ORDER OF PRESENTATION

- SOLID PROPELLANT REGRESSION RATE MEASUREMENT
- MICROWAVE DOPPLER SHIFT TECHNIQUE
  (FOR MEASUREMENT OF PROPELLANT COMBUSTION RESPONSE FUNCTION)
- PLASMA CAPACITANCE GAGE TECHNIQUE
  (FOR MEASUREMENT OF EROSION AUGMENTATION OF PROPELLANT BURNING RATE)
- HYBRID ROCKET COMBUSTION INSTABILITY MECHANISMS
- MINIATURIZED COMPONENTS FOR FUTURE SPACECRAFT PROPULSION SYSTEMS
MICROWAVE DOPPLER SHIFT
PROPELLANT BURNING RATE MEASUREMENT
TECHNIQUE
REVIEW OF COMBUSTION RESPONSE MEASUREMENT TECHNIQUES

ROCKET MOTOR STABILITY PREDICTION

- ROCKET CHAMBER PRESSURE DISTURBANCE VARYING SINUSOIDALLY IN TIME

\[ P' = \hat{P} e^{i\omega t} e^{\alpha t} \]

- EXPONENTIAL COEFFICIENT \( \alpha \) HAS GENERAL FORM

\[ \alpha = R_{PC(r)} I_{PC} + R_{VC(l)} I_{VC} + \alpha_{FT} + \alpha_{PD} + R_{N(r)} I_{N} + \alpha_{ST} \]

WHERE: 

\[ R_{PC} = \frac{m'_b / \bar{m}_b}{P' / \bar{P}} \]
\[ R_{VC} = \frac{m'_b / \bar{m}_b}{v'/\bar{a}} \]

- Definition of solid propellant pressure-coupled combustion response function

- \( R_N \) = NOZZLE RESPONSE FUNCTION
- \( I \) = ACOUSTIC STABILITY INTEGRALS
- \( \alpha_{FT} \) = FLOW TURNING COEFFICIENT
- \( \alpha_{PD} \) = PARTICLE DAMPING COEFFICIENT
- \( \alpha_{ST} \) = STRUCTURAL DAMPING COEFFICIENT
ACOUSTIC ADMITTANCE

ACOUSTIC ADMITTANCE OF PROPELLANT BURNING SURFACE

\[ A_{PC} = \frac{u' / \bar{a}}{P' / \gamma \bar{P}} \quad A_{VC} = \frac{u' / \bar{a}}{v' / \bar{a}} \]

RESPONSE FUNCTION AND ACOUSTIC ADMITTANCE RELATED

BY BURNING SURFACE MASS BALANCE

\[ A_b + \bar{M}_b = \gamma \bar{M}_b R_b \]

\( \bar{M}_b = \text{mean Mach No. of propellant surface gas evolution} \)

Relationship between combustion response function and acoustic admittance

03/30/88
DESCRIPTION OF
MICROWAVE BURNING RATE
MEASUREMENT TECHNIQUE

INCOMING MICROWAVE
AT FREQUENCY $f_1$

REFLECTED MICROWAVE
AT FREQUENCY $f_2$
WITH PHASE SHIFT $\phi$

UNBURNED PROPELLANT

BURN RATE, $r$

$\frac{dr}{dt} \propto \frac{d\phi}{dt}$

Rate of regression of propellant surface proportional to time rate of change of the Doppler phase shift.
COMBUSTION RESPONSE FUNCTION

\[ R_b = \frac{m'/\bar{m}}{P'/\bar{P}} = \frac{\rho r'/\bar{\rho}}{P'/\bar{P}} = \frac{r'/\bar{r}}{P'/\bar{P}} \]  

MICROWAVE DOPPLER SHIFT MEASUREMENT

\[ r = \frac{1}{2 k_p} \frac{d\phi}{dt} \]  

FROM EQN. (2):

\[ \bar{r} = \frac{1}{2 k_p} \frac{\dot{\phi}}{\dot{\phi}} \]  

