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Engineering Approach to Combustion Instability

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Prepared by O.W. DYKEMA

Propulsion Department, Applied Mechanics Division

Prepared for COMMANDER SPACE SYSTEMS DIVISION

AIR FORCE SYSTEMS COMMAND

LOS ANGELES AIR FORCE STATION

Los Angeles, California

ENGINEERING TECHNICAL OPERATIONS • AEROSPACE CORPORATION

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AN ENGINEERING APPROACH TO COMBUSTION INSTABILITY

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Approved

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This technical documentary report has been reviewed and is approved.

For Space Systems Division
Air Force Systems Command

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Director, Gemini Launch Vehicle
System Program Office
Deputy for Launch Vehicles

FOREWORD

This report was first presented as a professional paper at the 2nd ICRPG Combustion Conference, held at Aerospace Coporation, El Segundo, California, November 1-5, 1965. It is now issued as a Technical Documentary Report for release to the Air Force, and for distribution to industrial and scientific organizations active in the propulsion field.

ABSTRACT

A simplified engineering approach to the analysis of high frequency combustion instability in large liquid rocket engines is outlined. The approach stems from theoretical consideration of pressure and time dependent droplet combustion. There results a dimensionless correlating parameter, called a stability number (N_s), which essentially represents the dimensionless ratio of a characteristic molecular diffusion time to a characteristic acoustic time. Stable and unstable ranges of N_s are defined, and N_s is reduced and simplified to common, readily measurable engineering terms involving the injector orifice pattern (size and number of orifices), the frequency of the acoustic modes, chamber pressure, and propellant flow rate.

The main purpose of the report is to demonstrate the application of this approach to real engine instability problems. An example application is shown, using data from the Aerojet Gemini Stability Improvement Program. Other applications conducted over a period of five years, to a wide variety of large liquid rocket engines involving seven injector element types, four propellant combinations, and ten combustors ranging in thrust from 8,000 to 1,500,000 pounds, and in chamber pressure from 100 to 1,100 psia, are briefly reviewed.

The approach can be used with good accuracy to interpolate or extrapolate N_s to stable regions if some instability data is available and can be used (with some limitations) to generate design data for stability of a new engine.

CONTENTS

INTRODUCTION

In the process of development of large liquid rocket engines, the problem of combustion instability often becomes of very real and immediate concern. The development engineers, and their supporting analytical and research personnel, are faced with the necessity of obtaining some sort of solution of this problem for their particular case. Lacking complete understanding of the problem at that moment, the time honored process of cut-and-try has often been successful in obtaining a particular solution for stability within narrow limits of the specific hardware and operating conditions. This process can be appreciably accelerated, with large savings in cost and time, if even some simple correlating or scaling parameters can be utilized.

This paper is concerned with the description of such a parameter and the results of applications to large liquid rocket engines. The theoretical work involved in development of the parameter was performed by the writer over five years ago, but has never been published in the open literature. Recent theoretical work, notably by Strahle (Reference 1 - 3) and to some extent by Beltran, et al (Reference 4), covers much the same area, but with greater rigor. For the present paper, the theoretical considerations in the development of the parameter will be described only briefly.

The main strength of this approach lies in the ease with which it may be applied to real instability problems in large liquid rocket engines. The theoretical considerations involved have been reduced to common, easily measured engineering parameters. Successful application to a number of real instability problems in large liquid rocket engines has demonstrated that accurate solutions to combustion instability in existing engines and rough design criteria for new systems can be obtained quickly and easily through the use of this approach.

THEORETICAL BACKGROUND

High-frequency combustion instability has been observed in large engines at frequencies as low as 500 cps. This represents an oscillation period as long as 2 milliseconds. Instabilities of this type appear to be wholly thrust chamber oriented and appear to exhibit a time delay behavior (regions of stability and instability). Some process or series of processes in the spray combustion, therefore, must be sufficiently slow to appreciably affect the dynamics of spray combustion at frequencies as low as 500 cps.

The approach described herein assumes that only the process of mass transfer from the liquid surface to the flame by molecular diffusion due to concentration gradient can be sufficiently slow to appreciably affect the spray combustion dynamics. Based on this assumption, then, a model is developed which is sufficiently simplified that at least a correlation parameter can be obtained.

