that most rocket engines suffer a significant percentage of elements that impinge improperly or not at all.

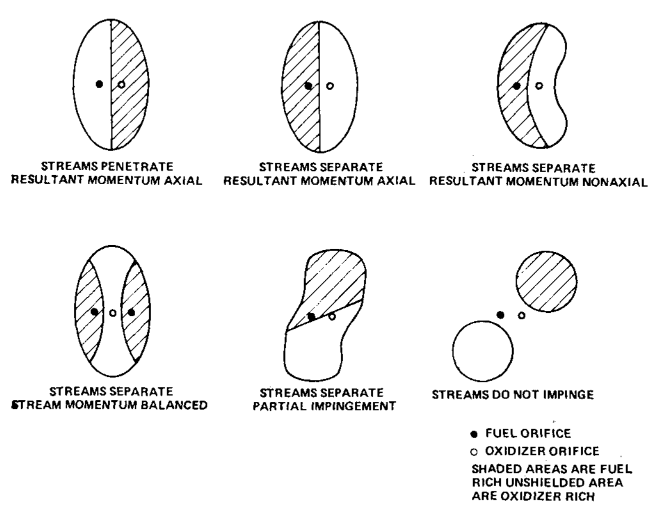

Fig. 10 Typical spray fan distribution.

Compatibility Analysis Techniques

Compatibility analysis has progressed from evaluation of material response to a carefully controlled thermal input to sophisticated techniques which consider chemical interaction and a three-dimensional flowfield. In a recently completed study,[76] all of the factors believed to be of significance to compatibility were considered in a calculational method using a system of seven computer programs.

Methods for evaluation of thermal and chemical effects are numerous. Where thermal effects are dominant, flame temperature and P_c were found to be the most significant gas properties.[77] Thermal conductivity and density are the controlling material properties. The ASTHMA (Axisymetric Transient Heating Materials Ablation) program predicts temperature history in depth and surface recession for a two-dimensional assymetric noncharring material.[78]

A program entitled ACE (Aerotherm Chemical Equilibrium) calculates surface thermochemical response with kinetically controlled surface reactions.[78] This study was based upon graphitic materials where pyrolysis was not a factor. The 2D-ABLATE (two-dimensional ablation) program includes chemical reaction, transient ablation and heat conduction. Recent improvements[79] permit consideration of anisotropic materials, more than one charring material and irradiation at the heated surface.

In another report,[80] four computer programs were used for prediction of erosion rates and thermal response, including chemical effects. The technique applies to both charring and non-charring materials. Either thermochemical equilibrium or kinetically-controlled surface reaction may be considered to evaluate thermal response and ablation rates. Results with methods for considering thermal and chemical effects are limited by the accuracy of experimental heat transfer data and thermodynamic input data.[81] They can also be affected by local variations in gas dynamic flow patterns.

A program entitled LISP (Liquid Injector Spray Patterns) may be used to calculate mass and O/F distribution downstream of the injector,[82] giving a three-dimensional evaluation of the gas flowfield. Some experimental results of the effects of transverse flow on compatibility are given in Ref. 72. This study used a geometric method to estimate the magnitude and direction of crossflows for a given injector design. Variations in chemical composition within stream tubes was also considered to define critical zones at the wall. The most complete program currently in use evaluates injector effects on wall response.[76] Using known mass and O/F distribution, a three-dimensional gas flowfield is evaluated considering droplet vaporization as the limiting process. Heat transfer to the wall is determined considering the non-uniform flowfield and spray impingement on the wall. Finally, charring and erosion rates are calculated.

Performance/Stability/Compatibility Interactions

Knowledge and understanding of the interactions between the factors influencing performance, stability and compatibility are the keys to minimizing development requirements. There has been general recognition of some basic interactions for many years. One such example is the fact that high performing engines are often unstable and incompatible. It has often been observed that a fix applied to one problem area leads to another for seemingly inexplicable reasons. Droplet vaporization theory appears to provide a mechanism for most of these observations, permitting understanding of the interrelated effects.

Random pressure fluctuations (combustion roughness) appear to have a generally bad effect upon thrust chamber operation. If several random events occur in phase, they may trigger an instability. The turbulence caused by pressure variations generally results in poorer compatibility because of the reduced boundary-layer thickness and higher transfer coefficients. Rough combustion may affect performance either way. The turbulence improves mixing which would raise I_s in chambers which are length limited. Lateral flows may cause imperfect impingement, however, reducing the degree of droplet atomization. Even this effect may be offset by secondary shattering or stripping due to transverse flow.