AND FOR A SINEOIDALLY VARYING P', (P' = \Delta P e^{i\omega t}):  

\[ \phi' = \Delta \phi e^{i(\omega t + \theta)} \]  

AND \[ r' = \frac{d\phi'}{dt} / 2 k_p \]

\[ = e^{i\omega t + \theta} / 2 k_p \]

\[ = 2\pi f \Delta \phi e^{i(2\pi f + \theta + \pi/2)} / 2 k_p \]
COMBINING \( r' \) AND \( r \) RESULTS WITH \( \text{MEASURED PRESSURE INFORMATION} \) \((P, P')\), FROM EQN. (1):

\[
b = \frac{r'/r}{P'/P} = \frac{2\pi f \Delta \phi}{\phi} \frac{\overline{P}}{\Delta P} e^{i(\theta + \pi/2)}
\]  

(6)

SETTING \( \phi = 360^\circ / \Delta t \), WHERE \( \Delta t \) IS THE PERIOD FOR A 360° SHIFT IN PHASE:

\[
R_b = \frac{2\pi f \phi \Delta t}{360^\circ} \frac{\overline{P}}{\Delta P} e^{i(\theta + \pi/2)}
\]

(7)

\[ |R_b| e^{i(\theta + \pi/2)} \]
CALCULATED MINIMUM MEASURABLE ABSOLUTE VALUE OF RESPONSE FUNCTION vs. FREQUENCY

- $\bar{v} = 1.3 \text{ cm/s}$
- $\lambda_p = 1.5 \text{ cm}$
- $\Delta \phi_{\text{min}} = 0.01^\circ$
- $\Delta P/P = 0.02$

Minimum measurable response function limited by oscillatory phase measurement resolution, $\Delta \phi_{\text{min}}$ (which is fixed) and pressure modulation amplitude, $\Delta P/P$. 

Measurement region

Data buried in noise

Signal-to-noise ratio = unity
OBJECTIVE:

- Upper frequency limit of microwave Doppler velocimeter technique for measuring pressure-coupled response function \( R_p \) of solid propellants is \( \sim 1 \) kHz, limited primarily by limiting amplitude of driven pressure oscillations in microwave burner. Objective is to develop revised system capable of making measurements at frequencies up to 2 kHz and more.

APPROACH:

- Use microwave Doppler velocimeter to measure combustion response to oscillating thermal radiation source (laser) rather than pressure. Relate measured burning rate response to thermal radiation \( R_q \) to equivalent oscillation in pressure using existing thermal combustion theory.

\[
R_q \left[ \frac{q'/q_0}{p'/p_0} \right]
\]

p = Theoretical transfer function between oscillatory pressure and propellant heating flux.
DEFINITION OF
OSCILLATORY HEAT FLUX RESPONSE FUNCTION

\[ R_q = \frac{r''/\bar{r}}{q'/\bar{q}} = \frac{r'/\bar{r}}{q'/\bar{q}} \quad \dot{r} = f(\dot{\phi}'/dt) \]
\[ \bar{r} = f(\bar{\phi}/dt) \]

FOR A SINUSOIDALLY VARYING \( q' \)

\[ R_q = \left[ \frac{c\phi \Delta \phi}{\Delta q,} \right] e^{i(\theta + \pi/2)} \]

\[ = \left[ \frac{2\pi f \Delta \phi_{\text{mean}}}{\Delta q_{\text{mean}}} \right] \frac{\dot{\phi}_{\text{CW}} - \dot{\phi}_0}{\Delta q_{\text{CW}}} e^{i(\theta + \pi/2)} \]

Phase angle between \( \phi' \) and \( q' \)

Argument of response

Absolute value of response

Continuous wave
CALCULATED TRANSFER FUNCTION VS FREQUENCY FOR TYPICAL AP COMPOSITE PROPELLANT

\[
TF = \left[ \frac{\rho_p}{\rho_p'} \right]
\]