Single Droplet Model. Major assumptions involved in the single droplet model are:

1. A spherical droplet surrounded by a concentric spherical flame.

2. Droplet and flame radii and droplet temperature are constant over the period of the oscillation.

3. The ratio of flame to droplet radii is independent of droplet size.

4. Liquid-vapor phase equilibrium is maintained at the droplet surface.

Thus, from the basic assumption that molecular diffusion by concentration gradient is the important mass transfer process, the unsteady state equation for concentration of fuel vapor as a function of radial position and time is

$$\frac{\partial Y_F}{\partial t} = D_F \left[\frac{\partial^2 Y_F}{\partial r^2} + \frac{2}{r} \frac{\partial Y_F}{\partial r} \right] \tag{1}$$

If the chamber pressure in the region of the burning droplet is oscillating sinusoidally, and phase equilibrium is maintained at the droplet surface, then the vapor concentration (mass fraction) at the surface will also oscillate and will be 180 degrees out of phase with the pressure. Eq. (1) can then be solved for the unsteady component of mass transfer into the surrounding flame, by molecular diffusion only, due to the oscillating concentration of the propellant vapor at the liquid surface. Since all processes are assumed to be very fast compared to those of molecular diffusion, the heat release rate in the flame also oscillates and is exactly in phase with the mass flow into the flame.

A solution can be obtained, then, for the unsteady component of the heat release rate in a unit volume of chamber by summing the contributions of all droplets in the unit volume. This is not as clear cut as it appears, however, since it requires knowledge of the spray drop size distribution at the particular location in the chamber. The following assumptions are made to obtain a solution.

1. Mass transfer by molecular diffusion is very rapid if the boundary layer around the droplet is scrubbed thin by high convective gas velocities. The chamber axial location where this layer is most thick is at the region where the relative gas velocity reverses and passes through zero. This region has been found to be close to the injector face, and does appear to coincide with a region of greatest sensitivity to combustion instability. Therefore, this axial location, the first few inches of chamber length, is assumed to be the important region.

2. As a result of the first assumption, only the initial spray size distribution is of interest. The decrease in spray mean size and/or the change in distribution as the burning spray proceeds down the chamber can be neglected.

3. It is further assumed that the initial spray distribution in the region of interest can be approximated by the empirical relation established by Ingebo (Reference 5) where the relative velocity between liquid and gas is set equal to zero.

4. Considering the accuracy with which a spray mean size and distribution can be predicted in a hot firing large liquid rocket engine with standard production injectors, it is considered that only the variation of the mean drop size with the orifice diameter and injection velocity (as described in Reference 5) can be assumed sufficiently general to be useful. Therefore, the spray is considered to be of a single mean size, and the magnitude and distribution are ignored.

5. Finally, the assumption is made, for purposes of simplicity, that the axial velocity of the spray near the injector face is constant and equal to the orifice exit velocity.

As a result of the above considerations, the number of droplets of uniform but unknown size in a unit volume of chamber in the axial region close to the injector face can be given by:

$$N = \frac{3}{4\pi} \frac{A_o}{A_{CH}} \frac{1}{r_d^3} \tag{2}$$

A solution can then be obtained for the unsteady component of the heat release rate resulting from all the droplets in the above unit volume:

$$\frac{\dot{Q}_u}{P_{cu}} = \frac{3A_o}{A_{CH}} \frac{\overline{Y}_{FS}H}{RT_c} K(K-1)\omega\left[\frac{\psi}{N_s}\right] \tag{3}$$

where ψ (defined in nomenclature) is a dimensionless term which is a function only of the dimensionless parameter N_s.

According to the Rayleigh criterion, if an excess of heat is added during the period when the chamber pressure is greater than the mean pressure, the oscillation will be unstable (neglecting damping in the wave propagation). Therefore, the total unsteady component of heat added during the period when the pressure is greater than the mean is:

$$\frac{Q_u}{P_c'} = 6 \frac{A_o}{A_{CH}} \frac{\overline{Y}_{FS}H}{RT_c} K(K-1)\left[\frac{\psi}{N_s}\right] \tag{4}$$

The conclusion is that, neglecting damping, the mode will be unstable when Q_u/P_c' is positive.

Several facets of Eq. (4) should be noted. With the exception of the term ψ/N_s, all terms are positive and concern only the magnitude of the response of the heat release, Q_u, to the chamber pressure amplitude, P_c'. Many of these terms (in particular $\overline{Y}_{FS}$, H, R, T_c, and K) are extremely difficult to estimate in the thrust chamber environment just a few inches from the injector face. For these reasons, no attempt is made toward further quantitative analysis beyond Eq. (4). Further, more rigorous work along these lines can be found in References (1 - 4).

The dimensionless term ψ/N_s, however, can be positive or negative, and is a function only of the dimensionless parameter, N_s. Thus, this term establishes the stability boundaries and the relative magnitude of Q_u/P_c'. The first four zeros of ψ/N_s occur at N_s = 6.4, 30.25, 72 and 100. While there is a region of positive gain in the range $72 < N_s < 100$, the gain is so small that most systems should have sufficient damping to remain stable. For engineering purposes, ψ/N_s can be considered zero for $N_s > 30.25$. Significant instability, then, should occur only in the range of $6.4 < N_s < 30.25$. Figure 1 shows a plot of ψ/N_s, expressed as the percent of maximum combustion gain, over this range of N_s.