Low-frequency instability may act similarly to rough combustion in providing a trigger for high-frequency instability. The degree of droplet atomization in an LO_2/GH_2 engine was found to increase significantly when feed system coupled instability occurred.[83] The higher degree of atomization leads to higher energy release near the injector. Periodic oscillations combined with the normal random fluctuations may provide the initial disturbance required to trigger high-frequency instability in modes where sensitivity exists. For gas/gas systems, feed system coupling may be a factor in high frequency instability.[36] Coupling of flow and pressure oscillations increases with propellant density.

When instability occurs, the I_s may either increase or decrease. The passage of a transverse wave over an evaporating droplet will result in the release of additional mass.[84] Transverse flow can result in highly atomized propellants with much different energy release rates than that produced by impingement.[43] Experimental results with low performing units showed a significant I_s increase when instability occurred.[85] With efficient combustors, however, combustion oscillations increased the length required to achieve essentially complete combustion.[86]

Instability in an LO/GH_2 combustor caused the controlling factor in the combustion to switch from turbulent mixing to vaporization.[87] In this case, the oxidizer was injected through showerhead elements into the gaseous fuel. Installation of a flow restrictor increased performance of a GO_2/heptane engine.[15] Droplet breakup caused by induced turbulence was thought to be the mechanism.

Controlled oscillations have been found to increase measured heat transfer rates. Oscillations in the 34–680 Hz range increased heat transfer coefficients by up to 65% in an air-water system through increased turbulence.[88] Transition from laminar to turbulent flow in the boundary layer caused an increase in heat transfer coefficient in another system when a 100 Hz oscillation was imposed.[89] Four-fold increases in heat transfer with 280 psi peak-peak tangential mode oscillations were observed in a 150 psi LOX/ethanol engine.[90] Heat transfer more than doubled at the mid-length for a similar amplitude longitudinal mode instability. Average drop size may also be reduced by imposed oscillations. A reduction from an average drop size of 217 μ to 62 μ was noted as a result of 1190 Hz oscillations.[14]

Changes in operating conditions on existing engines can lead to unexpected effects. An increase in propellant temperature would be expected to increase I_s to the extent the enthalpy of the reaction products is increased. Greater effects have been observed, however, than can be explained by enthalpy alone.[91] The difference may be explained by improved atomization resulting from reduced surface tension of the propellants at the higher temperature. Reduction in O/F led to increased performance in a fuel vaporization limited system.[91] It was believed that increased fuel flow led to a better atomization. In another engine, reduction in O/F led to severe degradation in compatibility even though the thermal and chemical environments were directionally better.[72] The increase in lateral flow due to unbalanced stream momentum was found to be the cause. Variations in flow rates, either up or down, may result in unstable orifice flow.[25] If hydraulic flip occurs, a 10–20% reduction in C_d may result.[26] This can lead to serious compatibility problems as a result of maldistribution of propellants, and may possibly lead to unstable operation as well.

The use of acoustic liners for stability control may eliminate the interactions between stability and performance.[33] If liners can provide sufficient damping to prevent pressure oscillations at sensitive frequencies from being amplified, then injector design can be optimized without consideration of stability effects.

Use of Analysis Methods for Thrust Chamber Design

The goal of an engine development program is to produce the best compromise between performance, stability, and compatibility consistent with specified design constants. Typical constraints are the propellant system, pressure schedule, duration requirements and engine envelope size. Other such as over-all O/F, minimum I_s, transient impulse response, maximum weight and cost are often applied. It is understood that the engine must be stable and compatible within these specifications.

The first task is to establish the relative importance of the various constraints. Some may be inviolate, others more flexible. Many tradeoffs are possible in the predesign phase. Stability margins may be increased at the expense of I_s. Film or barrier cooling may be provided for compatibility, again at the expense of I_s. Throat location may be adjusted to take maximum advantage of the envelope by designing elements which will permit complete vaporization in the available length. Fuel and oxidizer manifolds may be reduced in volume for improved transient performance, but with increased ΔP and probable maldistribution of O/F. Large elements and simple flow passages will result in reduced manufacturing cost, generally with a negative effect on I_s. It is highly desirable that the final design be operated at its design point. Requirements, for example, of additional coolant flow with a fixed design may result in serious degradation of the engine operation. Removal of fuel from the main core will result in reorientation of oxidizer rich element resultant vectors toward the wall, thus creating an even greater coolant requirement.

