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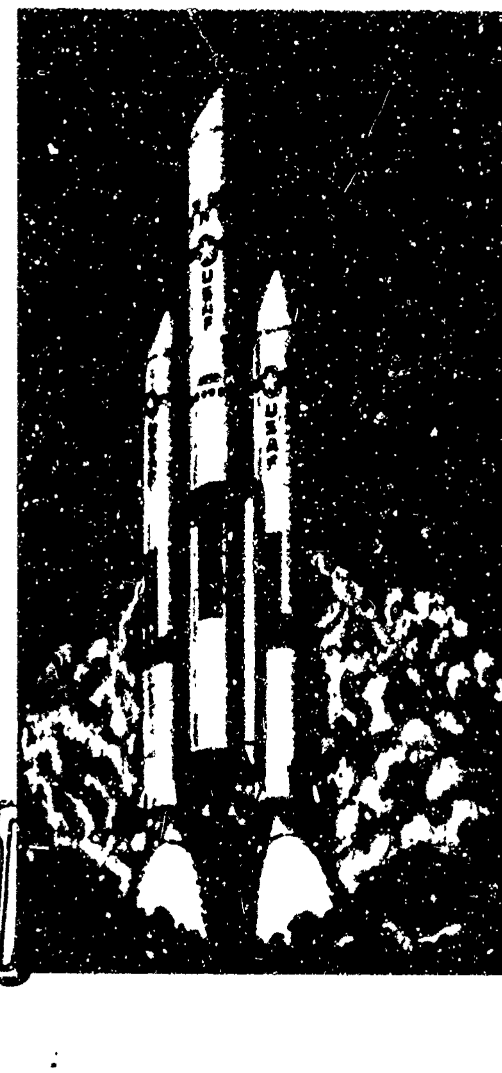
June 1-4, 1965

ABSTRACTS

AFOSR Combined Contractors Meeting on Combustion Dynamics Research



Thiokol CHEMICAL CORPORATION
REACTION MOTORS DIVISION
DENVILLE NEW JERSEY

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AFOSR 65-0590
AF 49(638)1505

ABSTRACTS

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LONGITUDINAL COMBUSTION INSTABILITY STUDIES USING A GASEOUS PROPELLANT ROCKET MOTOR*

C. T. Bowman and I. Glassman

Princeton University

As reported previously, a theoretical model for the longitudinal mode of high frequency combustion instability in rocket motors burning pre-mixed gaseous propellants had been formulated. In this model it was assumed that the driving mechanism for instability was related to chemical kinetic factors. The kinetics input takes the form of a simplified Arrhenius-type rate function.

The temperature dependence of the stability limits predicted by the theory has now been verified by experiments in which the combustion temperature was artifically reduced by the addition of various amounts of inert diluent (N_2) to several propellant combinations (CH_4 - O_2, H_2 - O_2). Experiments in which the over-all activation energy, but not the combustion temperature, of a propellant combination (CO-O_2) was altered by the addition of a trace amount of a catalyzing species (H_2) have verified the theoretical dependence of the stability limits on activation energy. The pressure dependence of stability limits and regions of harmonic modes of instability have been determined. It was found that as the chamber pressure was decreased, the regions of unstable operation narrowed and that the regions of harmonic modes shifted to longer chamber lengths. A complementary experimental study has been made of the effects of various nozzle and injector configurations on stability limits, instability wave form, and regions of harmonic modes. The limits and wave form were found to be essentially independent of the choice of nozzle or injector. The harmonic regions were found to be strongly dependent on the injector configuration. Agreement between theory and experiment indicates that the appropriate mechanism for gas-phase instability is related to chemical kinetics. The above results now permit a critical comparison of the experimental results with those of other workers who have studied gas-phase instability to be made. As well, the differences and similarities between gas-phase and two-phase high frequency combustion instability can be examined, and conclusions as to the importance of the gas phase to the overall instability phenomena drawn. These aspects of the work are currently in progress.

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* Research sponsored by AFOSR under Contract AF 49(638)-1268

BASIC MECHANISMS OF COMBUSTION INSTABILITY*

T. Y. Toong

Massachusetts Institute of Technology
Cambridge, Mass.

The main objective of this research program is to study the detailed mechanisms involved in the triggering, amplification and suppression of acoustic oscillations due to the presence of flames.

Results obtained (some of which have been reported at the Tenth International Symposium on Combustion) indicate the existence of both linear and nonlinear couplings between acoustic anf flame oscillations through flow oscillations. A theoretical analysis of the flame-flow-acoustic interactions is being carried out. It is based on the conceptual model discussed in the Symposium paper, and retains the intrinsic nonlinearity of the governing equations. The results of this analysis obtained so far agree well with experimental observations. They are summarized below:

(a) There are two characteristic times in the problem an acoustic time and a flow time.

(b) When the dependent variables are expanded in powers of a reference Mach Number, the zeroth-order terms describe the mean flow, the first-order terms, and Tollmien-Schlichting waves, and the second-order terms, the acoustic waves.

(c) Amplification or damping of the acoustic waves by the flame is caused by both the mean flow (zeroth-order terms) and the Tollmien Schlichting Waves (first-order terms).

(d) The acoustic waves (second-order terms) can influence the Tollmien-

Schlichting waves (first-order terms), and the Tollmien-Schlichting waves, in turn, can influence the mean flow (zeroth-order terms); thus, the acoustic waves can indirectly influence the mean flow.

Experiments have been performed to show the validity of the theoretical results. They are, among others, briefly summarized below:

(a) Since the theory indicates that the mean flow should have a strong effect on the acoustic-flame interactions, the effect of a forced-convective flow on these interactions inside an organ pipe has been investigated. Results indicate that, as the flow rate is increased, the acoustic-flame interactions pass through several maxima and minima.

(b) The influence of the mean flow on the acoustic waves has been further demonstrated by the amplification of the acoustic waves through augmenting (with vanes) the tangential flow near a disturbed flame.

(c) The effect of the externally imposed acoustic waves on the mean flow, as predicted by the theory, has been investigated by motion-picture photography, which confirms the existence of violent tangential motion.

Both theoretical and experimental investigations are in progress, to study in detail the nature of the flame-flow-acoustic interactions.

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*Research sponsored by AFOSR under Contract AF 49(638)-1354

HIGH-FREQUENCY COMBUSTION INSTABILITY AND SCALING OF LIQUID PROPELLANT ROCKET MOTORS*

Vito D. Agosta, Sanford S. Hammer and William T. Paschke

Polytechnic Institute of Brooklyn

The investigation on combustion instability pursued at the Propulsion Research Laboratory of the Polytechnic Institute of Brooklyn is directed toward a nonlinear analysis of the wave growth or attenuation in the longitudinal mode of oscillatory behavior in a liquid propellant rocket motor. This analysis includes a time dependent energy/mass source term in the wave equations which is obtained from the injector characteristics and the fuel and oxidant droplet ballistics, and serves to drive the waves. In order to solve for the propellant droplet ballistics, a sophisiticated non-oscillatory aerothermochemical analysis is necessary to determine the gas and droplet properties in the rocket motor chamber. In addition to being employed in the wave analysis, the non-oscillatory aerothermochemical program can be used to investigate the inverse problem, namely, for a heterogeneous reacting fluid field with large acoustic radiation and convective dissipation, what chamber geometry results? Thus the problem of combustion instability is approached from two points of view, in the first, a non-coupling of the energy/mass wave driving sources is sought, in the second, a rocket chamber geometry is generated such that wave attenuation and dissipation and acoustic radiation is maximized.

Experimental results in 500-pound thrust motors indicate three types of instability: In the first type, a pressure spike moving at sonic velocity occurs with concomitant increase of chamber pressure. In the second type, a pressure spike occurs moving at sonic velocity without any increase in chamber pressure. In the third type, a "standing" wave occurs in the chamber. All three types are included in the analytical analysis.

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*Research sponsored by AFOSR under Contract AF 49(638)-1263

COMBUSTION INSTABILITY IN LIQUID ROCKET MOTORS

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Princeton, New Jersey

Theoretical efforts have primarily included an extension of the nonlinear analysis to the transverse case, a study of the higher order Mach number effects on the stability of the longitudinal mode, a more general approach to the linear stability problem, the development of a computer program which calculates three-dimensional nozzle admittances, and a separate study of the asymptotic behavior of these admittances.

The nonlinear transverse theory has been completed up to and including third order effects. Calculations based on this theory will indicate in what cases the possibility of triggering action exists. The Mach number squared effects have, in general, a destabilizing effect on the longitudinal mode, i.e., the width of the instability zone in a vs n plot will increase. At the last Combustion Institute Meeting, a more general but yet simpler version of the linear stability theory was presented. This allowed for a better understanding of the terms in the equations which affect the stability criterion. Conical nozzle admittance coefficients are being calculated presently and should be presented in the near future. The development of an asymptotic theory has allowed for a better understanding of the unsteady flow in the nozzle and is useful in the interpretation of the computer results.