It should be recalled that a mode should be unstable for any positive value of ψ/N_s only if there is no damping in the mode. It is clearly possible that sufficient damping can be introduced (by baffles, liners, etc.) such that the mode can be dynamically stable even at the maximum of ψ/N_s. Thus, in the interpretation of Figure 1 (since it is assumed that nothing is known of the damping), if a mode is unstable, then the value of N_s, for that mode, should lie between 6.4 and 30.25, but if a mode is stable, the value of N_s is not necessarily outside of this region

The dimensionless parameter, N_s, also deserves some discussion. This term evolved from the solution of the single droplet diffusion equation and is defined as:

$$N_s = 2\pi f \frac{\ell^2}{2D_F} \qquad (5)$$

It can be shown that $\ell^2/2D_F$ represents the time, τ_D, for diffusion over the distance, ℓ, in the presence of a linear concentration gradient. The frequency, f, is obviously the inverse of the acoustic mode period, τ_a. Therefore, N_s represents the dimensionless ratio of a characteristic molecular diffusion time to a characteristic acoustic time. This correlating or scaling parameter is perhaps the most useful result of the theoretical considerations.

The practical usefulness of this dimensionless ratio is limited, however, unless it can be expressed in readily measurable engineering terms. Using the following relations:

$$\ell^2 = \frac{1}{4} D_{30}^2 (K - 1)^2 \qquad (6)$$

$$D_F = D_{Fo} \frac{P_o}{P_c} \qquad (7)$$

$$D_{30} = K_a \sqrt{\frac{d}{v}} \qquad (8)$$

$$\dot{w} = \rho A_o v \qquad (9)$$

the term, N_s, can be written:

$$N_s = A\left[nd^3\right]\left[\frac{fP_c}{\dot{w}}\right] \tag{10}$$

where:

$$\left[A = \left(\frac{\pi}{4}\right)^2 \frac{\rho}{144}\right]\left[K_a^2\right]\left[\frac{(K-1)^2}{D_{Fo}P_c}\right] \tag{11}$$

In Eq. (11), the first bracket is simply a dimensional constant times the density of the liquid. The second bracket is the atomization constant defined by Eq. (8). The third bracket essentially represents the kinetics of the droplet combustion. Since the second and third terms are very difficult to evaluate in an operating thrust chamber, the entire constant, A, is assumed to be unknown, and must be determined by correlation of instability data.

In Eq. (10), the first bracket includes the only terms which define the injector geometry. The second term includes the operating conditions of chamber mode frequencies, chamber pressure, and propellant flow rate.

In all of the above analysis, droplets of a single propellant are assumed to be burning in vapors of the other propellant. Thus, the controlling propellant must be determined or assumed and the appropriate terms used in Eq. (2) through (11). For example, in the LOX/RP-1 system, RP-1 is obviously the slower burning propellant and controls the stability of the thrust chamber. In storable propellant systems, the controlling propellant is not so obvious, and must be determined by trial correlations of stability data.

Thus, Eq. (10) consists of readily measurable engineering parameters by which instability data can be correlated, with an unknown constant, A. By comparison of the correlated stability data with the stability regions predicted by Eq. (4) or Figure 1, the constant may be roughly determined. Design data can then be extrapolated or interpolated for any other system involving the same injector element type and propellant system to obtain stability.

Example Application. One problem in applying this technique to instability in a real engine involves the type of instability data which is available. Most early engine development programs did not pulse or bomb the thrust chambers to evaluate instability. The only data available were obtained from spontaneous instabilities. If a given injector sustained one or two spontaneous instabilities, the design was usually abandoned. Often the necessary data for correlation was lost when the thrust chamber and instrumentation were destroyed. As a result, very little instability data was available, and often the most unstable regions were characterized by the complete absence of data, stable or unstable.

More recently, thrust chamber stability has been evaluated by pulse techniques. Here the stability of a given system can be more accurately evaluated, over wider ranges of operating conditions, before it is abandoned. Also, because a given system can be fairly accurately and completely evaluated in just a few pulse tests, a wider variety of systems can be evaluated in a single program.

As an example of the application of the correlation technique described herein, data from the Gemini Stability Improvement Program (GEMSIP) which utilized both tangential pulse guns and non-directional bombs to evaluate stability, will be used (Reference 6). This program was conducted by Aerojet-General Corporation, Sacramento, California.