Preliminary injector/chamber design is the next step in the development process. Elements should be chosen to provide essentially similar vaporization distribution for that element. Chamber A_c/A_t should be low enough to provide sufficient ΔV to insure that vaporization is complete upstream of the throat. Below about 1.5, however, boundary-layer losses in the chamber may become excessive. The chamber length should be at least 30 times the element spacing to avoid possible I_s loss due to incomplete mixing.

Hydraulic analysis of the feed system and preliminary injector/chamber design will provide mass flux and O/F distribution. Combined with droplet vaporization profiles, this determines the energy release profile for the chamber, both axially and radially. This information provides the basis for stability and compatibility analysis. Weaknesses in these areas can be resolved by iteration of the injector pattern until a satisfactory compromise is reached.

Stability can be improved without degradation of I_s if acoustic liners are applied. Baffles also improve stability, but normally require changes in the injector pattern which reduce I_s. Compatibility may be improved without affecting I_s if improvements in materials or regenerative cooling techniques are the basis. Subsequent to all changes, the final design should be adjusted to insure proper element momentum ratio and resultant vectors within the final hydraulic design. Ideally, this might require a large variety of orifice sizes for direct control of mass and O/F distribution, particularly if feed manifold ΔP is high. In practice, satisfactory control can be achieved with few orifice size changes by controlling element spacing.

To apply the specific analysis techniques considered earlier to a given design, the two basic requirements are hydraulic analysis and atomized drop size distribution. The former can be calculated from design dimensions, or evaluated by cold flow testing. The latter can rarely be simulated satisfactorily in cold flow and can best be estimated by correlation with results of previous tests with similar element configurations. If such data are not available, the published techniques of the previously cited literature may be used as a first approximation.

Through the methodology which has become available in the last few years, it is now possible to predict performance, stability, and compatibility characteristics from a mechanical design with little or no test data. Accuracy of the available methods is such that convergence to the optimum design should require no more than two or three design iterations. These significant technological advances will undoubtedly result in reduced costs for future engine developments compared to what would have been possible as recently as two years ago.

References

[1] Priem, R. J. Heidmann, M. R., "Propellant Vaporization as a Design Criteria for Rocket Engine Combustion Chambers," TR R-67, 1960, NASA.

[2] Pieper, J. L., "ICRPG Liquid Propellant Thrust Chamber Performance Evaluation Manual," Rept. 178, Sept. 1968, Chemical Propulsion Information Agency, Silver Spring, Md.

[3] McKevitt, F. X. and Walsh, T. J., "Space Engine Performance Predictions," *Journal of Spacecraft and Rockets*, Vol. 2, No. 3, May–June 1965, pp. 374–383.

[4] Valentine, R. S., Dean, L. E., and Pieper, J. L., "An Improved Method for Rocket Engine Performance Prediction," *Journal of Spacecraft and Rockets*, Vol. 3, No. 9, Sept. 1966, pp. 1409–1414.

[5] Pieper, J. L., Dean, L. E., and Valentine, R. S., "Effects of Chamber energy Release Efficiency upon Nozzle, Performance," *Journal of Spacecraft and Rockets*, Vol. 4, No. 5, May, 1967, pp. 700–702.

[6] Branch, M. C., and Sawyer, R. F., "Nonequilibrium Combustion and Nozzle Flow in Propellant Performance," *Journal of Spacecraft and Rockets*, Vol. 6, No. 11, Nov., 1965, pp. 1322–1325.

[7] Adams, D. M., "Theoretical Assessment of Some Combustion and Expansion Problems in Rocket Motors," *Journal of Spacecraft and Rockets*, Vol. 5, No. 2, Feb., 1968, pp. 173–178.

[8] Priem, R. J., "Liquid Rocket Combustion—An Aerodynamically Controlled Process," TMX-52334, Sept. 1967, NASA.

[9] Hersch, M., "A Mixing Model for Rocket Engine Combustion," TN D-2881, June 1965, NASA.

[10] Hersch, M., "Experimental Method of Measuring Intensity of Turbulence in a Rocket Chamber," *ARS Journal*, Vol. 31, No. 1, Jan. 1961, pp. 39–45.

[11] Pieper, J. L., Dean, L. E., and Valentine, R. S., "Mixture Ratio Distribution—Its Impact on Rocket Thrust Chamber Performance," *Journal of Spacecraft and Rockets*, Vol. 4, No. 6, June 1967, pp. 786–789.

[12] Priem, R. J., "Liquid Rocket Combustion—An Aerodynamically Controlled Process," *XVIII International Astronautical Congress*, Pergamon Press, New York, 1968, pp. 529–537.