- \( P, \text{ atm.} \)
  - -- 10
  - --- 20
  - -- 30

![Graph showing transfer function vs frequency for typical AP composite propellant](image-url)
A SECOND TRANSFER FUNCTION ANALYSIS

\[ \frac{q}{p} = \frac{q}{p} = \frac{dq}{d\bar{p}} \]

THERMAL ANALYSIS OF SOLID PHASE YIELDED

\[ \frac{dq}{d\bar{p}} = \alpha \] the propellant burning rate
pressure exponent
SCHEMATIC OF TEST CONCEPT

using microwave Doppler velocimeter technique to measure the propellant combustion response to an incident, oscillating thermal radiation source (CO₂ laser).
MICROWAVE WINDOW BURNER ASSEMBLY

- Dielectric Transition Holder
- Closure Support
- Propellant Sample Holder
- Burner Support
- Pressure Transducer Port
- IR Window
- Window Retainer
- Burner
- Pressurization Inlet
- Pressurization Outlet
- Base Plate
- Waveguide
- Lock Nut

Waveguide

16 GHz

laser heat flux
Microwave window burner and waveguide system
If in-depth transmission of the IR radiation (low absorption coefficient) were a factor, it would have to be included in the interpretive combustion model.

PROPELLANT IR
RADIATION ABSORPTION
CHARACTERISTICS

LOW ABSORPTION COEFFICIENT

INCIDENT IR RADIATION

HIGH ABSORPTION COEFFICIENT
JPL

TEST PROPPELLANT FORMULATIONS

<table>
<thead>
<tr>
<th></th>
<th>A-13 (control)</th>
<th>A-13, MOD. 1</th>
</tr>
</thead>
<tbody>
<tr>
<td>AP (90 μm)</td>
<td>76.0%</td>
<td>75.75%</td>
</tr>
<tr>
<td>CARBON</td>
<td>0</td>
<td>0.25 (Added to increase IR radiation absorption coefficient)</td>
</tr>
<tr>
<td>PB&lt;sup&gt;2&lt;/sup&gt;N-787</td>
<td>20.4</td>
<td>20.4</td>
</tr>
<tr>
<td>EPON RESIN 828</td>
<td>3.6</td>
<td>3.6</td>
</tr>
</tbody>
</table>

A-13 is a greater than 25 years old Navy propellant formulation that served as a JANNAF round robin test propellant and whose combustion response has been well characterized.
LASER MEAN HEAT FLUX VS FREQUENCY, A-13 TESTS

Measured variation in laser mean heat flux with frequency for the sinusoidally modulated CO₂ laser.
LASER MEAN HEAT
LUX vs FREQUENCY,
A-13, MOD. 1 TESTS

Results for second propellant test series
The reduction in $\Delta \phi$ amplitude with time is attributed to the attenuation of the radiant flux by the growing column of combustion gases in the propellant tube. The zero burn time intercept was taken as the non-attenuated value.
JPL OSCILLATORY PHASE SHIFT VS TIME

TEST 210-5 A-13, MOD. 1 PROPELLANT
f = 5 Hz, P = 300 Psia
measured mean Doppler phase shift at a constant (continuous wave) heat flux. The zero time intercept was taken as the non-attenuated value.

RATE OF CHANGE OF DOPPLER PHASE SHIFT vs TIME, CONTINUOUS WAVE HEATING TEST

A-13, MOD. 1 PROPELLANT

A $q_{cw} = 30.2$ Cal/cm²·s

2.1 MPa (300 Psia) NOMINAL PRESSURE
RATE OF CHANGE OF DOPPLER PHASE SHIFT vs LASER CONTINUOUS WAVE HEAT FLUX, A-13, MOD. 1 PROPellant

Determination of \( \frac{\dot{\Phi}_{cw} - \dot{\Phi}_0}{\Delta \Phi_{cw}} \) slope constant

