DYNAMIC STABILITY TEST RESULTS

ON AN 0.024 SCALE B-1 AIR VEHICLE
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DATE 17 March 1972.
NO. OF PAGES 145

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FOREWORD

This report was prepared by the Aerodynamics Group of the Los Angeles Division of North American Rockwell Corporation at Los Angeles, California under Air Force contract No. F33657-70-C-0800, and the NASA Vehicle Dynamics Section of Langley Research Center, Langley Air Force Base, Virginia.

Three Langley Research Center wind tunnels were utilized to obtain the main damping derivatives ($C_{mq}$, $C_{nr}$ and $C_{dp}$) over the Mach number range of the air vehicle: the Langley 4 foot Unitary Plan wind tunnel (supersonic), the Langley 8 foot Transonic Pressure tunnel and the Langley 7 x 10 foot High Speed tunnel (transonic and subsonic).

Messrs. R. R. Becman and R. Whitmoyer (NR Aerodynamics) and Mr. Cecil Berthold (NR Wind Tunnel) were assigned to this test under the overall technical direction of Mr. L. Gaines (NR Aerodynamics).

NASA, LRC, Vehicle Dynamics Section personnel conducting the test were Messrs. R. A. Kilgore, E. E. Davenport, J. Adcock and R. Boyden under the supervision of Mr. H. G. Wiley.
Dynamic longitudinal and lateral-directional stability characteristics of the B-1 air vehicle have been investigated in three wind tunnels at the Langley Research Center. The main rotary derivatives were obtained for an angle of attack range of -3 degrees to +16 degrees for a Mach number range of 0.2 to 2.16. Damping in roll data could not be obtained at the supersonic Mach numbers. The Langley 7 x 10 foot high speed tunnel, the 8 foot transonic pressure tunnel and the 4 foot Unitary Plan wind tunnel were the test sites. An 0.024 scale light-weight model was used on a forced oscillation type balance. Test Reynolds number varied from 0.474 x 10^6/Ft. to 1.55 x 10^6/Ft. through the Mach number range tested.

The results of the investigation showed that the dynamic stability characteristics of the model in pitch and roll were generally satisfactory up to an angle attack of about +6 degrees. In the wing sweep range from 15 to 25 degrees the positive damping levels in roll deteriorated rapidly above +2 degrees angle of attack. This reduction in roll damping is believed to be due to the onset of separation over the wing as stall is approached.

In the subsonic and transonic speed range, yaw damping levels are negative in some cases (wing A = 25°, 55° and 65°), vertical tail inputs are zero to negative and body input is about twice the estimate. The afterbody of the rotary derivative model is distorted from the air vehicle lines to permit insertion of the balance and to provide clearance between the base and the sting for model oscillation.

The resulting large diameter base is believed to have caused an aerodynamic interference between the model afterbody and the horizontal and vertical tails which significantly altered the vertical tail input to yaw damping. Since the B-1 fuselage has a much different shape than the model in the vicinity of the vertical tail, the yaw damping characteristics measured on the model are probably not representative of the B-1 air vehicle.
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**ROLL DAMPING**

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NOMENCLATURE

Symbols

- $\ell$ reference length = 10.33 ft.

- $C_l$ rolling moment coefficient, $\frac{\text{Rolling moment}}{q_\infty S\ell}$

$C_{lp} = \frac{\partial C_l}{\partial \left(\frac{p_\infty}{2V}\right)}$ per radian

- $C_m$ pitching moment coefficient, $\frac{\text{Pitching moment}}{q_\infty S\ell}$

- $k$ reduced frequency parameter, $\frac{\omega \ell}{2V}$ in pitch and yaw

- $M$ freestream Mach number

- $q$ angular velocity of model about body $Y$-axis, rad/sec

- $q_\infty$ freestream dynamic pressure, psf

- $r$ angular velocity of model about body $Z$-axis, rad/sec

- $S$ reference area = 60.44 ft.$^2$

- $V$ freestream velocity ft/sec

- $\alpha$ angle of attack, degrees or radians; mean angle of attack, deg.