[13] Morrell, G., "Critical Conditions for Drop and Jet Shattering," TN D-677, Feb. 1961, NASA.

[14] Ingebo, R. D., "Atomization of Ethanol Jets in a Combustor with Oscillatory Combustion Gas Flow," TN D-3513, July 1966, NASA.

[15] Heidmann, M. R., "Experimental Effect of Gas Flow Transients on the Heat Release of Burning Liquid Drops in a Rocket Combustor," TN D-1576, March 1963, NASA.

[16] El Wakil, M. M., Priem, R. J., Brikowski, H. J., Myers, P. S., and Uyehara, O. A., "Experimental and Calculated Temperature and Mass Histories of Vaporizing Fuel Drops," TN 3490, Jan. 1956, NASA.

[17] El Wakil, M. M., Uyehara, O. A., and Myers, P. S., "A Theoretical Investigation of the Heating-up Period of Injected Fuel Droplets Vaporizing in Air," TN 3179, May 1954, NASA.

[18] Manrique, J. A., "Theory of Droplet Vaporization in the Region of the Thermodynamic Critical Point," CR-72558, June 1969, NASA.

[19] Riebling, R. W., "Criteria for Optimum Propellant Mixing in Impinging-Jet Injection Elements," *Journal of Spacecraft and Rockets*, Vol. 4, No. 6, June 1967, pp. 817–819.

[20] Lawver, B. R. and Breen, B. P., "An Experimental Study of the N_2O_4/N_2H_4 Jet Separation Phenomena," Final Report, NASA-CR-72444, Sept, 1968, Dynamic Science, Monrovia, Calif.

[21] Rupe, J., Dipprey, D., Kushida, R., and Clayton, R., "The Stability Criterion for Impinging Jets and its Impact on Combustion," Publ. 192, Vol. I, 1969, Chemical Propulsion Information Agency, Silver Spring, Md., p. 1.

[22] Campbell, D. T., Clive, G. L., Cordill, J. D., Proffitt, R. L., and Sloat, T. N., "Reactive Stream Separation Photography," Rept. R-8110, Jan. 1970, Rocketdyne, Canoga Park, Calif.

[23] Kushida, R. and Houseman, J., "Criteria for Separation of Impinging Streams of Hypergolic Propellants," TM 33-395, July 1968, Jet Propulsion Lab. Pasadena, Calif.

[24] Evans, D. C. and Riebling, R. W., "The Effect of Injector Element Scale on the Mixing and Combustion of N_2O_4/N_2H_4 Propellant," TR 32-1178, Nov. 1967, Jet Propulsion Lab., Pasadena, Calif.

[25] Riebling, R. W. and Powell, W. B., "The Hydraulic Characteristics of Flowthrough Miniature Slot Orifices," TR 32-1397, Sept. 1969, Jet Propulsion Lab., Pasadena, Calif.

[26] Valentine, R. S., Rossi, F. S., and Kromrey, R. V., "Effect of Fluid Dynamic Phenomena on Spontaneous Pressure Spikes on the Apollo SPS Thrust Chamber," *Journal of Spacecraft and Rockets*, Vol. 5, No. 1, Jan. 1968, pp. 31–35.

[27] Rao, G. V. R., "Exhaust Nozzle Contour for Optimum Thrust," *Jet Propulsion* Vol. 28, July 1958, pp. 377–382.

[28] Powell, W. B., "ICRPG Liquid Propellant Thrust Chamber Performance Evaluations Methodology," *Journal of Spacecraft and Rockets*, Vol. 7, No. 1, Jan. 1970, pp. 105–107.

[29] Kors, D. L., Bassham, L. B., and Walker, R. E., "A Liquid Rocket Performance Model Based on Vaporization Interactions," *Journal of Spacecraft and Rockets*, Vol. 6, No. 10, Oct. 1969, pp. 1133–1138.

[30] "Handbook of Recommended Practices for Measurement of Liquid Propellant Rocket Engine Parameters," Doc. 179, Jan. 1969, Chemical Propulsion Information Agency, Silver Spring, Md.

[31] Colbert, D. L., Powell, B. D., and Abernethy, R. B., "ICRPG Measurment Uncertainty Model for Liquid Rockets," *Journal of Spacecraft and Rockets*, Vol. 7, No. 1, Jan. 1970, pp. 95–97.

[32] "ICRPG Handbook for Estimating the Uncertainty Measurements Made with Liquid Propellant Rocket Engine Systems," Doc. 180, April 1969, Chemical Propulsion Information Agency, Silver Spring, Md.