The Aerojet Combustion Dynamics groups normally use the engineering parameter thrust-per-element (T/E) to characterize the injector configuration. In order to be consistent with their data, (T/E) can be substituted for the injector term (nd^3) in Eq. (10) according to:

$$nd^3 = K_1 \sqrt{T/E} \tag{12}$$

where

$$K_1 = \left[\frac{4\dot{w}}{C_d \pi \sqrt{2g\rho\Delta P}}\right]^{3/2} \left[\frac{1}{2F_{nsl}}\right]^{1/2} \tag{13}$$

and all the terms in Eq. (13) are the nominal values which result in the nominal sea level thrust, F_{nsl}. Eq. (10) then can be written:

$$N_s = B \sqrt{T/E} \left[\frac{fP_c}{\dot{w}}\right] \tag{14}$$

Early screening of a wide variety of injectors with 100 lb thrust per element showed that the first radial mode could be triggered unstable, and that a 1200 cps mode was often spontaneously unstable. Preliminary analysis, considering the baffle pockets as side branches on the transverse modes, indicated that the 1200 cps mode could be the first tangential mode (approximately 1900 cps when unbaffled) reduced in frequency by the 5-inch long baffles. (This was later confirmed by phase and amplitude analysis.)

Injectors with larger orifices (thrust per element greater than 100 lb) could not be triggered in the first radial mode. The majority of the baffle configurations tested were radial baffles only. Thus, no baffle damping was introduced in the first radial mode. The observed stability in the first radial mode (uniform injection density) with larger thrust per element injectors, then, must be due to dynamic stability of the mode itself. Since the value of N_s increases with thrust per element and stability increased with thrust per element, data on the first radial mode must lie in the region of N_s around the upper stability boundary at $N_s = 30.25$ (see Figure 1). Similarly, since the first tangential mode was spontaneously unstable with the 100 lb per element injector, in spite of the large damping introduced by the 5-inch long baffles, N_s values for this mode must lie near the maximum gain at $N_s = 9.0$.

Table I shows calculated values of N_s/B for the first radial mode. Although two data points disagree, it is clear that operation at N_s/B values less than 1.55 is significantly more unstable than for values greater than 1.55. Considering this boundary to correspond to N_s of 30.25 yields a value of 1.95×10^{-4} for the unknown constant, B.

Applying this constant to data on the first tangential mode (Table II) yields values of N_s which are in the region of maximum gain, as expected. Again, the stability data is not exact, but Table II shows that all five tests with gain less than 96 percent of maximum were stable, while 9 of the 12 tests with gain greater than 96 percent were unstable. Table II also shows that the pulse type used in the tests varied from the largest (220 grain non-directional bomb), which could not trigger the mode when the gain was low, to no pulse at all (spontaneous instability) required to trigger the mode at maximum values of gain.

Using the above correlations to conclude that the unknown constant B for this configuration is 1.95×10^{-4}, the thrust chamber could be made dynamically stable, without the need for baffles, by increasing the thrust per element to greater than 450 lb (considering that the first tangential mode frequency increases to approximately 1900 cps when the baffles are removed). This is considered too large a modification, from the standpoint of performance and injector face cooling. The final configuration uses 200 lb per element. Thus, the first and second tangential modes still have approximately 41 and 3 percent of maximum gain, respectively, and therefore must depend on some baffle damping to remain dynamically stable. A single test with the baffles removed confirmed that the first tangential mode with a 200 lb thrust per element injector was still unstable. All modes higher than the second tangential are dynamically stable, and need no baffle damping.

Results of Other Applications. The general technique described herein has been applied to a wide variety of large liquid rocket engine systems over a period of five years. These systems included seven injector element types, four propellant combinations, and ten combustors and ranged from a hydrazine monopropellant model motor with full cone swirl nozzle injectors to the LOX/RP-1, F-1 thrust chamber with a doublet injector. A short chronological summary of the major results of these applications is listed below:

1. Hydrazine Monopropellant - A model motor program in which droplet mean sizes were estimated by the light transmission technique. Good correlation obtained in longitudinal mode. Operation could be made stable or unstable at will (Reference 7).

2. Rocketdyne Jupiter Thrust Chamber - Inconclusive as a result of insufficient instrumentation to clearly determine whether the instability was combustion instability or feed system coupled instability.

3. Rocketdyne F-1 Thrust Chamber - Obtained reasonably good correlation which predicted minimum fuel orifice diameters to stabilize the baffled and unbaffled first tangential modes. Subsequent pulse testing showed excellent agreement (Reference 8).

4. Rocketdyne H-1 Thrust Chamber - Results inconclusive because of the lack of spontaneously unstable tests with the unbaffled injector and a predominant low frequency feed system coupled modo with the baffled injector.

5. F-1 Gas Generator - Good correlation obtained in longitudinal and first tangential modes. Longitudinal modes minimized by reducing reflection efficiency in combustor body, tangential modes reduced by increasing fuel orifice diameters (Reference 8).

6. Rocketdyne LEM Thrust Chamber - Good correlation obtained initially but effort terminated before conclusive results could be obtained.

7. Rocketdyne Experimental Engine (B_5H_9/N_2H_4) - Reasonable correlation obtained with three widely different injector types. Attempt to develop a stable design by placing the frequency of maximum combustion gain between two chamber frequencies was partially successful (Reference 9).