The influence of droplet size on stability remains an important aspect of both the theoretical and experimental efforts. Under conditions which simulate densities as high as those associated with 2000 psi chamber pressures droplet size and distribution data have been obtained covering a range of orifice sizes. General correlations of these data has been completed and provide valuable reference data for the determination of unstable combustion parameters.

The influence of various baffle configurations on the injected propellants has been studied in the "pseudo" rocket motor. Measurements of pressure and velocity are made in addition to the photographic records of vapor displacement effects associated with the Freon jet. Indications of the protection afforded by the baffles from transverse velocities are apparent in these tests. Velocity measurements under these pulsed conditions are extremely difficult.

Rocket firings have emphasized the effects resulting from interactions of like and unlike spray fans on the instability regimes. Both the transverse and variable-length square-motor tests have documented these effects. Pressure level increases have produced higher modes of instability and in some cases have provided stable regimes at intermediate pressures. Control of axial combustion distribution via droplet size and mixed injector types has been also investigated. Effects due to injection near baffle surfaces have also been observed.

EXPERIMENTAL MEASUREMENTS ON A ROTATING DETONATION-LIKE WAVE OBSERVED DURING LIQUID ROCKET RESONANT COMBUSTION

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A single, high amplitude pressure wave rotating with supersonic velocity about the combustion chamber axis has been observed during the resonant (oscillatory) combustion mode of several liquid rocket research engines. The occurence of this very steep fronted disturbance has led to an investigation of the applicability of a rotating detonation-like wave concept to explain the phenomenon. Results of a portion of the experimental phase of the investigation are presented showing the chamber boundary pressure distribution associated with the resonant combustion exhibited by one of these engines. The engine was operated with HNO_3 + aniline/furfuryl alcohol propellants at a nominal thrust of 20,000 pounds and 300 psia chamber pressure.

The pressure distribution was obtained by several simultaneous high frequency response measurements across a radius of the injector face and along the length of the 11 inch I.D. cylindrical chamber. The pressure wave-to-chamber wall intersection was found to curve in the direction of wave rotation with the nozzle end of the intersection leading the injector end by approximately 30 circumferential degrees. The wave-to-injector face intersection was found to be non-radial and to extend into the central area of the face, though the definition of the intersection was poor in this area.

The pressure ratio across the wave front (ratio of peak to minimum pressures during a wave rotation period) varied along the chamber wall from in excess of 20 near the injector to 4 near the nozzle entrance. The pressure ratio at the face varied from greater than 10 in the outer half radius to less than 3 near the center. The non-symmetrical wave exhibited a rise time of less than 10 u seconds.

A brief description of the apparatus and instrumentation will be included.

COMBUSTION INSTABILITY WITH LIQUID OXYGEN AND LIQUID OR COLD GASEOUS HYDROGEN

D. E. Dahlberg

Pratt & Whitney Aircraft
Florida Research and Development Center

Work accomplished under contract NAS8-11024 (under the cognizance of Marshall Space Flight Center) since the last meeting will be reviewed. Combustion tests with oxygen and hydrogen in a 1 inch by 5 inch two-dimensional chamber with a coaxial injector have been continued to determine the effects of injection momentum ratio, hydrogen temperature and chamber contraction ratio on combustion stability. For this size chamber, hydrogen injection temperature appears to have more influence on combustion stability (less stable at lower temperature) than injection momentum ratio or chamber-to-throat contraction ratio. Decreasing contraction ratio and decreasing injection fuel-oxidizer momentum ratio increased the tendency toward instability. Comparisons of the effect of pressure (500 to 900 psia) on stability are not conclusive.

STUDY OF ROCKET COMBUSTION INSTABILITY*

E. K. Bastress and A. C. Tobey

Arthur D. Little, Inc.

The program objective is to provide both analytical and experimental information concerning the flame-piloting mechanism within a rocket motor, and the effect of external disturbances on its characteristics.

The thesis is that in a stable system the flame-piloting mechanism provides energy for continuous reaction of the mixture; and energy imbalance, as might be caused by small perturbations, creates ignition time-lag changes; this in turn creates changes in over-all reaction rates and flow characteristics within the chamber, thus leading to instability.

The experimental program consists of detailed examination of rocket systems fed with combinations of gaseous and liquid propellants. The validity of various conceptual models in describing the reactive system and its response to small perturbations is being evaluated. Determination is being made of whether or not correlation exists between experimental data and theories evolved during this investigation and the work of other researchers. Measurements are being made of local concentrations, chamber pressure and its axial gradient and the pressure-time variation as functions of oxidant to fuel ratio, chamber length, injector configuration, inert gas injection rate, position of inert gas injector, and propellant phase (liquid or gaseous). High speed motion pictures are being obtained along lines of sight perpendicular and parallel to the motor axis of the reacting combustion gases for various injector and chamber configurations. The effects of fuel-oxidant combinations and motor geometry on the motor stability and the character of the flame-piloting mechanism are being examined.

* Research sponsored by AFOSR under Contract No. AF49(638)-1120, Task No. 37510

LIQUID ROCKET INJECTOR DESIGN AND SCALING

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Liquid rocket Combustion Dynamics work at Aerojet-General covers a broad range, from basic research to development problems, and from instability initiation to stabilization. Particular attention has been given recently to two aspects of the overall situation relating to injector design and scaling: injector pattern effects on stability; and the mechanisms governing baffle stabilization.

The Sensitive Time Lag Theory has been applied to the correlation of experimental combustion instability data so that injector design and scaling criteria could be derived. According to this theory, the stability of a given combustor depends on the geometry of the combustor and on two basic parameters that describe the dynamics of the combustion process. These parameters are the sensitive time lag τ and the interaction index n. Experimental results interpreted in the light of this theory have yielded correlations of n and τ with injector design and operating conditions. The injector design factor in these empirical correlations is the weight flow per orifice of the rate-controlling propellant (usually selected as the fuel, except in LO_2/LH_2 systems). The chamber pressure has been found to be the most significant operating variable, at least for the data assembled to date. The time lag τ has been found to increase with increasing flow-per-orifice, and to decrease with increasing chamber pressure. The interaction index n appeared to be insensitive to both correlating factors.

A new device for determining combustion response has been undergoing preliminary feasibility testing. The excitation chamber consists essentially of a long tube with an injector at one end, an oscillating piston at the other end, and a central, annular exhaust nozzle. In the preliminary testing program, the piston was replaced by a fixed end. Combustion oscillations were obtained with two different chamber configurations. An approximate analysis based on the Sensitive Time Lag Theory showed that the test results were in agreement with the n, τ correlations of full scale experimental data.

Injector-face baffles have been used for stabilization of transverse modes for several years although the mechanisms governing their effects have not been determined. Recent work, both in the acoustics laboratory and on large engine development programs, has indicated three basic baffle effects: (1) blockage, or protection of the injector face region against transverse gas motion, (2) reduction of the frequencies of lower-order transverse modes, and (3) damping or scattering of wave energy from lower into higher modes.

PARAMETRIC STUDY OF ROCKET MOTOR INSTABILITY*

M. J. Zucrow, J. R. Osborn, G. M. Lehmann, D. W. Netzer, P. J. Goede, L. T. Wettersted

Purdue University

The experimental investigation is concerned with a liquid fuel, gaseous oxidizer rocket motor system. The rocket motor employed will use bi-phase propellants, RP-1 and air, and will operate at a nominal thrust of 1000 pounds. A two-dimensional slab motor will be utilized for photographing the combustion process across a cross-section of the motor, and for facilitating the measurement of the temperature of the gases in the chamber. Initial experimentation will be concerned with the following parameters:

(1) momentum ratio of oxidizer to fuel streams

(2) combustion pressure (100, 300, 500 psia)

(3) orifice diameter (initial droplet size)

(4) pattern density

(5) mixture ratio (stoichiometric $\pm$ 10%)

The data obtained will be utilized in the parametric analysis which now includes the data collected from the experimental work using premixed and unmixed gaseous propellants.

The analytical prog am includes a linear analysis as well as a non-linear analysis of combustion pressure oscillations in a rocket motor.

In the linear analysis the rocket motor is treated as a constant-diameter duct with a finite width of combustion zone located between two uniform regions. It is also considered to have finite wave-transmission coefficients at the nozzle and the injector ends. For these conditions, stability criteria are established for the longitudinal mode of pressure oscillations. It is shown that, while the energy release per unit width of the combustion zone is significant in determining whether the motor will be stable, the location of the combustion zone in the motor, the over-all length of the motor, and the total amount of energy released also influence the stability of the motor in the longitudinal mode.

In the non-linear analysis the propagation of a plane, shock-fronted pressure pulse through a combustion region is investigated. The combustion region is represented as a one-dimensional flow regime in which temperature and concentration gradients and heat generation are present but in which transport processes are neglected.