[33] Weiss, R. R., "An Introduction to Combustion Instability in Liquid Propellant Rocket Engines," RPL TR-66-150, July 1966, Rocket Propulsion Lab., U.S. Air Force Systems Command, Research and Technology Div.

[34] Crocco, L. and Cheng, S.-I, *Theory of Combustion Instability in Liquid Propellant Rocket Motors*, Butterworth, London, 1956.

[35] Nestlerode, J. A., Chadwick, W. D., and Gleason, J. L., "Investigation and Suppression of Rocket Feed System Oscillation," *ICRPG Third Combustion Conference*, Publ. 138, Vol. I, Chemical Propulsion Information Agency, Silver Spring, 1967, p. 341.

[36] Heidmann, M. F., Sokolowski, D. E., and Diehl, L. A., "Study of Chugging Instability with LOX and GH_2 Combustors," TN D-4005, June 1967, NASA.

[37] Feiler, C. E. and Heidmann, M. F., "Dynamic Response of GH_2 Flow Systems and Its Application to High Frequency Combustion Instability," TN D-4040, June 1967, NASA.

[38] Priem, R. J. and Morrell, G., "Application of Similarity Parameters for Correlating High Frequency Instability Behavior of Liquid Propellant Combustors," *Progress in Astronautics and Rocketry: Detonation and Two-Phase Flow*, Vol. 6, edited by S. S. Penner, and F. A. Williams, Academic Press, New York, 1962, pp. 305–320.

[39] Smith Jr., A. J. and Reardon, F. H., "The Sensitive Time Lag Theory and Its Application to Liquid Rocket Combustion Instability Problems," AFRPL TR-67-314, Vol. 1, March 1968, Air Force Rocket Propulsion Lab. Edwards Air Force Base, Calif.

[40] Beltran, M. R. and Frankel, N. A., "Prediction of Instability Zones in Liquid Rocket Engines," *AIAA Journal*, Vol. 3, No. 3, March 1965, pp. 516–518.

[41] Clayton, R. M., Rogero, R. S., and Sotter, J. G., "An Experimental Description of Destructive Liquid Rocket Resonant Combustion," *AIAA Journal*, Vol. 6, No. 7, July 1968, pp. 1252–1259.

[42] Beltran, M. R., Breen, B. P., Hoffman, R. J., Kosvic, T.C., Sanders, C. F., and Wright, R. O., "Analysis of Liquid Rocket

Engine Combustion Instability," AFRPL-TR-65-254, Jan. 1966, Air Force Rocket Propulsion Lab., Edwards Air Force Base Calif.

[43] Clayton, R. M., "The Influence of Several Near-Wall Injection Conditions on the Combustion Performance of a Liquid Rocket Engine," TR 32-1283, Sept. 1968, Jet Propulsion Lab., Pasadena, Calif.

[44] Abbe, C. J., McLaughlin, C. W., and Weiss, R. R., "Influence of Storable Propellant Liquid Rocket Design Parameters on Combustion Instability," AFRPL TR-69-14, Jan. 1969, Air Force Rocket Propulsion Lab., Edwards Air Force Base, Calif.

[45] Morrell, G., "Breakup of Liquid Jets by Transverse Shocks," *Eighth Symposium on Combustion*, Williams and Wilkins, Baltimore, Md, 1962, pp. 1059–1068.

[46] Beltran, M. R., Breen, B. P., Kosvic, T. C., Sanders, C. F., Hoffman, R. J., and Wright, R. O., "Liquid Rocket Engine Combustion Instability Studies," Dynamic Science Final Report, Contract AF 04(611)-10542, AFRPL-TR-66-125, July 1966, Air Force Rocket Propulsion Lab., Edwards Air Force Base, Calif.

[47] Heidmann, M. R., "Oscillatory Combustion of a Liquid Oxygen Jet with Gaseous Hydrogen," TN D-2753, March 1965, NASA.

[48] Heidmann, M. F., "Effect of Steady Velocities on the Dynamic Response of Liquid Jet Atomization," Publ. 192, Vol. I, Chemical Propulsion Information Agency, Silver Spring, 1969, p. 121.

[49] Wieber, P. R., "Calculated Temperature Histories of Vaporizing Droplets to the Critical Point," *AIAA Journal*, Vol. I, No. 12, Dec. 1963, pp. 2764–2770.