- $\beta$ angle of sideslip, degrees or radians; mean angle of sideslip, degrees

- $\omega$ angular velocity, $2\pi f$, rad/sec

$C_{mq} = \frac{\partial C_m}{\partial \left(\frac{q_\infty}{2V}\right)}$ per rad

$C_{mq} + C_{m\alpha}$ damping in pitch parameter, per radian
\[ C_{m_1} = \frac{\partial C_m}{\partial \delta l^2} \text{ rad} \]

\[ C_{m_\alpha} = \frac{\partial C_m}{\partial \alpha} \text{ rad} \]

\[ C_{m_\alpha} = k^2 C_{m_1} \text{ oscillatory-longitudinal-stability parameter, per radian} \]

\[ C_{m_\dot{\alpha}} = \frac{\partial C_m}{\partial \dot{\delta l}} \text{ rad} \]

\[ C_n = \text{ yawing-moment coefficient, } \frac{\text{ Yawing moment}}{q_\infty S} \]

\[ C_{n_r} = \frac{\partial C_n}{\partial \frac{r}{V}} \text{ rad} \]

\[ C_{n_r} = C_{n_\dot{\beta}} \cos \alpha \text{ damping-in-yaw parameter, per radian} \]

\[ C_{n_\dot{\beta}} = \frac{\partial C_n}{\partial \dot{\delta l}} \text{ rad} \]

\[ C_{n_\beta} = \frac{\partial C_n}{\partial \beta} \text{ rad} \]

\[ C_{n_\beta} \cos \alpha = k^2 C_{m_1} \text{ oscillatory-directional-stability parameter, per radian} \]

\[ C_{n_\dot{\beta}} = \frac{\partial C_n}{\partial \dot{\delta l}} \text{ rad} \]

\[ C_{\dot{l_\beta}} = \frac{\partial C_l}{\partial \frac{\delta l}{V}} \text{ rad} \]

\[ C_{\dot{l_\beta}} = \frac{\partial C_l}{\partial \dot{\delta l}} \text{ sin } \alpha \text{ damping in roll parameter, per radian} \]
INTRODUCTION

Design of guidance and control systems for aerodynamic configurations requires knowledge of the various dynamic stability derivatives through wide ranges of flight speeds. Experimental data and theories are available at low speeds, and a limited amount of data is available at supersonic speeds. However, at transonic speeds no adequate theories for predicting these derivatives are available, especially at angle of attack, and little experimental data exists.

Therefore, wind tunnel test investigations have been conducted by the National Aeronautics and Space Administration and North American Rockwell Corporation to substantiate the levels of the main dynamic stability derivatives of the B-1 air vehicle. The information obtained will be used to determine their effect on the flight dynamics of the air vehicle.
DISCUSSION

MODEL DESCRIPTION

The model is a relatively lightweight 0.024-scale replica of the B-1 air vehicle with provision for sting mounting to the NASA-Langley rigidly forced oscillation system. Aerodynamic similarity is maintained except for the fuselage modifications necessary to accommodate the support sting and oscillating mechanism, figures 1 through 8. The upper fuselage moldline follows the air vehicle lines, reference 1, but a significant deformation of the lower surface was necessary to form the required circular cross section aft of fuselage station 950. Forward of this station, the circular section gradually blends into the air vehicle mold lines at fuselage station 150.

The wing and empennage surfaces are fabricated of aluminum alloy and the forward fuselage section is constructed of laminated fiberglass. The wing is pivoted to provide four discrete sweep positions and the horizontal tail is adjustable to $-10^\circ$, $0^\circ$, and $+10^\circ$ deflection.

The balance mounting bulkhead can be moved in the mounting tube to place the balance center-of-rotation at the longitudinal position corresponding to the full scale air vehicle center of gravity for the various wing sweep positions.