An analytical solution is obtained in which the change in strength of the pulse as it propagates is given as a function of the original strength of the pulse, the incremental changes in the flow properties, and a nondimensional heat generation parameter.

Results calculated by employing the aforementioned analytical solution were found to be in good agreement with results obtained by numerically intergating the governing conservation equations.

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* Research sponsored by AFOSR under Grant AFOSR 753-65.

INTENTIONAL OSCILLATIONS IN COMBUSTION SYSTEMS*

J. Swithenbank

University of Sheffield

The application of intentional combustion oscillations to the generation of M. H. D. power has been studied both theoretically and experimentally. Theoretical studies have shown that the temperature modulation during instability can be used to increase the ionization in an M. H. D. system due to the exponential nature of the ionization/temperature relation. This leads to doubling of the generator power density for high amplitude instability waves. The limiting instability amplitude is a detonation wave, however the theoretical studies show that M. H. D. interaction distance is then reduced to such an extent that it is not possible to remove an appreciable (~1%) amount of the detonation wave energy as electricity. An experimental unstable rocket motor has been designed and operated using nonlinear jet mixing to give unstable combustion. High amplitude tangential waves can be obtained readily with this device and measurements of the ionization are now being made to confirm the theoretical predictions.

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* Research supported under E. O. A. R. Grant No. 65-23

SUPERSONIC COMBUSTION RESEARCH*

J. Swithenbank

University of Sheffield

A new phase started in January 1965 is aimed at the study of supersonic combustion with particular emphasis on instrumentation techniques. A large (6") shock tunnel specially designed for supersonic combustion research is in use for this work and a variable geometry two dimensional test section is being built. A large electromagnet provides 5000 gauss across the supersonic combustion chamber, and the voltage generated on probes inserted into the flow will give both velocity profiles and the axial change in velocity due to the combustion. Other instrumentation techniques which are being developed include high speed (1 millisec) gas sampling and optical methods. Studies are also being carried out to determine the effect of elimination of the tunnel dump tank for supersonic com combustion studies.

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*Research supported under E. O. A. R. Grant No. 65-23

COMBUSTION INSTABILITY TECHNOLOGY
AT THE MARSHALL SPACE FLIGHT CENTER

Robert J. Richmond

Four technology programs dealing with combustion instability are presently being pursued in-house at the Marshall Space Flight Center. These are: (1) An acoustic Liner Study, (2) F-1 Gas Generator Study, (3) 30K Combustion Dynamics Study and (4) High Pressure Combustion Investigation with Hydrogen-Oxygen.

Acoustic Liner Study

The performance and stability characteristics were determined for both a cooled and uncooled acoustic liner in a 4,000 pound thrust oxygen-RP-1 rocket engine. The liner design was based on an analysis by Pratt and Whitney under a Marshall Space Flight Center Contract. The tests were conducted at a nominal chamber pressure of 1000 psi over a mixture ratio range from 1.8 to 2.6 using a concentric tube injector with known instability characteristics. Combustion in all tests using the liner was stable.

F-1 Gas Generator Study

Two basic injector types are being investigated for the purpose of reducing combustion oscillations and improving performance. These are the swirl cup and concentric tube.

A full scale concentric tube injector for the F-1 gas generator has been designed, built and tested at the Marshall Space Flight Center. The individual elements consist of a concentric orifice operating at a mixture ratio of 1.0 with additional fuel to reduce the overall mixture ratio to below 0.5 provided by two fuel jets impinging on the propellant emitting from the concentric orifice. A total of two hundred twenty-eight of these elements are used. Fifteen tests have been conducted to date. Performance is comparable to that observed by Rocketdyne in their tests with impinging jet injectors. Combustion oscillations have also been observed under certain conditions.

The swirl cup is being evaluated as a single element in a small motor about two and one-half inches in diameter and ten to 20 inches long. The element itself is a double concentric swirl cup with the two propellants swirling in opposite directions. Thirty-seven of these elements would be used in a full scale injector. The performance observed is being compared to the produced by an impinging jet model of an early F-1 gas generator injector.

30K Combustion Dynamics Study

The effect on stability of fuel orifice diameter and fuel injection velocity is being investigated in a 30K motor using liquid oxygen and RP-1. Three like-on-like impinging jet injectors have been designed and fabricated. Two have identical fuel orifice diameters but different total areas, and two have identical total areas but different diameters. Oxidizer injection velocity is being maintained constant in all three injectors. The objective is to separate the effect of injection velocity from injection stream size. This will be done by attempting to determine the sensitive time lag and interaction index for each injector and comparing the results.

High Pressure Combustion Investigation with Hydrogen and Oxygen

The combustion characteristics of the hydrogen-oxygen system at high chamber pressure (3000 psi) are being explored in a 13K motor. Current testing is being conducted with gaseous hydrogen and liquid oxygen. Liquid hydrogen will be introduced at a later date. The current hardware consists of a transpiration cooled wafer chamber assembly with a rigimesh face, concentric tube injector.

SOME CONSIDERATIONS OF THE DESIGN OF SOUND ABSORBERS FOR ROCKET ENGINE COMBUSTION CHAMBERS*

Harvey J. Ford

Pratt & Whitney Aircraft
Florida Research and Development Center

Sound absorbing liners have been applied to rocket engine combustion chambers to suppress the pressure oscillation due to unstable combustion. The demonstrated success of acoustic liners in this application bears further evidence of the effectiveness of absorbing devices in ensuring stability in aeroelastic systems. The principle of operation and equations governing the performance of resonant absorbers is presented herein. Limitations of the performance calculation procedure exist due to a lack of knowledge of the nonlinear effects of intense sound and high velocity flow of the acoustic impedance of sound absorbers. The dependence of absorber performance on the properties of the sound medium requires a knowledge of temperature in and near the apertures of the absorber. Analyses have been made of the cooling requirements of sound absorbing liners installed in the combustion chamber of several hypothetical rocket engines. The results of the analyses indicate that sound absorbers can be adequately cooled in rocket engines having regenerative cycles.

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*This work was performed under contract NAS8 11038.

A THEORETICAL AND EXPERIMENTAL STUDY OF THE JET MIXING PROBLEM IN THE PRESENCE OF A SOURCE OF VORTICITY AT THE ORIFICE*

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Dublin, Ireland

It is postulated that high 'g' vibration of an injector plate produces vortex shedding at the orifice and so forms a source of vorticity in the gas jet.

Initial Experimental work on

(a) the mixing of such a jet in air - hot-wire technique

(b) the burning of such a jet in air - spectroscopic and photographic techniques

will be presented.

A theoretical analysis of the case of a jet expanding into stationary air in the presence of such a source of vorticity will be presented. Both the case of the far-field (beyond the potential core) mixing region and the near field region (which includes the potential core) will be dealt with. The numerical technique used to determine the mean velocity component distributions will be described.

It is intended to extend this analysis to the case of a chemically reacting, or burning jet, and preliminary results of this work will be given.

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*Research supported by the U. S. Air Force under Grant E. O. A. R. 65-43

ANALYSIS OF STEADY-STATE ROCKET COMBUSTION OF GASEOUS HYDROGEN AND LIQUID OXYGEN WITH COAXIAL JET INJECTION*

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A complete combustion model is described. Simultaneous equations describing propellant injection, atomization, mixing, vaporization, and combustion are formulated for a cylindrical liquid oxygen jet surrounded by a gaseous hydrogen stream. Required model input parameters are those normally known for a combustor: injector face area, propellant injection areas, flowrates and temperatures, chamber pressure and combustion chamber geometry.

The formulation is based on division of the combustion chamber into a non-burning region near the injector and a combustion region further downstream. Upstream circulation of combustion gases into the non-burning region is presumed not to occur, and a criteria is calculated to indicate whether or not that is valid for the particular combination of input parameters.

In the non-burning region, a portion of the liquid oxygen jet is atomized and mixed in a controlled manner with the surrounding hydrogen stream. A cylindrically annular, layered spray structure is hypothesized with the earliest formed spray occupying the outer layers. The hydrogen stream is decelerated and expanded radially by diffusion and mixing with the surrounding (stagnant) gases and by momentum exchange with the contained liquid oxygen spray. Spray vaporization effects a gradually increasing oxygen concentration in the hydrogen stream.

The gases in the non-burning region are considered to be combustible if their hydrogen concentration is lower than the upper flammability limit and if their velocity is lower than the local turbulent flame speed. When these conditions are satisfied, flame spreading is rapid enough that the non-burning region is terminated by a plane flame front standing in the spray-laden gases. The model's combustion region begins at that position in the chamber.

A portion of the gases in the non-burning spray zone is assumed to be diverted around the flame front and form a fuel-rich gaseous mantle around a central, well-mixed, one-dimensional spray combustion zone. The progress of propellant combustion can be limited by these processes: (1) atomization of the residual liquid oxygen jet that penetrates into the combustion zone, (2) vaporization of liquid oxygen droplets, and (3) turbulent mixing of the gaseous mantle into the spray combustion zone.