8. Titan II Stage II Thrust Chamber (Gemini Stability Improvement Program - Successful correlation indicated in Example Application Section of this report (Reference 6).

Three other injector combustor systems are presently being analyzed. One is the Titan III Stage I thrust chamber. The stability of this system appears to be very much the same situation as in the Stage II engine (GEMSIP), and the same solution should apply for a dynamically stable injector.

The other two systems are the Aerojet (AGC) and the United Technology (UTC) Transtage systems. Since the operating conditions of these two completely independent systems are identical, the frequencies of their predominant modes of instability should be inversely proportional to the values of (nd_o^3) for their injectors. Based on preliminary frequency data (References 10 and 11):

	nd_o^3	Predominant Frequency, cps.
AGC	0.0419	3900
UTC	0.0192	8800*

Clearly, the frequencies are inversely proportional to the values of (nd_o^3) within the accuracy to which the frequencies could be determined.

*Based on the only two tests where instrumentation was known to be reliable.

A further interesting point in the UTC data concerns the lower transverse modes. The predominant mode of instability appears to be either the fourth or fifth tangential mode near 8800 cps. If this corresponds to the point of maximum combustion gain at $N_s = 9.0$, then N_s values for the first, second, and third tangential and the first radial modes will be less than 6.4 and should be stable. Extensive frequency analysis of one of the tests where the instrumentation was known to be reliable clearly showed that all transverse modes up through the fifth tangential were excited by the tangential pulse gun, but within 15 - 30 milliseconds all modes lower than the fourth tangential damped out and the instability sustained in the fourth and fifth tangentials.

Table III lists the various combustor assemblies for which correlations have been obtained and values of the unknown constant, A, determined. Although the reason is not apparent, it appears that the constants are remarkably similar despite the rather wide variation of injector element types and propellant systems shown in the table. This appears to indicate that stability is nearly independent of these parameters. If this is the case, then rough initial design criteria can be established for stability of an entirely new engine system. An average value of A (5.7×10^{-3}) can be assumed initially, recognizing that this value is based on only seven samples with a range from 4.0×10^{-3} to 6.9×10^{-3}. While this range of uncertainty in N_s seems very large, it must be recalled that the unstable region shown in Figure 1 covers a range of N_s from 6.4 to 30, or a factor of nearly 5. Of course, stability data from early tests of the new system can be used to quickly refine the initial estimate of A.

An attempt to use the above procedure to analyze the stability of the United Technology Transtage engine before any stability data was obtained resulted in the prediction that all modes below 9250 cps (lower N_s boundary) should be stable. Subsequent pulse tests showed all modes below 7300 cps were stable. Thus, the initial assumption was in error by 27 percent. This error seems reasonable, particularly if it is noted that the Transtage engine represents an extrapolation by a factor of about 10 in chamber pressure and about 25 in flow rate from the engines used to determine the average value of A. Thus, the fairly close agreement in the case of the UTC Transtage engine further supports the observation that stability is relatively independent of injector element types or propellant systems.

This result agrees with the experimental observations of Reardon (Reference 12) that the sensitive time lag and interaction index for a wide variety of injector types and propellant combinations can be correlated by single curves. According to theory, the sensitive time lag is equal to half the acoustic mode period. Therefore, the frequency term in Eq. (10) can be converted to a sensitive time lag and a direct comparison made between Eq. (10) and the empirical plot of Reference 12.

Using the value of the constant, B, determined from the GEMSIP data, curves of sensitive time lag versus thrust per element can be developed for given values of the stability number, N_s. Figure 2 shows that the empirical curve of Reference 12 corresponds closely to a curve developed from Eq.(14) for $N_s = 30$, the so-called upper (in N_s) stability boundary. Clearly, the empirical curve of Reference 12 shows that the sensitive time lag is approximately proportional to the square root of the thrust per element, as predicted by Eq. (10).

The sensitive time lag theory assumes that the actual combustion interaction index is symmetrically distributed about a given value of the sensitive time lag. Because of the relationship between the assumed combustion interaction index and that necessary for instability, the sensitive time delay is assumed to correspond most nearly to the highest observed unstable frequency (or the smallest time delay). Thus, it is reasonable that the empirical curve of Reference 12 should correspond to the high frequency boundary of Eq. (10) and Figure 1.

CONCLUSIONS

A simple correlating parameter has been developed which can be extremely useful in the engineering solution of high-frequency combustion instability problems in large liquid rocket engines. The parameter has been reduced to common, easily measured terms for direct application to engine development programs. This approach has been successfully applied to a number of full-scale engine systems ranging in thrust from 8,000 to 1,500,000 pounds, and in chamber pressure from 100 to 1100 psia.