The reference dimensions used in the data calculation are the following:

<table>
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<th>0.24-Scale</th>
<th>Full Scale</th>
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<tr>
<td>$S_W = 1.1209$ sq. ft.</td>
<td>1946 sq. ft.</td>
</tr>
<tr>
<td>$b_W = 39.364$ in.</td>
<td>1640.17 in.</td>
</tr>
<tr>
<td>$c_W = 4.417$ in.</td>
<td>184.05 in.</td>
</tr>
<tr>
<td>F.S. of $.25c_w = 23.708$ in.</td>
<td>987.85 in.</td>
</tr>
<tr>
<td>F.S. of $.45c_w = 24.59$ in.</td>
<td>1024.66 in.</td>
</tr>
<tr>
<td>F.S. of $.75c_w = 25.92$ in.</td>
<td>1079.88 in.</td>
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Additional model and test information is contained in references 2 and 3.

WIND TUNNELS

Three wind tunnels were used to obtain the data presented herein. All three are equipped for control of relative humidity and total temperature of the air in the tunnel to minimize the effects of condensation and for control of total pressure in order to obtain the test Reynolds number.
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Langley Unitary Plan Wind Tunnel.—The data at Mach numbers 1.8 and 2.16 were obtained in test section number 1 of the Langley Unitary Plan wind tunnel. This single return tunnel has a test section about 4.0 feet by 4 feet. An asymmetric sliding block, which varies the area ratio, is used to change Mach number from about 1.47 to 2.86. The angle of attack mechanism used for this investigation has a total range of about 25 degrees when used with the oscillation balance mechanism.

Langley 8-foot Transonic Pressure Tunnel.—The data at Mach numbers from 0.40 to 1.20 were obtained in the Langley 8-foot transonic pressure tunnel. The test section of this single return closed-circuit wind tunnel is about 7.0 feet by 7 feet with slotted upper and lower walls to permit continuous operation through the transonic speed range. Test section Mach numbers from about 0.2 to 1.30 can be obtained and kept constant by controlling the speed of the tunnel fan drive motor. The Mach number is reasonably uniform throughout the test section, with a maximum deviation from the average freestream Mach number of approximately 0.01 at the higher Mach numbers.

The sting support strut is designed to keep the model near the center line of the tunnel through a range of sting angles of attack from about -4 to +22 degrees when used with the oscillation balance mechanism.

Langley 7 x 10 foot High Speed Tunnel.—Correlation data for Mach numbers from 0.2 to 0.8 were obtained in the Langley 7 by 10 foot high speed tunnel. It is a single return wind tunnel which operates with atmospheric stagnation pressure and has a conventional, closed, rectangular fixed geometry test section. The sting support strut is designed to keep the model near the center line of the tunnel through a range of sting angles of attack from about -12 to +16 degrees when used with the oscillation balance mechanism.

Oscillation Balance Mechanism

Two oscillation balance mechanisms were used for these tests, one for pitch and yaw and the other for roll. The damping terms are measured as a function of how much torque is required to oscillate the models at a certain amplitude. A description of these mechanisms is given in reference 3.

Axis System

The aerodynamic parameters are referred to the body system of axes originating at the oscillation center of the model as shown in figure 7. The reference dimensions are based on the geometric characteristics of the model with the wings in the swept forward position (A = 15°) regardless of the actual test wing-sweep position.
PRESENTATION OF RESULTS

The actual run schedules for the five tests presented in this report are shown on Tables I through V (pages 20 through 25). The tests were as follows:

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<th>Derivative</th>
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<td>952</td>
<td>4' UPWT</td>
<td>Pitch &amp; Yaw</td>
</tr>
<tr>
<td>II &amp; III</td>
<td>596</td>
<td>8' TPT</td>
<td>Pitch &amp; Yaw</td>
</tr>
<tr>
<td>IV</td>
<td>599</td>
<td>8' TPT</td>
<td>Roll</td>
</tr>
<tr>
<td>V</td>
<td>934</td>
<td>7 x 10' HST</td>
<td>Yaw &amp; Pitch</td>
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Test Reynolds number, based on the wing mean aerodynamic chord was approximately 1,100,000 in the Unitary Plan tunnel, 1,550,000 in the 8' TPT, and between 1,100,000 and 1,450,00 in the 7 x 10 tunnel with the exception of the data taken at Mach 0.2, which was at a Reynolds number of 500,000.