[50] Tobey, A. C. and Bastress, E. K., "Flame-Piloting Mechanisms in Liquid Propellant Rocket Engines," AFOSR 67-0938, Oct. 1966, Air Force Office of Scientific Research, Arlington, Va.

[51] Bowman, C. T. and Glassman, I., "Longitudinal Combustion Instability Studies Using a Gaseous Propellant Rocket Motor," *First ICRPG Combustion Instability Conference*, Publ. 68, Chemical Propulsion Information Agency, Silver Spring, Md., Jan. 1965, pp. 125–138.

[52] Teague, A. J., *Optimized SPS Injector Development and Verification Program*, Final Report 108581, Contract NAS 9-8285, April 1971, Aerojet Liquid Rocket Co., Sacramento, Calif.

[53] Harrje, D. T., ed., "*Liquid Propellant Rocket Combustion Instability*," NASA Special Publication, 1972, to be published.

[54] Priem, R. J. and Guentert, D. C., "Combustion Instability Limits Determined by a Nonlinear Theory and a One-Dimensional Mode," TN D-1409, Oct. 1962, NASA.

[55] Dykema, O. W., "An Engineering Approach to Combustion Instability," Doc. 105, 1966 Chemical Propulsion Information Agency, Silver Spring, Md., p. 205.

[56] Dykema, O. W., "Application of an Engineering Approach to the Stability of a Proposed Advanced Agena Engine," Publ. 192, Vol. I, 1969, Chemical Propulsion Information Agency, Silver Spring, Md., p. 343.

[57] Summerfield, M., "A Theory of Unstable Combustion in Liquid Propellant Rocket Systems," *ARS Journal*, Vol. 21, No. 5, Sept. 1951, pp. 108–114.

[58] Sabersky, R. H., "Effect of Wave Propagation in Feed Lines on Low Frequency Rocket Instability," *Jet Propulsion*, Vol. 24, 1964, p. 172.

[59] Crocco, L., "The Relevance of a Characteristic Time in Combustion Instability," Second ICRPG Combustion Conference, Publ. 105, Vol. I, May 1966, Chemical Propulsion Information Agency, Silver Spring, Md., p.115.

[60] Waugh, R. C., "Stability Characterization of Advanced Injectors," Rept. 20672-P2D (NAS 8-20672), Jan. 1970, NASA.

[61] Hoffman, R. J., Wright, R. O., and Breen, B. P., "Combustion Instability Prediction Using a Nonlinear Bipropellant Vaporization Model," Final Report, NAS 7-442, May 1967, Dynamic Science, Monrovia, Calif.

[62] Campbell, D. T. and Chadwick, W. D., "Combustion Instability Analysis at High Chamber Pressure," AFRPL-TR-68-179, Aug. 1968, Rocketdyne, Canoga Park, Calif.

[63] Weiss, R. R., "Investigation of Hybaline A_{14} as a Combustion Instability Suppressant in a LO_2/RP-1 Combustion System," AFRPL-TR-66-130, June 1966, Air Force Rocket Propulsion Lab., Edwards Air Force Base, Calif.

[64] Kosvic, T. C. and Breen, B. P., "Study of Additive Effects on Hydrazine Combustion and Combustion Stability at High Pressure," AFRPL-TR-69-12, Nov. 1969, Contract FO4611-68-C-0027, Air Force Rocket Propulsion Lab., Edwards Air Force Base, Calif.

[65] Heidmann, M. F. and Povinelli, L. A., "An Experiment on Particle Dropping in a Two-Dimensional H_2/O_2 Combustor," TMX-52359, 1967, NASA.

[66] Kesselring, R. C. and Oberg, C. L., "Combustion Stability of Metallized Gelled Propellants," AFRPL-TR-164, May 1967, Rocketdyne, Canoga Park, Calif.

[67] Heidmann, M. F. and Feiler, C. E., "Evaluation of Tangential Velocity Effects on Spinning Transverse Combustion Instability," TN D-3406, May 1966, NASA.

[68] Reardon, F. H., McBride, J. M. and Smith Jr., A. J., "Effect of Injection Distribution on Combustion Stability," *AIAA Journal*, Vol. 4, No. 3, March 1966, pp. 506–512.

[69] Priem, R. J. and Rice, E. J., "Combustion Instability with Finite Mach Number Flow and Acoustic Liners," *Twelfth Symposium on Combustion*, The Combustion Inst., Pittsburgh, Pa., 1969, pp. 149–159.

[70] Crocco, L. and Harrje, D. T., "Nonlinear Aspects of Combustion Instability in Liquid Propellant Rocket Motors," Publ. 68, Jan. 1965, Chemical Propulsion Information Agency, Silver Spring, Md., pp. 47–72.