The approach may be used with good accuracy to interpolate or extrapolate to stable conditions if instability data is available, or can probably be used for initial design data, with somewhat limited accuracy.

The theoretical considerations from which the approach was developed may serve to indicate fruitful avenues for research leading to complete, rigorous solution of the high frequency combustion instability problem.

FIGURE 1

THEORETICAL VARIATION OF COMBUSTION GAIN WITH THE STABILITY PARAMETER N_S

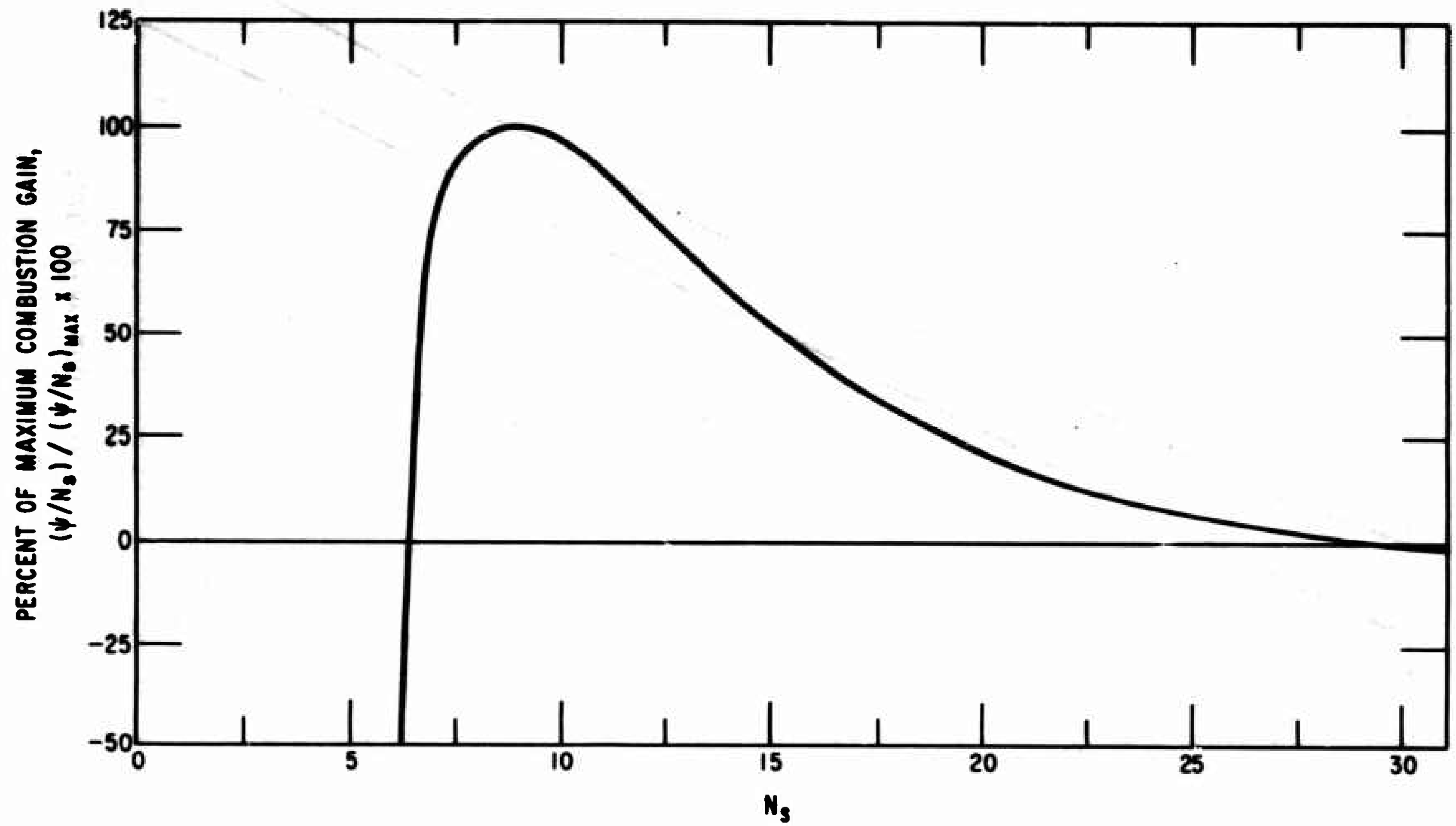

FIGURE 2

COMPARISON OF PREDICTED VARIATION OF COMBUSTION TIME DELAY WITH EMPIRICAL CORRELATION OF REARDON

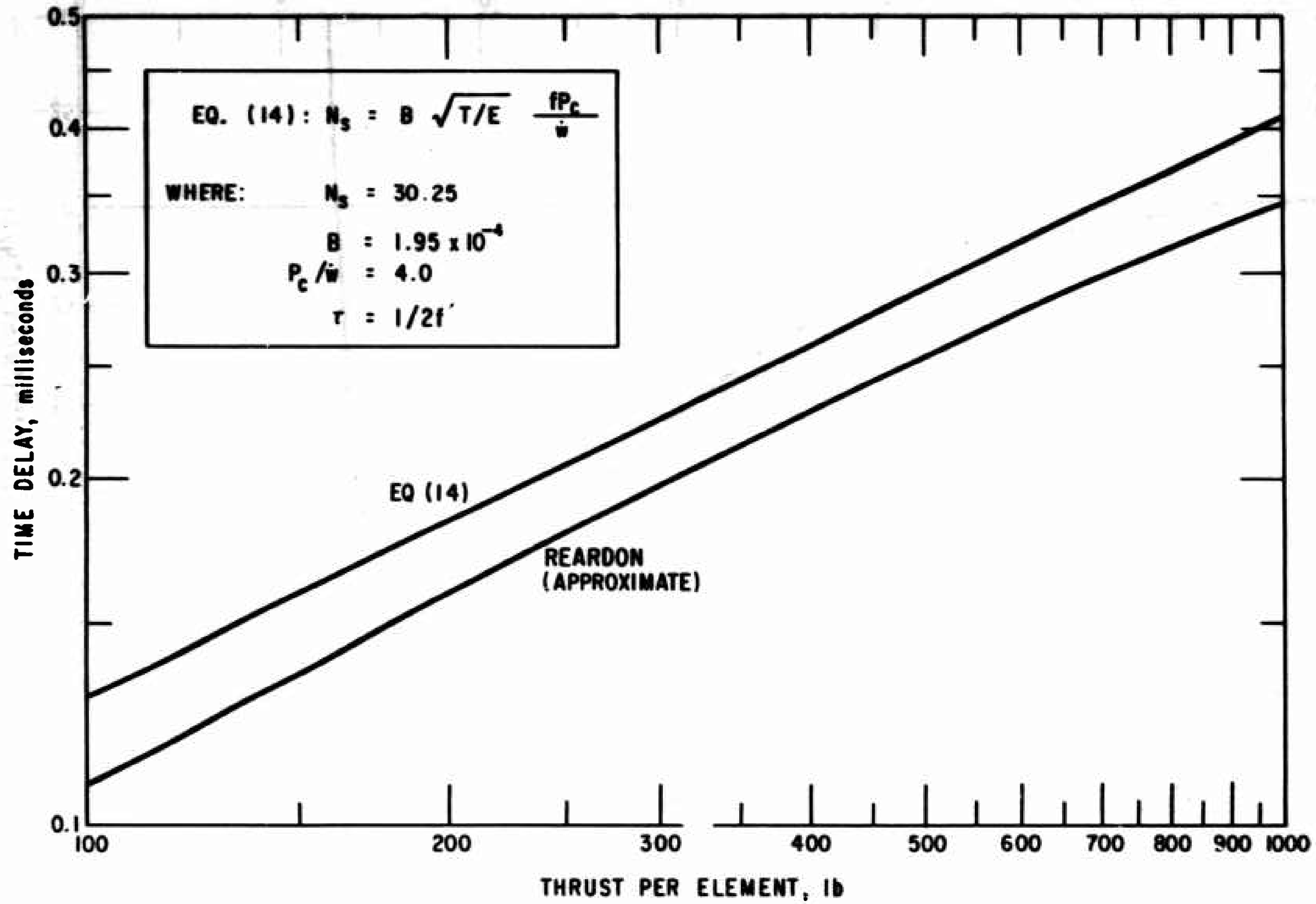

TABLE I

CORRELATION OF GEMSIP FIRST RADIAL MODE STABILITY

(Radial baffles, flat and ramp distributions)

Test Number	Thrust Per Element, lb.	$N_s/B \times 10^{-5}$	Stable (S) or Unstable (U)
117	100	1.38	S
017	100	1.38	U
103	100	1.40	U
125	100	1.41	U
115	100	1.55	U
012	100	1.61	S