Few data exist concerning some of the processes, so the model formulation includes a number of arbitrary constants and assumptions. Selected values for these were modified until a combination was found that would permit the model predictions to reproduce experimental information from two particular transparent model rocket motor tests. The results from those best fit model calculations are presented in enough detail to characterize prominent features of the model predictions.

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*Research supported by AFOSR under Contract AF 49(638)-817

COMBUSTION INSTABILITY STUDIES*

M. Gerstein and M. R. Beltran
Dynamic Science Corporation

The principal objective of this study has been the development of a combustion instability model based on droplet combustion and emphasizing droplet dynamics. Previous reports have dealt with the motions of droplets in an oscillating flow field and the relationship between droplet drift velocity and droplet size. Droplet drift has then been related to combustion stability.

During the past year, the model has been extended under AFOSR Contract Number AF49(638)-1552 to include the combination of droplet dynamics and droplet combustion as a step toward establishing stability zones and the relationship between combustion stability to injector variables, particularly those related to droplet size and velocity. In the process of performing the analysis, it has been found that the relative droplet velocity, droplet Reynolds number and droplet burning rate appear in the equations as primary variables.

In a program supported by the Rocket Test Laboratory at Edwards Air Force Base under Contract Number AF04(611)-1-542, the results of the analytical study are being applied to experimental pulse motor data. The experimental data are obtained and reduced by Rocket Test Laboratory personnel. A correlating parameter which we have called a stability index has been applied to data obtained at Edwards and to some data available in the literature. Reasonable correlation of the data with this index have been obtained and the trends predicted theoretically are supported qualitatively.

Future work will be concerned with refining the theoretical model to obtain quantitative relationships between injector variables and stability as well as the application of these quantitative relationships to pulse motor data. These studies will include the effects of propellant type, atomization characteristics and scaling.

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* Research supported by U.S. Air Force under Contracts AF49(638)-1552 and AF04(611)-10542.

COMBUSTION INSTABILITY PROGRAM ACTIVITIES

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California

The combustion instability activities currently being conducted at or sponsored by the Air Force Rocket Propulsion Laboratory are briefly presented and discussed. These activities include five in-house program tasks and three contractual efforts:

In-House Program Tasks

(1) Spray Analysis Investigation

(2) Pulse Motor Combustion Instability Investigation

(3) Transparent Motor Combustion Investigation

(4) Combustion Correlation Investigation

(5) Transtage Combustion Stability Evaluation

Contracted Program Tasks

(1) Periscope Feasibility Investigation

(2) Drop Size Distribution and Combustion Instability

(3) Stability Rating Techniques Evaluation

All of these program tasks represent exploratory development efforts on the problem of combustion instability. They include the evaluation of combustion systems (injectors, thrust chambers and propellants) and the development of technology oriented toward specific applications. In general, the work is conducted with storable propellants. However, limited test experience with LO_2/RP-1 propellants are reviewed.

In addition, selected film clips of the combustion field as viewed through the optical periscope as well as that viewed through a transparent motor are shown.

RESEARCH STUDY OF LIGHT EMISSION CAUSED BY PRESSURE FLUCTUATIONS IN ROCKET MOTORS*

B. Hornstein

Thiokol Chemical Corporation
Reaction Motors Division

Simultaneous high frequency measurements of light emission and pressure were made in an oscillating premixed methane-air, 3 inch id X 16 inch long rocket engine, at pressures from 50 to 150 psig. Using filtered photomultiplier tubes, emission from the OH (0, 0) band (3064A), the CH (0, 0) band (4315A), and total luminosity ($CO + O = CO_2 + h\nu$) over the range of the IP28 phototube were monitored.

Depending on injector arrangement, longitudinal oscillations of nominally 1150 cps and transverse oscillations of mainly 7100 cps and 14000 cps (first tangential and first radical) were encountered. With transverse oscillations, no corresponding emission fluctuations were observed, which may be owed to the aspect angle of the detectors, 90° to the engine axis (0° to the plane of pressure propagation). The longitudinal pressure oscillations were always accompanied by emission fluctuations of the same frequency. Emission from the chain-branching radicals OH and CH lagged the pressure by about .2 milliseconds except at 150 psig where the lag was zero. At 50 and 100 psig, the total luminosity was in phase with pressure.

Visible spectra were photographically recorded, and showed similarity to previous laboratory spectra of flames as a function of pressure.

These spectra and the high frequency photomultiplier data are interpreted to show that the CH and OH emission fluctuations are indicative of pressure response of the reaction in the early chain-branching phase, and that the CH response to pressure is different for the high frequency record and the time-averaged photographic spectrum indicating non-equilibrium kinetics under transient conditions.

During the evolution of a satisfactory experimental engine, observations were made on the relationship between design and the occurrence of oscillations.

Although the gas-fed engine is not a true model of real-engine instability, and studies with liquid injection should be made, it is useful for the optical study of kinetic aspects of the coupling processes, and as a means for separating the various physical and chemical components of the complex pressure-combustion interaction.

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*Research sponsored by AFOSR underContract AF49(628)-1274

THERMAL RADIATION OF HIGH TEMPERATURE COMBUSTION PRODUCTS*

T. G. Rossman

Bell Aerosystems Company

The purpose of this program is to amplify the scarce existing information concerning those physical properties of gases which determine their thermal radiation, especially at high temperatures and pressures.

The experimental approach consists in determining directly by means of emission radiometry and spectroradiometry, the magnitude and spectral distribution of the thermal radiation. These measurements will be conducted on gases which appear as or at least are contained in the combustion products of actual combustors, e.g. rocket combustion chambers, and will extend over ranges of temperature, pressure and radiating path lengths as close as possible to the ranges encountered in practice in such combustors. This information will enable the designer to predict, with greatly improved accuracy, the radiant heat flux from the combustion gases to the walls of the combustor.

In addition to this direct approach, the experimental results will be utilized for checking and, if necessary, improving the reliability of methods suggested by several authors for predicting the thermal radiation of gases for values of radiating path length, temperature and pressure beyond the limits of the range in which measurements have been conducted.

It is recognized that, for this purpose, experimental work at elevated temperatures, at elevated pressures, and at elevated temperatures and pressures has been done and/or is being conducted by other investigators on some gases.

However, this program will permit experiments over much larger ranges of temperature and pressure and on gas mixtures actually present in combustion chambers. These investigations are also expected to give additional basic information concerning the rotational-vibrational bands of various molecules, for example, the dependence of bandwidth and integrated absorption on temperature, pressure and the presence of other species.

The technical approach and the experimental results obtained from investigations conducted at atmospheric pressure on carbon monoxide produced by the oxyacetylene flame are described. Measurements were made on 5 and 10 centimeter radiating path lengths of CO at 0.61 atmospheres partial pressure and 3300°K (5500°F).

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* Research sponsored by the U. S. Air Force under AFOSR Contract AF49(638)-1498.

SPECTROSCOPIC DIAGNOSTICS OF INHOMOGENEOUS HOT GASES AND PLASMAS*

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The working fluid in a real propulsion system is inhomogeneous. Variations in space and time of temperature, composition, and other fluid properties affect system performance. We have developed and applied optical spectroscopic techniques to analyze properties of inhomogeneous hot gases and plasmas quantitatively. The following specific problems have been attacked:

(1) determination of temperature distributions in hot gases and plasmas by line-of-sight quantitative infrared spectroscopy (1, 2), as well as by geometrical inversion (3, 4);

(2) determination of radiant heat transfer coefficients of inhomogeneous hot gases from radiant heat transfer coefficeints of homogeneous gases (5, 6),

(3) infrared gas analysis *in situ*, without sampling (7, 8);

(4) measurement of the time-dependence of temperature, composition, and internal energy of shock-heated gases (9) and transient solid propellant flames (10).

Problems (1) - (3) have been solved in principle, and applications are being studied. Techniques to solve problem (4) have been developed and preliminary results obtained.

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*Supported in part by AFOSR, (N49(638)-1132), NASA and ARPA

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PROPULSION CHEMISTRY RESEARCH

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The Chemistry Research Laboratory pursues a rather modest in-house research program in propulsion chemistry conditioned by the availability of military and civilian personnel and the fields in which they are technically competent. Current problems under investigation are concerned with flame kinetics, reactions in shock and detonation waves, heat transfer in dissociating gases, and non-Newtonian flow. With but one exception, these will be described in highly abbreviated fashion.

In the flame kinetics area an examination is being made of the effect of additives on the hydrogen-chlorine flame. The chlorides of the Group IV elements, which are very effective inhibitors for hydro-carbon and hydrogen-air flames, are found to exert a rather surprising promotional effect in hydrogen-chlorine mixtures. Current evidence indicates that the promotion may arise from a decomposition reaction of the additives.

The research on detonation waves seeks to determine the effect of finite relaxation and reaction rates on the structure of the reaction zone. It is still in the equipment construction and refinement phase. A preliminary examination has been made of the induction zone in the hydrogen-air reaction in order to check out the operation of a chemical shock tube.