Slope = 0.26

2.1 MPa (300 Psia) NOMINAL PRESSURE
ABSOLUTE VALUE OF OSCILLATORY RESPONSE vs FREQUENCY, A-13 PROPELLANT

$|R_q|$ vs $f$, Hz

2.1 MPa (300 Psia) NOMINAL PRESSURE

Laser modulation frequency
ABSOLUTE VALUE OF OSCILLATORY RESPONSE vs FREQUENCY, A-13, MOD. I PROPELLANT

2.1 MPa (300 Psia) NOMINAL PRESSURE
ARGUMENT OF OSCILLATORY RESPONSE vs FREQUENCY, A-13 PROPELLANT

2.1 MPa (300 Psia) NOMINAL PRESSURE

- MEDIAN OF REDUCED DATA POINTS FOR INDIVIDUAL TESTS
- Range of reduced data points

Data scatter believed due to heterogeneous nature of the test propellant.
ARGUMENT OF OSCILLATORY RESPONSE vs FREQUENCY, A-13, MOD. 1 PROPELLANT

2.1 MPa (300 Psia) NOMINAL PRESSURE

O MEDIAN OF REDUCED DATA POINTS FOR INDIVIDUAL TESTS
Product of curve fit to the $|R_q|$ data and the cosine of the curve fit to the argument data yields the real (in-phase) component...
Real component of the pressure-coupled response function, obtained by multiplying the $R_q(r)$ results by their respective transfer function, $n$. Differences for two test propellants concluded to be within experimental margin of error. In-depth heat transfer not tested.
REAL (IN-PHASE) COMPONENT OF PRESSURE-COUPLED RESPONSE FUNCTION vs FREQUENCY, A-13 PROPellant

Pressure-coupled responses measured for two test pressures, showing increasing response with decreasing pressure - consistent with literature.
CONCLUSIONS

- MEASURED RESPONSE FUNCTION RESULTS FOR A-13 PROPELLANT FORMULATION (MAGNITUDE AND FREQUENCY RANGE OF MAXIMUM RESPONSE) COMPARABLE WITH EXISTING $R_p(r)$ RESULTS OBTAINED BY A NUMBER OF DIFFERENT TECHNIQUES

- ADDITION OF QUARTER PERCENT CARBON HAD NO APPRECIABLE EFFECT ON MEASURED COMBUSTION RESPONSE CHARACTERISTICS - SUPPORTS CONCLUSION THAT IN-DEPTH TRANSMISSION OF RADIANT HEAT FLUX NOT A SIGNIFICANT FACTOR AT CO$_2$-LASER WAVELENGTH

- TESTS NEED TO BE PERFORMED ON HOMOGENEOUS PROPELLANTS TO DETERMINE VALIDITY OF POSTULATED REASON FOR LARGE TIME-VARIATION IN RESPONSE FUNCTION ARGUMENT DATA

- TECHNIQUE SHOWS PROMISE AS A TOOL FOR CHARACTERIZING PROPELLANT COMBUSTION STABILITY
Results of test series on high burning rate (≈ 2 in./s at 2000 psia) igniter propellant

300 Psia Nominal Pressure

Normalized to unity, per theory

Figure 9 Absolute value of heat flux response function vs. frequency for ASRM Igniter Propellant
For this propellant the dynamic burning rate, \( r' \), leads the oscillatory heat flux driver, \( q' \), over the frequency range of 300 to 1500 Hz.

Figure 10: Argument of response function vs frequency for ASRM Igniter propellant
Figure II. Absolute value of pressure-coupled response function vs. frequency for ASRM Igniter propellant determined by multiplying the curve fit to the $|R_f|$ data by the propellant's burning rate pressure exponent, $n$.  

**300 Psia Nominal Pressure**
Normalizing the test results by plotting versus frequency over burning rate squared in an attempt to apply the results to pressures other than the 300 psia test pressure.

*Figure 12.* Absolute value of pressure-coupled response function vs. normalized frequency for ASRM Igniter Propellant.
Real component obtained by taking the product of the $|R_p|$ curve and the cosine of the curve fit to the argument data, $\Theta + T/2$. The argument results produce a characteristic bi-modal response, with peaks at frequencies of 400 and approximately 1250 Hz (at this test pressure).