[71] Oberg, C. L., "LM Ascent Engine Acoustic Cavity Study," Publication No. 192, Vol. I, 1969, Chemical Propulsion Information Agency, Silver Spring, Md., p. 303.

[72] Conn, T. E., Hester, J. N., and Valentine, R. S., "Environmental Effects upon Rocket Injector/Chamber Compatibility," *Journal of Spacecraft and Rockets*, Vol. 4, No. 12, Dec. 1967, pp. 1581–1585.

[73] Ladacki, M., Hamilton, J. V., and Cohz, S. N., "Heat of Pyrolysis of Resin in Silica-Phenolic Ablation," *AIAA Journal*, Vol. 4, No. 10, Oct. 1966, pp. 1798–1802.

[74] Rupe, J. H. and Evans, D. C., "Designing for Compatibility in High Performance LP Engines," *Astronautics & Aeronautics*, Vol. 3, No. 1, Jan. 1965, pp. 68–74.

[75] Zung, L. G. and Breen, B. P., "Hypergolic Impingement Mechanisms and Criteria for Jet Mixing or Separation," Publ. 192, Vol. I, 1969, Chemical Propulsion Information Agency, Silver Spring, Md., p. 21.

[76] Hines, W. S., Combs, L. P., Ford, W. M., and Van Wyk, R., "Development of Injector Chamber Compatibility Analysis," AFRPL-TR-70-12, March 1970, Air Force Rocket Propulsion Lab., Edwards Air Force Base, Calif.

[77] Tick, S. J., Huson, G. R., and Griese, R., "Design of Ablative Thrust Chambers and Thin Materials," *Journal of Spacecraft and Rockets*, Vol. 2, No. 3, May-June 1965, pp. 325–331.

[78] Moyer, C. B., "Two Dimensional Transient Heating and Surface Thermochemistry Computer Program," AFRPL-TR-68-116, July 1968, Aerotherm Corp., Palo Alto, Calif.

[79] Friedman, H. S., Hines, W. S., and Cunial, G. D., "Designer's Guide and Computer Program for Ablative Materials in Liquid Rocket Thrust Chambers," AFRPL-TR-67-159, June 1967, Rocketdyne, Canoga Park, Calif.

[80] McCuen, P. A., Schaefer, J. M., Lundberg, R. E. and Kendall, R. M., "A Study of Solid Propellant Rocket Motor Exposed Materials Behavior," Final Report AF04(611)-9073, Feb. 1965, Air Force Materials Lab., Wright-Patterson Air Force Base, Ohio.

[81] Andrepont, W. C., "Evaluation of Ablation Prediction Techniques for Solid Rocket Nozzle Materials," AFRPL-TR-68-142, Aug. 1968, Air Force Rocket Propulsion Lab., Edwards, Calif.

[82] Hines, W. S., Hyde, J. C., Chadwick, W. D., and Ford, W. M., "Analytical Description of Injected Propellant Spray Mass Distribution," Publ. 192, Vol. I, 1969, Chemical Propulsion Information Agency, Silver Spring, Md., p. 61.

[83] Heidmann, M. R., "Oxygen Jet Behavior During Combustion Instability in a Two-Dimensional Combustor," TN D-2725, March 1965, NASA.

[84] Sotter, J. G., "Nonsteady Evaporation of Liquid Propellant Drops: The Grossman Model," TR-32-1061, Jan. 1968, Jet Propulsion Lab., Pasadena, Calif.

[85] Heidmann, M. F., "A Study of Injection Processes for 15% Fluorine-85% Oxygen and Heptane in a 200-lb Thrust Rocket Engine," RM E56J11, Jan. 1957, NASA.

[86] Burrows, M. C. and Ragner, R., "Relation of Emitted Ultraviolet Radiation to Combustion of Hydrogen and Oxygen at 20 Atmospheres," TN D-2548, Dec. 1964, NASA.

[87] Burrows, M. C., "Radiation Processes Related to Oxygen-Hydrogen Combustion at High Pressures," *Tenth Symposium on Combustion*, The Combustion Inst., Pittsburgh, Pa., 1965, pp. 207–215.

[88] Feiler, C. E. and Yeager, E. B., "Effect of Large Amplitude Oscillations on Heat Transfer," TR-R-142, 1963, NASA.

[89] Feiler, C. G., "Experimental Heat Transfer and Boundary Layer Behavior with 100 CPS Flow Oscillations," TN D-2521, Dec. 1964, NASA.