124	150	1.65	S
005	100	1.69	U
114	150	1.87	S
138	150	1.89	S
008	200	2.02	S
118	200	2.03	S
116	200	2.06	S
127	200	2.06	S
123	200	2.08	S
119	200	2.15	S
137	200	2.18	S
113	200	2.21	S
126	200	2.22	S
143	200	2.28	S
101	200	2.42	S

TABLE II

CORRELATION OF GEMSIP FIRST TANGENTIAL MODE STABILITY

(5-inch long baffles, flat injection distribution)

Test Number	N_s	Gain, Percent of Maximum	Stable (S) or Unstable (U)	Pulse* Type
132	7.50	88.0	S	220 ND
019	7.91	95.7	S	220 ND
017	8.07	96.8	U	220 ND
117	8.09	97.1	S	100 ND
103	8.20	97.9	U	100 ND
011	8.20	97.9	U	80 T
104	8.47	99.2	U	80 T
136	8.57	99.2	S	80 T
006	8.88	100.0	U	Spont.
140	8.86	100.0	U	Spont.
016	9.18	100.0	U	100 ND
002	9.66	98.5	U	100 ND
124	9.66	98.5	S	220 ND
005	9.87	97.1	U	100 ND
138	11.05	88.1	S	220 ND
008	11.85	81.1	S	220 ND
101	14.15	60.9	S	100 ND

* Number indicates charge size in grains.
ND indicates non-directional bomb.
T indicates tangential pulse gun.
Spont. indicates self-triggered instability.