Figure 13. Real component of pressure-coupled response function vs. frequency for ASRM Igniter propellant.
Figure 14. Real component of pressure-coupled response function vs frequency for ASRM Igniter Propellant

For any operating pressure, the frequencies of peak response can be estimated by multiplying the respective $f/r^2$ value by the square of the burning rate at that pressure.
PLASMA CAPACITANCE GAGE

PROPellant BURNING RATE MEASUREMENT

TECHNIQUE
EROSIVE BURNING

- The augmentation in burning rate of solid propellants produced under conditions of high-speed flow parallel to propellant burning surface.

- Predicted to be enhanced by factors that increase level of turbulence close to propellant burning surface.
  - High cross-flow velocity.
  - Low surface blowing rate (low propellant burning rate, high chamber pressure).
  - Propellant surface roughness.
JPL EROSION BURNING RESEARCH

PURPOSE

- To measure effects of parameters that are considered to most strongly influence scaling to larger rocket motor sizes of transition or threshold conditions for erosive burning rate augmentation
- Develop a scaling criterion for threshold conditions for erosive burning

APPROACH

- Carry out series of large-scale motor firings systematically varying (1) motor cross-flow velocity or Mach No., (2) chamber pressure, (3) propellant non-erosive burning rate, (4) propellant surface roughness, and (5) motor port diameter, and measuring motor transition length to erosive burning rate augmentation
- Incorporate erosive burning dependencies on test parameters into scaling criterion

Measured erosive augmentation of propellant base burning rate \( r_0 \) using plasma capacitance gage technique.

\[ \frac{r}{r_0} \text{ propellant base burning rate without crossflow of combustion gases} \]
Each PCG consists of a capacitor formed between an electrode located at the outer surface of the propellant grain and the ionized flame zone (as ground), with the propellant web as the capacitor dielectric. The changing capacitance with regression of the burning surface, input into an analytical model of the device, yields a continuous measurement with time of the local propellant web thickness and consequently burning rate.
Each gage circuit is a resistor to capacitor AC voltage divider with two rectifier diodes and a filter capacitor.
AIAX 89-2528

JPL BATES EROSIVE BURNING TEST MOTOR
12 inch (135TES) rocket motor segment instrumented with plasma capacitance gage electrode
SEGMENTED 5 × 1° IN. BATCH-CHECK CHAMBERS

41"
Schematic of 5x10 rocket motor segment instrumented with plasma capacitance gage electrode.
CHAMBER PRESSURE vs TIME
TEST NO. 10, 5 x 10 MOTOR

AIAA 89-2578

Plot of the mean pressure at axial measurement stations along the rocket motor for a typical 5x10 segmented motor test, showing the higher than predicted initial pressure caused by the excessive burning-rate augmentation and an axial pressure gradient down the motor port that diminishes with time (increasing port diameter).
Example of measured burning rate results, normalized by the strand burner, no-crossflow rates at the corresponding pressures, for four segment Sx10 motor test.
Example of measured burning rate results, normalized by the re-cosflow rates, for a five segment 5x10 motor test.
BURNING RATE AUGMENTATION vs TIME
TEST NO. E-1923

Example of measured burning rate results, normalized by the non-crossflow rates, for a four segment 5x10 motor test.
Correlation of threshold conditions for erosive burning rate augmentation. The critical crossflow mass flux (crossflow Re. No.) increases with propellant base burning rate (surface-transpiration Re. No.) and the motor length-to-diameter ratio.
Correlating the magnitude of erosive burning rate augmentation by the amount that the Reynolds No. driving parameter exceed an erosive threshold value. A distinct size scaling effect is apparent for the two different sized rocket motors.