The non-Newtonian flow investigation has been primarily analytical in nature. Topics treated have included laminar flow of non-Newtonian fluids in cylindrical tubes and conical sections. In the course of this work, a finite difference analysis has been developed for the laminar flow of viscous Newtonian liquids in conical tubes and a new generalized viscosity model has been proposed for non-Newtonian fluids. Currently an attempt is being made to solve the problem of the fall of a sphere in a gel-like material in order to obtain a valid criteria for measuring the yield stress of a gel.

To be discussed in slightly greater detail will be the investigation of heat transfer behind a reflected shock wave in a dissociating gas. A shock wave reflected from the end plate of a shock tube leaves behind it a plug of more or less dissociated gas at a uniform temperature and pressure. As the gas in contact with the end plate begins to cool, convection and diffusion currents are induced which augment the conductive heat transfer. An analysis has been made which relates the free gas temperature to the temperature rise measured at the end plate. It differs from previous treatments of the problem in considering the convective contribution to the heat transfer. By measurement of the temperature rise of the end plate by thin film thermometry and estimation of the free gas temperature from measured shock velocities, the analysis permits a check to be made of the calculated thermal conductivity of a dissociating gas over a range of temperatures. Experimental results obtained to date will be presented.

RESEARCH IN PROPULSION

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Studies in the field of propulsion that have been conducted at AFIT have been largely in support of graduate students. Areas of investigation have been oriented towards the type that give the student an opportunity to select a faculty suggested topic, plan an investigation, assemble and/or have constructed the apparatus, conduct the experiment, and write a report - all resulting in partial fulfillment of the requirements for the degree of Master of Science in Astronautical Engineering or Aerospace Engineering. Recently, faculty research has been initiated that will supplement these efforts. The areas of investigation are briefly described in the following paragraphs.

Twelve theses have been conducted on a reverse-flow film cooled rocket motor using gaseous H_2 and O_2. The investigations included studies to determine the effect of chamber pressure, mixture ratio, characteristic length, and injector design on overall engine performance.

An investigation has been conducted to determine the effect of parallel injection of a secondary fluid on a two-dimensional coanda-effect nozzle used as a thrust vectoring device.

Another investigation has been conducted on the performance of preheated, gaseous methane and oxygen as the propellant for propulsion systems. This study will continue on to include the use of CH_4 - H_2 - O_2 as the propellant.

Faculty research has been started in the area of supersonic combustion. Both the detonative and mixing types of supersonic combustion are being investigated.

Finally, in order to improve the facilities that will be used for future research, a thorough study and design have been made for an advance propulsive research laboratory for AFIT.

ON THE GENERATION OF PRESSURE PULSES IN REACTIVE MEDIA *

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One of the particularly interesting modes of combustion instability which takes place especially in large rocket thrust chambers is that associated with local explosion of the burning substance. In technical literature it is referred to as the self-triggered or surge instability. The outward dynamic manifestation of such an event is the appearance of a pressure pulse. This can take up two quite distinct forms. One is associated with the growth of the process, when the accelerating reaction zone generates a pressure wave that oropagates ahead of it and preconditions and unburned substance, before it becomes subjected to chemical reaction. The other occurs when the flow field is already affected by the action of shock waves, and makes its appearance in the form of an "explosion in an explosion". In a reactor vessel, where the medium is kept initially at rest, the former takes place in the course of initial flame acceleration and develops gradually while the combustion process is deflagrative in character, and the latter develops abruptly and leads to the onset of detonation. The paper summarizes our studies of the gasdynamic aspects of these phenomena.

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* Research supported under NASA Grant 129-65

FORMATION OF DETONATION WAVES IN FLOWING COMBUSTIBLE GASEOUS MIXTURES *

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The Ohio State University

Research has been completed on the studies concerning the formation of detonation waves in flowing hydrogen-oxygen and methane-oxygen mixtures. Induction distances were measured for various linear flow velocities in a 9-mm inner diameter tube. The flow velocities ranged from 0-100 m/sec in hydrogen-oxygen mixtures. For methane-oxygen mixtures, the flow velocities ranged from 0-30 m/sec.

The induction distances were determined both upstream and downstream from the ignitor, which was located near the middle of the tube. Usually a Pyrofuse ignitor was used because of its excellent performance. However, some catalytic action occurred between the hydrogen and the Pyrofuse wire for a mixture of 55 per cent hydrogen, flowing at 20 m/sec at five atmospheres initial pressure. Satisfactory results were obtained with copper ignitor wire.

The induction distance for a particular set of experimental conditions appears to be a function of the Reynolds number of the initial gas flow. When the nondimensional induction distance is compared with the logarithm of the Reynolds number based on the initial linear gas velocity, a fairly linear relationship is obtained. For a 75 per cent fuel concentration (H_2-O_2 mixture) and for flow velocities ranging from 10-30 m/sec (corresponding to Reynolds numbers ranging from 2250 to 6750), the flame propagation rate measured upstream of the ignitor deviated from the straight-line relationship obtained from all other experiments. The deviation may have been casued by the flame propagating upstream, for this particular fuel concentration and linear flow velocity, in such a manner that tunnelling developed wherein the flame propagated upstream through the central core of the unburned mixture and the flame did not propagate toward the wall uniformly. Thus, the probes sensed longer time intervals which gave rise to an apparent longer induction distance. Another possibility is that, with this small tube, the flame became rather cool near the wall because of the particular flow conditions during this experiment and sufficient ionization was not present for the probes to respond and trigger the chronograph.

For all other combinations of fuel concentrations and linear flow velocities, the straight-line relationship between induction distance and the logarithm of the Reynolds number held fairly well. At an initial pressure of five atmospheres, relatively few experiments were conducted because the induction distances became very short. The results for the mixture at one atmosphere clearly indicate that increasing the initial flow felocity and, therefore, the turbulence level is equivalent to increasing the initial pressure.

The induction distances for the methane-oxygen mixtures did not decrease with increasing linear velocity as did those for the hydrogen-oxygen mixtures, they remained fairly constant for the initial flow conditions employed. For gas velocities from 0-10 m/sec and at low ambient temperatures (5-10°C), it was difficult or sometimes impossible to ignite the methane-oxygen mixture containing 50 per cent methane. When the flow rate was increased to 20 and 30 m/sec, ignition was accomplished easily and the data for the flame propagation rates were repeatable.

With a 9-mm diameter tube, probably the quenching distance was reached for detonation waves in this mixture. A flame existed in the mixture which propagaged into the exhaust vent where a detonation developed. Because of quenching, however, the ionization probes did not respond. No information was found in the literature concerning the quenching diameters for detonations in this fuel-oxidizer mixture, it is known that the quenching diameters for detonation waves is greater than that for flames.

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*Research sponsored by U, S. Air Force under AFOSR - 203-65 Grant.

RESEARCH ON STABLE, NORMAL AND OBLIQUE SHOCK-INDUCED SUPERSONIC COMBUSTION

R. P. Rhodes

Arnold Engineering Development Center

The more recent work done at Arnold Engineering Development Center in this field can be divided into two phases: (1) oblique shock-induced combustion in constant area ducts, and (2) normal shock-induced combustion for kinetic studies.

1. Oblique Shock Experiments.

These experiments were run in an axisymmetric model which captures the flow behind a reflected oblique shock system. It was possible to add limited amounts of heat to the stream and have the flow remain supersonic. These results were compared with calculated hydrogen air reaction kinetics, with reasonable agreement with regard to the ignition delay, but much poorer correlation in the heat release portion of the reaction.

2. Normal Shock Kinetic Experiments.

Recently, normal shock-induced combustion experiments have been run with the reaction in a constant area tube with an attached normal shock on the lip. The objective of these experiments is to use the pressure rise resulting from heat release to predict the overall rate of the chemical reaction and use this to evaluate various chemical kinetic calculations. The normal shock system was chosen to avoid two-dimensional effects which occurred in the oblique system and complicated the analysis. This system seems to be working and some preliminary results have been obtained.

OTHER COMBUSTION ACTIVITIES

The other combustion activities at the Arnold Engineering Development Center are:

1. Chemical Reactions in Turbulent Systems With Mixing.

This work includes a theoretical study of turbulent mixing with large density and velocity gradients and the experimental work on rocket thrust augmentation and free jet mixing.

2. Measurements in High Temperature, High Speed Reactive Flows.

Experimental techniques are being developed to measure temperature, composition, and density in regimes which might be experienced in supersonic combustion.

RESEARCH ON SUPERSONIC COMBUSTION*

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General Applied Science Laboratory
New York

GASL effort under AFOSR Contract AF 49(638)-991 and its extension, AF 49(638)-1503, has been directed principally in the general fields of reaction kinetics of the hydrogen-air system, diffusive mixing of reacting gases, and experimental testing of combustion mechanisms.