TABLE III

SUMMARY OF STABILITY CONSTANTS OBTAINED IN VARIOUS CORRELATIONS

Combustor	Propellant Combination	Controlling Propellant	Injector Element	Stability Constant ($A \times 10^{+3}$)
F-1 Thrust Chamber	LOX/RP-1	RP-1	Doublet	5.02
Rocketdyne LEM	N_2O_4/A-50	N_2O_4	Doublet	6.40
F-1 Gas Generator	LOX/RP-1	RP-1	Triplet	5.61
Rocketdyne Experimental Engine	B_5H_9/N_2H_4	N_2H_4	Triplet	5.14
Titan II Stage II	N_2O_4/A-50	N_2O_4	Quadlet	6.88
Rocketdyne Experimental Engine	B_5H_9/N_2H_4	N_2H_4	Two-on-Two	4.03
Rocketdyne Experimental Engine	B_5H_9/N_2H_4	N_2H_4	Recessed Quintuplet	6.92
Rocketdyne Research Model Motor	N_2H_4 Monopropellant	N_2H_4	Full Cone Swirl Nozzle	Not Applicable

Avg. $A = 5.7 \times 10^{-3}$

NOMENCLATURE

A	=	stability constant defined by Equation (11)
A_{CH}	=	cross sectional area of the combustor at the injector face, $in.^2$
A_o	=	total injector orifice area for a given propellant, $in.^2$
B	=	AK_1 = stability constant defined by Equation (14)
C_d	=	injector orifice discharge coefficient
D_F	=	binary diffusion coefficient, $\frac{in.^2}{sec.}$
D_{Fo}	=	binary diffusion coefficient at standard atmospheric pressure, $\frac{in.^2}{sec.}$
D_{30}	=	volume-number mean spray droplet size, inches
d	=	injector orifice diameter, in.
F_{nsl}	=	nominal sea level thrust, lb.
f	=	frequency of the acoustic modes, cps
g	=	32.2 ft/sec
H	=	heating value of propellant, Btu/lb
K	=	ratio of the radius of the flame zone surrounding a droplet to the radius of the droplet, dimensionless
K_a	=	proportionality constant for drop size (Reference 1) defined by Equation (8)
K_1	=	constant defined by Equation (13)
ℓ	=	radial distance from the droplet to the surrounding flame zone, in.
N	=	number of droplets of mean size, D_{30}, per unit volume of combustor near the injector face
N_s	=	stability number or correlating parameter, defined by Equation (5), Equation (10), or Equation (14), dimensionless
n	=	number of injector orifices for a particular propellant
P_c	=	injector end combustion chamber pressure, psia

NOMENCLATURE

P_{cu}	=	unsteady component of chamber pressure, psia
P'_c	=	amplitude of unsteady component of chamber pressure, psia
P_o	=	atmospheric pressure, 14.7 psia
ΔP	=	pressure drop across the injector orifices, psi
Q_u	=	unsteady component of heat added per unit volume of chamber during the period when the pressure is greater than the mean, Btu
$\dot{Q}_u$	=	unsteady component of the heat release rate per unit volume of the chamber near the injector face, Btu/sec.
R	=	gas constant
r	=	radius, in.
r_d	=	droplet radius, in.
T_c	=	gas temperature, ^{o}R
T/E	=	thrust per element, lb.
v	=	injection velocity at the injector orifice exit, in./sec.
$\dot{w}$	=	total flow rate of the injected propellant, lb./sec.
Y_F	=	mass fraction of the specie F
Y_{FS}	=	mean mass fraction of the specie F at the droplet surface
τ_a	=	acoustic mode period, sec.
τ_D	=	diffusion time delay, sec.
ϕ	=	dimensionless parameter related to the in-phase combustion gain and defined by

$$\phi = -\sqrt{N_s}\,\frac{\cos\varphi}{(\cosh 2\sqrt{N_s} - \cos 2\sqrt{N_s})^{1/2}}$$

$$\text{where } \varphi = \frac{\pi}{4} - \text{arc tan}\left[\coth\sqrt{N_s}\ \tan\sqrt{N_s}\right]$$

ω	=	acoustic mode frequency, rad./sec.

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