**BURNING RATE AUGMENTATION VS RE NO. - RE NO.,th**

**PBAN PROPELLANT FORMULATIONS**

**SEGMENTED BATES MOTOR TESTS**

**SEGMENTED 5X10 MOTOR TESTS**
The size scaling effect was accounted for by adding the motor port radius to the -2.4 power to the independent parameter. The crossflow Reynolds No. was also raised to the 0.8 power, as suggested by correlations for convective heat transfer in pipe flow.

**Burning Rate Augmentation vs**

\((\text{RE NO.}^{(0.8)} - \text{RE NO.},\text{th}^{(0.8)})/\text{R}^{(2.4)}\)

**PBAN Propellant Formulations**

- SEGMENTED BATES MOTOR TESTS
- SEGMENTED 5X1 O MOTOR TESTS
BURNING RATE AUGMENTATION vs

\( (G^{*0.8} - G_{th}^{*0.8})/R^{*0.4} \)

PBAN PROPPELLANT FORMULATION'S

- - - - - SEGMENTED BATES MOTOR TESTS

△ △ △ △ SEGMENTED 5X10 MOTOR TESTS

Erosive burning rate augmentation with mass flux, \( G \), as the driving parameter. The independent parameter \( (G^{*0.8} - G_{th}^{*0.8})/R^{*0.4} \) provided the required size scaling.
HYBRID ROCKET

COMBUSTION INSTABILITY

MECHANISMS
HYBRID SLAB WINDOW MOTOR

Laboratory hybrid window motor used for fuel decomposition and combustion studies.
<table>
<thead>
<tr>
<th>Parameter</th>
<th>Instrument</th>
</tr>
</thead>
<tbody>
<tr>
<td>CORE GAS TEMPERATURE</td>
<td>MIKRON SERIES M77S 2-COLOR INFRARED THERMOMETER</td>
</tr>
<tr>
<td>FUEL SURFACE TEMPERATURE</td>
<td>MIKRON SERIES M67S INFRARED THERMOMETER (3.86 μm)</td>
</tr>
<tr>
<td>TOTAL HEAT FLUX</td>
<td>HYCAL ASYMPTOTIC CALORIMETER</td>
</tr>
<tr>
<td>RADIATION HEAT FLUX</td>
<td>HYCAL ASYMPTOTIC RADIOMETER</td>
</tr>
</tbody>
</table>
Instrumentation configurations used for the tests.
Comparison of regression rate and total heat flux variations vs. total mass flux

Good agreement in the respective correlations of fuel regression rate and total heat flux through the motor port, indicating a direct dependency of fuel regression on total heat flux to fuel burning surface.
Simultaneous d.c. shifts in chamber pressure and total heat flux, while rate of oxygen injection remained constant, supporting postulation that the driving mechanism for the sub-acoustic pressure irregularities is some type of flow-combustion turbulence interaction along the fuel surface.

HEAT FLUX & PRESSURE vs TIME

TEST NO. E2091      FUEL: HTPB

CHAMBER PRESSURE

TOTAL HEAT FLUX

Frequency of oscillations ~ 30 Hz, well below acoustic range for this motor.
Instability accompanied by d.c. shifts in gas (positive) and fuel surface (negative) temperatures

TEMPERATURE vs TIME

TEST NO. E2091      FUEL: HTPB

TEMPERATURE, °F

FUEL SURFACE TEMPERATURE

CORE GAS TEMPERATURE
MINIATURIZED COMPONENTS

FOR FUTURE SPACECRAFT

PROPULSION SYSTEMS
Preliminary design of a miniaurized spacecraft for a mission to the planet Pluto. Designed to be a pathfinder for future, lower cost deep space exploration missions.
JPL PLUTO FAST FLYBY - SPACECRAFT PROPULSION

**LEGEND**
- Latching Isolation Valve
- Fill Valve
- Pressure Regulator
- Filter
- Orifice
- Pressure Transducer
- Temperature Transducer

Schematic of candidate monopropellant hydrazine (course correction) and cold-gas (attitude and roll control) propulsion system.
Test system was used for testing prototype cold-gas propulsion components at simulated space flight temperature conditions.