The following principal results have been accomplished:

1. Empirical correlations have been developed which predict with good accuracy, the ignition delay time t_{ID} and the reaction time t_R. The results are of direct interest to the analysis of supersonic combustion processes.

2. The basic characteristics of supersonic combustion processes controlled by diffusion have been analytically investigated in configurations involving (A) the turbulent diffusion and combustion of a circular jet of hydrogen discharging into a stream of air parallel to the jet and (B) the diffusion and combustion of hydrogen injected into a high speed laminar flow by means of a wall slot.

3. Experiments were performed to determine spontaneous combustion characteristics of hydrogen injected tangentially into a high speed stream (speeds of the order of 10^4 ft/sec) using the GASL shock tunnel. These gave results consistent with the foregoing reaction rate predictions.

4. Experiments with applications to combustion controlled by the thermal transport process were performed in an axially symmetric premixed supersonic jet. An important detail of the experimental work was the development of a small pilot flame for producing stable ignition of the combustible mixture.

As a result it was concluded that a cold hydrogen-air mixture could be burned at supersonic speeds.

The flow field associated with the propagation of axisymmetrical flames from a pilot into a premixed hydrogen-air flow was studied using analytical and numerical techniques developed at GASL. Comparisons between experimental and computed values of the flame spreading rates indicated excellent agreement of the theory with experiments.

This current effort is being concentrated in (A) continuation of the research on ignition and flame propagation in fuel-air mixtures at temperatures below the auto-ignition levels, with applications to supersonic combustion. Typical fuels to be investigated will include hydrocarbons such as methane and propane, and (B) analytical and experimental studies of low density combustion processes are being conducted.

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*Research supported by U.S. Air Force under contracts AF 49(638)-991 and AF 49(638)-1503

ANALYSIS OF SUPERSONIC COMBUSTION EXPERIMENTS WITH GASEOUS HYDROGEN FUEL*

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The Johns Hopkins University
Applied Physics Laboratory

Previous analytical studies have shown an attractive potential for hypersonic ramjets utilizing supersonic combustion of hydrogen. In order to realize this potential, efficient fuel-injection systems and combustion chambers must produce high combustion efficiency with low entropy rise. Accordingly, the research program being conducted at the Applied Physics Laboratory for the National Aeronautics and Space Administration is directed toward understanding the basic phenomena that control the supersonic mixing and combustion of H_2 and air, in order to determine how the critical parameters may be varied to obtain an optimum balance between heat release and stagnation pressure loss. The discussion herein includes a description of the instrumentation and data analysis techniques used in conjunction with tests of a conical combustor. The combustor entrance conditions of $M_c = 2.8$, $T_c = 2500^{\circ}R$ static, and $P_c = 1$ ATM static, simulating high altitude flight at $M_o = 8.5$ are provided by a DC electric arc heater. Gas sampling, pitot and cone static pressure probes have been developed to operate in this severe environment. The data gathered from this instrumentation along with steam calorimeter, mass and pressure data serves as input into a rigorous steam-tube analysis of the combustor flow field. Results of a typical test are presented to indicate the validity of this approach for a complete description of the supersonic combustor performance including effective heat release and combustor entropy rise.

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* Research supported by National Aeronautics and Space Administration

DIFFUSION FLAMES AND SUPERSONIC COMBUSTION

Dr. I. da Riva

Instituto Nacional de Technica Aeronautica (INTA)
Madrid, Spain

To be supplied at a later date.

RESEARCH ON SUPERSONIC COMBUSTION

F. D. Stull

Ramjet Engine Division
Aero Propulsion Laboratory

In addition to sponsoring contractual efforts with Industry the Air Force Aero Propulsion Laboratory has been conducting in-house studies on the Supersonic Combustion Ramjet (Scramjet) for over five years. These studies have been primarily analytical efforts dealing with cycle analysis, component performance and heat transfer pertaining to Scramjet. Recent efforts in the experimental area have been associated with the design and calibration of a 3" shock tube for conducting chemical kinetic studies, and the acquisition of a 120" shock tunnel which is currently being installed by AVCO.

A brief discussion of a few of the results obtained in the analytical area are presented, along with a description of the experimental effort.

COMBUSTION OF OXYGEN DROPLETS IN A HYDROGEN ATMOSPHERE

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As an application of the general study of the H_2 - O_2 flame with spherical symmetry, which is being conducted under Grant AF-EOAR 65-70 the particular case of the combustion of oxygen droplets in a hydrogen atmosphere is presented in this work.

The problem is analytically studies by considering spherical symmetry and stationary conditions.

It is shown that in this case, in which the molecular masses of the reactant species are so different, a great care must be taken in selecting proper values of all physical properties: thermal conductivities, diffusion coefficients, etc. and that the influence of the composition of the gaseous mixture in them has to be taken into account.

In the first place the solution obtained by taking an infinitely fast reaction rate is calculated. It has been shown by the authors that this solution is an asymptotic value which gives approximated results provided that the pressure or the droplet radius are not too small.

In order to study the influence of chemical kinetics on the process, an overall reaction rate is required in our model. It was not possible to derive that overall reaction rate from the H_2 - O_2 reaction models proposed by several authors, and therefore, it is expected to obtain such reaction rate from the experimental values of the H_2 - O_2 flame with Spherical symetry which is presently being studies under the aforementioned Grant.

Gaseous oxygen is injected through a porous sphere and burned in hydrogen at different pressures. A theoretical treatment of this process has also been performed, and the comparison between theoretical and experimental results is until now in excellent qualitative agreement. This comparison will be also used to verify the theoretical results of the combustion of oxygen droplets in a hydrogen atmosphere, from which no direct experimental data are yet available.

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*Research supported by U. S. Air Force under Grant E.O.A.R. 65-70

FLAME PROPAGATION ALONG THE FREE SURFACE OF A LIQUID FUEL*

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Flame propagation along the surface of a liquid fuel in contact with air is studied in the present work. Initial temperature of the fuel is assumed to be under flash point temperature.

The analytical treatment of the process is carried out by considering stationary conditions with respect to a system of reference moving at the speed of the flame.

Heat transfer from the flame to the fuel is assumed to take place through radiation, although heat conduction within the liquid is also considered.

The film of liquid fuel is supposed to be supported by a homogeneus and conductive medium of thickness much larger than that of the fuel film.

An analytical solution of the problem is obtained by approaching the radiant heat transfer function by means of an exponential expression, and by disregarding heat fluxes within the liquid and within the conductive medium in a direction parallel to the fuel surface as compared to heat fluxes perpendicular to that surface. Flame propagation speed is given by the conditions that the flame reaches a given point when the temperature of the fuel surface at that point reaches flash point temperature.

Dimensionless results are obtained showing the flame propagation velocity as a function of fuel initial temperature, fuel depth, and for several values of the most important parameters of the process.

Temperature profiles within the liquid and within the supporting medium are also shown.

An experimental investigation has also been carried out by measuring flame propagation velocities and fuel temperatures in an open channel containing either kerosene JP-1 or JP-4. The supporting medium has been mainly water, although some tests have also been performed on cement.

The channel is 10 feet in length and of variable width. Flame depth can be kept constant.

Initial fuel thickness ranged from a few tenths to several millimeters and fuel initial temperature varied from -20°C to + 60°C.

Theroetical and experimental results are in excellent agreement. They show, for example, that flame propagation speed increases very rapidly as fuel temperature approaches flash point temperature, and that flame speed may increase or decrease as the fuel film decreases, depending on the ratio of thermal diffusivities of fuel and supporting medium.

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* Research supported by U.S. Air Force under Grant E.O.A.R. 63-44

INGITION AND BURNING OF SOL'D PROPELLANTS*

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The ignition response to high convective heat fluxes of a series of AP-PBAA propellants was determined. Analysis of the data obtained from these tests shows that the gas-velocity effect on ignition, often noted with convective heating, is the result of surface roughness. If the effect of surface roughness is eliminated by use of very smooth propellant surfaces or by use of very high gas velocities, the high-flux ignition data are consistent with the results of low-flux tests on the same propellant and no effect on the ignition times of oxygen in the heating gas is noted. Additional ignition tests have been made employing electrically heated wires cast into propellant samples as the ignition energy source. Ignition times can be accurately determined by observation of the voltage drop across the wire, and the ignition data obtained by this method are reasonably consistent with the results obtained by other techniques.

The polymer decomposition reactions which are important during the ignition process are being studied by exposing films of propellant fuel-binder polymer to thermal radiation and measuring the weight loss and the gaseous products composition as a function of termperature and heat flux. The purely thermal effects of this pyrolysis appear not to be very important, and rapid evolution of gaseous materials occurs only at temperatures high enough to be of interest in high-heat-flux ignition.

The extinguishment characteristics of catalyzed and uncatalyzed AP-PBAA propellants have been determined in tests in which a rarefaction tube was used to produce controlled rates of pressure decrease. Tests were made in which burning samples were extinguished by use of rapidly decreasing pressure with no gas flow near the propellant surface and by use of decreasing pressure and cold gas flows across the burning surface. The results of these tests are in general agreement with previously published data concerning the extinction of burning propellants.