[90] Harrje, D. T. and Condomines, A., "Combustion Instability in Liquid Propellant Rocket Motors—Twenty-ninth Quarterly Progress Report," Aeronautical Engineering Rept. 216-66, Sept. 1959, Princeton Univ., Princeton, N.J.

[91] Heidmann, M. F., "Propellant Vaporization as a Criterion for Rocket Engine Design; Experimental Effect of Fuel Temperature on Liquid Oxygen-Heptane Performance," RM E57E03, July 1957, NASA.

MAY 1972 J. SPACECRAFT VOL. 9, NO. 5

High-Thrust Density Colloid Source Development

C. W. Lear* and M. N. Huberman†
TRW Systems Group, Redondo Beach, Calif.
and
J. W. Geis‡
Air Force Aero Propulsion Laboratory, Wright Patterson Air Force Base, Ohio

An improved high-thrust colloid emitting source is being developed as the basic microthruster for use in 0.1-1 mlb (0.5-5 μN) electric thruster systems in the 1000–2000 sec specific impulse range. The source, a thick needle type geometry, is shown to be a result of work with earlier linear and annular slit geometries. Short and long term performance, including a 2000-hr continuous life test, is discussed. Electrostatic thrust vectoring performance characteristics and reliability implications are also discussed.

Introduction

Although colloid microthruster systems are already competitive as secondary propulsion systems for certain missions on Earth orbiting spacecraft,[1] further increases in thrust density can lead to additional size and weight savings. A research program is currently underway with the objective of attaining improvements over the performance of conventional needle type colloid thrusters. Specific advantages sought are 1) high-thrust density leading to a minimum thruster size and weight, 2) fewer number of individual components per thruster, and 3) the development of a two-dimensional electrostatic thrust vectoring capability.

Thruster Development

The search for a high-thrust density colloid source began at TRW with three independent lines of investigation: a) the thick needle concept was originated in 1965 and was investigated until late 1967 when 0.028″ OD needles were tested,[2] b) the linear slit geometry research dates back to mid-1964 and was the subject of extensive investigation over the 1966–68 period,[3] and c) annular slit geometry research was begun in early 1965 and work has continued to the present time by several researchers.[2,5]

The linear slit geometry requires higher voltages than conventional needle geometries to produce a given extraction field. Thus, a higher I_{sp} is attained for a given charge to mass ratio. In addition, the linear source is a geometrically continuous emitter, resulting in high-thrust density. Scalability of this concept was demonstrated by developing a module[4] consisting of two parallel slits, each having an emitting length of 1.5 cm. The module produced 220 μN thrust at 1500 sec specific impulse. Excessive fabrication difficulties, wide beam spread and limited thrust density have discouraged further development of this concept.

Small diameter double rimmed annular slits have also been investigated. These have been shown to have better focussing properties than linear slits, but field enhancement at the smaller diameter inner extractor has caused voltage breakdown problems. Increasing the source diameter can eliminate this problem. However, this results in a return to the basic performance features of the linear slit geometry since its radial field divergence is now less significant in the vicinity of the emitting rim.

A cure for the central extractor voltage breakdown problem would be to eliminate the center extractor. One such configuration[4,6] consists of a small diameter annular slit in which the inner emitting rim and extractor are replaced by a central plug, held at source potential. Recently, devices of this type have achieved 110–180 μN of thrust at 1500 sec I_{sp} with high-beam distribution efficiency and low-beam spread. In addition to single unit tests, multisource module experiments have also been performed. A seven unit module has been life tested for 500 hrs. Tests such as these indicated that the life limiting factor for these devices was performance degradation due to sediment buildup on the center plug. This problem was then solved by eliminating the center plug, thus letting the center fill with sufficient fluid to maintain a liquid solution. In the following, we discuss the performance of this latter device.

Thruster Design

Figure 1 shows the major design details of the thruster. The main features of the emitter are a sharp radius of curvature ($\simeq$0.025 mm) and an experimentally designed "meniscus

Presented as Paper 71-694 at the AIAA/SAE 7th Propulsion Joint Specialist Conference, Salt Lake City, Utah, June 14–18, 1971: submitted June 22, 1971; revision received January 3, 1972. This work was sponsored by AFAPL under Contract F33615-70-C-1405.

Index category: Electric and Advanced Space Propulsion.

* Member Professional Staff. Member AIAA.
† Head, Colloid Thruster Technology Section.
‡ Aerospace Engineer.