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*Research supported by AFOSR under Grant 40-65

A STUDY OF IGNITION AND COMBUSTION MECHANISMS OF SOLID PROPELLANT SYSTEMS*

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The purpose of this work, which was started 1 April 1965, is to investigate by theoretical and experimental methods the ignition and combustion processes of selected oxidizers and composite propellants. The program encompasses an evaluation of possible chemical reaction mechanisms occurring in both the transient and steady-state domains. A principal objective is to develop consistent models of composite propellant ignition and oxidizer deflagration.

Work presented in this summary is centered on an exploration of the role of environmental chemical reactivity on ignition. A shock tube was used as an experimental tool and all tests were performed with flush-mounted end-wall samples. With this test configuration, conductive heating from the doubly compressed stagnant gas behind the reflected shock wave is the principal external energy source. Initial phases of this study have been completed and the principal results will be presented at this time

Furthermore, the scope of several separate research efforts being conducted at UTC will be discussed. These projects include studies of hypergolic ignition, flame propagation, steady-state combustion, combustion instability, and combustion termination.

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* Research supported by AFOSR under contract AF49(638)-1557.

IGNITION MECHANISM OF SOLID PROPELLANTS*

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Princeton University

The objective of this research program is to identify the basic processes that lead to ignition of a solid propellant when it is subjected to a source of heat and/or reactive gas, and to develop a theoretical framework for predicting ignition behavior under various conditions. Under earlier contracts evidence was collected in various types of experiments that led to a gas phase model of the ignition process, and a theoretical approach was developed for the prediction of ignition time lag as a function of pressure and oxygen content in the igniting gas.

Under the present contract, the following achievements can be reported. First, with respect to diagnostic experiments, a fast-response infrared radiometer has been applied to the measurement of the surface temperature transient of an igniting solid propellant. Another series of experiments of diagnostic value was the high speed photographic analysis of an igniting surface under shock tube conditions. A third diagnostic series was the study of the systematic effect of fuel vapor pressure on ignition time lag. Taken together these results tend to support the gas phase ignition model, although some unexpected phenomena require further interpretation.

On the theoretical side, a fairly complete analysis of the ignition process has been carried out which explains satisfactorily the observed ignition behavior under shock tube conditions. It is sufficiently comprehensive to permit extension to other conditions as well. These studies are being extended to see whether analytical criteria for ignition can be evolved to facilitate the application of this theoretical approach to more complex situations.

The research that will be reported is based on the theses projects of A. Azcarraga, F. J. Kosdon, and C. H. Waldman, and in addition, on the work of Dr. C. E. Hermance, Prof. R. Shinnar, and Prof. S. I. Cheng.

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* Research supported by AFOSR under Contract AF 49(638)-1268

RESEARCH ON THE DEFLAGRATION OF HIGH-ENERGY SOLID OXIDIZERS*

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Studies of the deflagration of hydrazine perchlorate (HP) and hydrazine diperchlorate (HDP) have been carried out, and various properties of these oxidizers pertinent to the question of their self-deflagration have been investigated.

It has been found that hydrazine perchlorate will deflagrate reproducibly if a few per cent of fuel is present. Deflagration rates have been measured photographically with cylindrical strands pressed to 95-98% crystal density for ambient pressures from 0.26 to 7.7 atmospheres. Steady deflagration could not be attained outside these pressure limits. Deflagration experiments were also performed with hydrazine perchlorate-catalyst mixtures.

Additional experiments were also performed: Temperature profile measurements of the deflagration wave by means of fine thermocouples indicate temperatures as high as 450° C in the condensed phase and are consistent with little heat release in the condensed phase. Vaporization rate measurements are consistent with a dissociative vaporization of HP. Spectroscopic measurements above deflagrating strands yielded a flame temperature of 2275 ± 50°K, in satisfactory agreement with the calculated value of 2224°K. A publication of these results is in press (1).

The study of hydrazine diperchlorate is in a more preliminary state than that of hydrazine perchlorate, but some meaningful results have been obtained. Unlike HP, strands of HDP (95% of crystal density) exhibit steady self-deflagration and this has been measured at pressures of 6 to 66 atm. However, the data are characterized by a high degree of scatter. As with HP, but unlike ammonium perchlorate, the deflagration proceeds with a liquid layer at the surface. A limited number of experiments in which small amounts of fuels have been added to HDP have been performed, and effects of some catalysts have been investigated.

The calculated adiabatic flame temperature of HDP is 1600°K. Attempts have been made to measure the flame temperature with fine thermocouples and to characterize the structure of the combustion wave through the gas and condensed phases. The results of these measurements will be discussed.

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(1) "The Deflagration of Hydrazine Perchlorate" to be published in Advances in Chemistry (ACS), by J. B. Levy, G. von Elbe, R. Friedman, T. Wallin and S. J. Adams

* Research supported by AFOSR under contract AF 49(638)-1169

RESEARCH ON HIGH DENSITY PLASMA PRODUCTION FROM CHEMICAL SOURCES*

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In an earlier phase of this study, electron generators functioning by combustion of a cesium nitrate-aluminum pressed charge were studied. Thermodynamic calculations, confirmed by experiment, showed that about 10^{20} electrons/gm can be produced (1). The spectroscopic line-broadening electron measuring technique has been published (2). These plasmas contain alumina droplets as well as ionized gas.

In a later phase of this study (3), a unique, self-oxidized combustion system, consisting initially of a solid mixture, was devised which burns to yield an all-gas plasma containing over 10^{15} (measured) or 10^{16} (calculated) electrons/cm^3 at atmospheric pressure. The ingredients are tetracyanoethylene, hexanitroethane, and cesium azide, formulated to be stoichiometric to CO, N_2, and Cs. The mixture is compacted to 1.6 gm/cm^3 by pressing at 20,000 psi. The strand burning rate at 1 atm is 1/3 mm/sec and the pressure exponent is about 0.6. The flame temperature, calculated as 3660°K, was measured as 3520°K by the spectral-line-intensity method and 3000° - 3100°K by the line-reversal method. In view of the radial non-uniformity of the plasma, the former value is considered more likely to be correct. Electron densities were determined both from measurements of individual spectral-line shapes (Stark-broadening) and by the Inglis-Teller series limit method. The two methods agreed with one another. Strands of 1.27-cm diam gave three or four times higher electron concentrations than strands of 0.79-cm diam; however, electron concentrations, while extremely high, were still below calculated values, even after the lowering of flame temperature below theoretical (because of heat loss) was considered. We believe that adequate time was available for cesium ionization. under our experimental conditions, but time was not adequate for complete oxidation of CN and CN^- to CO and N_2, so electrons were largely attached as CN^- rather than free. Hence, longer residence times allowing a closer approach toward equilibrium should give even higher yields.

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(2) L. W. Fagg, J. Quant. Spectroscopy and Rad. Transfer, March-April 1965

(3) R. Friedman and A. Macek, Tenth Symposium (Intl.) on Combustion, Combustion Institute, 1965

*Research sponsored by AFOSR under Contract AF49(638)-651

THE FORMATION OF SODIUM METABORATE*

David Fleischer

Thiokol Chemical Corporation
Reaction Motors Division

The equilibrium constant for the reaction

$$NaCl(g) + HBO_2(g) = NaBO_2(g) + HCl(g)$$

is determined to be, $K_p = 2.50$ (1980°K) Sodium vapor and boron trichloride (BCl_3) are reacted in a hydrocarbon-air flame contained in a tunnel burner. Conditions are such that the concentrations of HBO_2 and HCl are equal to the total flows of boron and chlorine. The concentration of unreacted sodium is found by line absorption, using an experimentally determined absorption coefficient. The relative contribution of NaCl and $NaBO_2$ to the chemically combined sodium concentration can be assessed by removing the direct proportionality between chlorine and boron flows. This is done by adding chlorine, beyond that present in the BCl_3. The H-atom concentration is also required, and is inferred from the intensity of the ^{2} ^{2} (0, 0) OH-band.

* Research supported by AFOSR under Contract AF 49(638)-1197

CONTINUOUS MEASUREMENT OF SOLID PROPELLANT BURNING RATES*

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The basic operating principles of an experimental system for the direct and continuous measurement of solid propellant burning rates are presented. The system has been designed and fabricated. Several components of the measurement system have been modified in order to increase the precision of the burning rate measurements.

A feasibility study was conducted for determining the adaptability of microwave techniques to the measurement of the burning rate of a solid propellant. It was concluded, because of the dependence of the microwave attenuation upon the combustion conditions present in a research rocket motor, that microwave techniques are not readily adaptable to such burning rate measurements.

A feasibility study indicates that a technique employing ultrasonic pulses can be developed for obtaining direct measurements of the burning rate of a solid propellant. The technique is based on measuring the time for an ultrasonic pulse to travel through a propellant sample.

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*Research sponsored by AFOSR under Grant AFOSR 207-64.

SHOCK INITIATION OF LOW DENSITY PRESSINGS OF AMMONIUM PERCHOLORATE*

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Initiation and detonation behavior of ammonium percholorate with a 50% grain diameter of 11 μ and a loading density of 1.00 g/cm^3 was studied. Ideal (or plane wave) steady detonation velocity was determined experimentally. Chapman-Jouguet detonation characteristics and the detonation product composition were calculated assuming the BKW equation of state. Initiation and build-up behavior was observed by introducing into pellets plane shocks at strengths somewhat larger than the minimum required to initiate detonation. The change in wave velocity from that of an initially unreactive wave to one travelling at C-J velocity was measured. Pressures in the entering wave and during its acceleration to plane wave steady velocity were determined by free surface velocity measurements on thin sheets of Plexiglas at the top surfaces of the pellets. Points on a shock Hugoniot for unreacted material were determined. The effect on build-up behavior of interstital air pressure and surface catalysts was measured.

Measured and computed steady-state characteristics are compared. Results are interpreted in terms of current theories of initiation, detonation, propagation, and behavior of shocked granular materials.

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* Research supported by AFOSR under Contract AF49(638)-1124

CAVITIES AND MICRO MUNRO JETS IN LIQUIDS: THEIR ROLE IN EXPLOSION

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Earlier work from this laboratory has shown that discontinuities in liquid and solid explosives play an important role both in the initiation of explosion by shock and in the growth of the explosion to detonation. The initiation is essentially a thermal process and the discontinuities concentrate the energy of the shock and produce a localized hot spot. These discontinuities or cavities in the explosive act in a variety of ways.

In this paper it will be shown that cavities of appropriate shape can produce micro munro jets which increase the velocity of impact and can also disperse the explosive as fine drops. The formation of these micro jets under various experimental conditions of impact and shock is demonstrated.

The presence and formation of cavities in liquid (and in solid) explosives is also of major importance in the growth from burning to explosion. Cavitation is produced by precursor waves which run ahead of the reaction zone. It will also be shown that the pressure developed in the reaction zone is effective in removing bubbles and cavities in the explosive directly ahead of it, so that the explosion cannot grow rapidly until the burning eventually succeeds in breaking through this homogeneous high-pressure region. This process is illustrated by high-speed photographs.

The major experimental difficulties in making these fast observations at high magnification have now been overcome.

We wish to extend this work to solid and plastic explosives, provided they can be made sufficiently transparent. To assist in this, we wish to use laser sources. We also propose to extend the work to higher speeds so that the detonation process itself can be studied in some detail. We wish to increase the magnification still further so that the role of micro-discontinuities can be investigated in more detail.

Research supported under Grant EOAR 65-24

NON-STEADY COMBUSTION CHARACTERISTICS OF SOLID PROPELLANTS WITH REFERENCE TO COMBUSTION INSTABILITY*

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The object of this experimental program is to investigate in detail the response of the combustion zone to an oscillating pressure field in a solid propellant rocket motor. Theroetical considerations based on previously accepted physical models of the flame led to the expectation that the interaction at low frequencies (ca. 100 cps) for normal propellants would be quasi-steady, and in most rocket designs this would not lead to instability. However, experimental evidence from other sources indicates that this is not the case. In our experiments under the previous contract period, with a t-tube rocket oscillator, observations of the flame zone under sinusoidal pressure variations did not show the expected quasi-steady interaction. It was decided to explore this phenomena further.

In the present contract period, two series of experiments are underway. The t-tube motor is being used to look for possible non-steady interaction effects with larger amplitudes of oscillation. A second apparatus consists of a vented combustion chamber designed to produce a steep pressure transient by a sudden change of throat area during burning. Oscillographic traces of flame luminosity coordinated with pressure seem to indicate, at this writing, that the interaction is not quasi-steady. If these preliminary observations can be confirmed by additional related tests now underway, the finding would be significant for low frequency rocket behavior. This report is based on the theses research projects of R. H. W. Waesche, H. Krier, and G. F. DiLauro.

* Research supported by AFOSR under Contract No. AF 49(638)-1405.

SOLID PROPELLANT COMBUSTION INSTABILITY*

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Work under ARPA and AFOSR sponsorship, monitored by AFOSR, has been concerned with both acoustic and non-acoustic instability of solid propellants. The results reported here, the fruits of the efforts of Mr. Carl Oberg and Mr. Merrill Beckstead, will be published shortly as special reports.

Acoustic Instability

In a project aimed originally at utilizing the pressure and phase measurements at the ends of the gas column in a side-vented end-burner to calculate the response function of the burning surface, a new characteristic of the burner was discovered. To our knowledge, no other experimentalist has recognized it, though it must have been encountered before and has probably misled some investigators into erroneous interpretations of their results.

When the burner is operated with a burning grain in only one end, after a short initial period of operation the acoustic pressure amplitude at the non-burning end becomes appreciably greater (factor of two) than that at the burning end. This fact is interpreted as indicating that a signal from the non-burning end is reactively reflected with reversal of sign from a region in the neighborhood of the centered critical flow nozzle. There appears to be an acoustic trap in the non-burning end.

No explanation of the effect is offered at this time. The conventional use of the burner, to measure growth and decay constants, is, as far as the acoustic trap is concerned, valid.

Non-Acoustic Instability

The work in non-acoustic instability has been carried out chiefly in a low-L* motor closely resembling its older cousin, the NOTS E. V. E. burner. A family of propellants, based on an 80/20 AP-PBaa composition, varying in catalyst and aluminum content have been tested. They are found to have nearly the same behavior, when described in terms of the appropriate dimensionless frequency and L* variables.

An analytical description of low-frequency instability in a low-L* motor has been developed along the lines of classical perturbation theory. The description places some of the problems of non-acoustic instability in useful, perhaps constructive perspective.

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* Research supported by AFOSR under Grant 446-65

SOLID PROPELLANT ROCKET COMBUSTION*

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In a previous report ** it was stated that our studies indicated that wave triggered thermal explosions could be the energy source for high frequency acoustic waves in the combustor while processes such as mixing at the surface might be associated with lower frequency wave phenomena. A model based on "erosive burning" phenomena associated with mixing processes within the granular diffusion flame was developed for composite propellants; this model had the same response to pressure and burning rate as we have experimentally observed for a large number of "state-of-the-art" high energy composite propellants. (For double base propellants the critical time parameter might be related to the fizz burning reaction.)

Satisfactory as our model has appeared to be in explaining our experimental results in the pressure/burning rate plane, it was nevertheless considered desirable to examine the model further with a view to confirming the over-all validity of the formulae developed. Preparatory work on several novel propellant formulations has been carried out, this is aimed at verifying source spacing effects, wave travel time factors, and the burning rate relationship over the entire burning rate spectrum. (Oxidizers other than ammonium perchlorate are also being investigated.)

Experiments with an opposed slab motor, fitted with closely coupled piezo-electric pressure transducers at four stations along the length of the motor, have been performed. After a short transient period, the observed wave pattern was that of a complex standing wave of roughly constant amplitude. Steady oscillations of this type are typical of strongly damped nonlinear systems.

Investigation of phenomena associated with double base propellants is continuing. A systematic study of their macro-ballistic, thermochemical and steady state combustion properties is under way. This will seek to correlate the instability behavior of this type of propellant with specific propellant properties.

It is hoped that the investigations under way will lead ultimately to a better understanding of those physico-chemical processes, occurring at the burning surface, which result in amplication of pressure waves. Furthermore, the results should also help guide rocket designers and propellant formulators in developing rocket motors which are inherently stable.

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* Research supported by AFOSR under Contract AF 49(638)-1367

** Annual Report, Research on Unstable Combustion in Solid Propellant Rockets, Stanford Research Institute, January 13, 1965

FUNDAMENTAL PROCESSES OF SOLID PROPELLANT ROCKET COMBUSTION*

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Shock tube techniques are being utilized to determine the response of a burning solid propellant surface to pressure perturbations of small, but finite amplitudes. This method of obtaining responses of various propellants will be described with emphasis upon the techniques developed to remove undesirable temperature gradients. Other experimental techniques utilized to obtain high frequence response pressure measurements under reproducible pressure, flow and temperature conditions will be described. The propellants which have been used in this study include those (both aliminized and non-aliminized) with a wide range of burning rate exponents as well as absolute burning rate. Most of these experiments have been conducted about 400 psia ambient pressure.

A discussion of the handling of the experimental data to obtain Fourier Transforms and the resultant response as a function of frequence will be presented, with particular emphasis upon significance and reproducibility of the data.

A comparison of these results with empirical instability data as well as prevailing theoretical treatments will be presented. Particular emphasis will be placed upon the relationship of the response measured in this program to reflection coefficients and spacific acoustic admittance (impedance) as measured in various other experimental techniques and as predicted by analysis.

* Research supported by AFOSR under Contract AF49(638)-1208.