The damping in pitch derivative $C_{m_\alpha} + C_{m_\alpha}$ and the oscillatory-stability parameter $C_{m_\alpha} - k^2 C_{mq}$ are presented in figures 9 through 53 for subsonic, transonic and supersonic Mach numbers. The data obtained show positive damping and are in fair agreement with estimated levels at zero angle of attack, and in general show an increase in damping with increase in angle of attack. Component build-up data, such as figures 13 thru 19, show that the horizontal tail contributes approximately 50 percent of the damping in pitch. Plots showing the effect of wing leading edge sweep, Mach number, center of gravity shift and wing input on the damping in pitch parameters are also presented. The effects of Mach number at constant angles of attack are presented on figures 9 through 12 for various leading edge sweep angles. Estimated data are superimposed on these plots.

The damping in yaw derivative $C_{n_\beta} - C_{n_\beta}$ and the oscillatory stability parameter $C_{n_\beta} \cos + k^2 C_{n_\beta}$ are presented on figures 54 through 102 for subsonic, transonic and supersonic Mach numbers. Inspection of the supersonic data reveal that $C_{n_\beta} - C_{n_\beta}$ measured approximately 50 percent more damping than was estimated. The increased damping was due primarily to body input, which was probably influenced by the enlarged model base. Vertical tail input to the damping in yaw parameter was about 70 percent of the estimated data. This reduction in tail damping may be due in part to the distorted model afterbody, as suspected in the subsonic tests. Directional stability, $C_{n_\beta}$ due to the tail was slightly less than that measured in the trisonic wind tunnel.
The subsonic damping in yaw data measured in the Langley 8' TPT was questionable because of variation with frequency, and zero or negative damping due to vertical tail. The model was shifted to the Langley 7 x 10 HST for further investigation with a stiffer model support system. The variation of $C_{\eta_\alpha} - C_{\eta_\beta}$ with frequency was reduced but the vertical tail input to yaw damping was still zero or negative. Removing the horizontal tail improved the vertical tail damping input. A fairing was tested on the vertical tail in the area between the horizontal tail and aft fuselage mold line, page 32, to partially simulate the three dimensional relief between the horizontal tail and fuselage as it exists on the air vehicle. Data obtained with this "channel filler block" brought the tail on yaw damping level closer to estimated values, fig 68 & 69. Tail input, however, was still less than 50 percent of estimated. The data also show that damping in yaw gets less positive with increase in Mach number from 0.2 to 0.8 for all sweeps. Installation of the "filler blocks" reversed this trend through this Mach number range. Since either the removal or deflection of the horizontal tail, or the addition of the filler blocks on the vertical tail drastically altered the vertical tail input to yaw damping, it is apparent that aerodynamic interference between the fuselage afterbody, horizontal and vertical tails significantly affected the yaw damping of the model configuration. Since the model lines are significantly different than the B-1 air vehicle in this area, the yaw damping characteristics measured on the model are probably not representative of the B-1 air vehicle.

The damping in roll derivative $C_{\eta_p} + C_{\eta_\beta} \sin \alpha$ is presented on figures 103 through 119. These data were obtained at subsonic through transonic Mach numbers for the wing leading sweep range of the air vehicle. Damping data obtained with the wing in the aft sweep positions, $\Lambda = 65$ and 55 degrees show good agreement with estimated data at a fairly constant level of damping with variation of angle of attack. Data obtained with the forward leading edge sweep positions $\Lambda_{FG} = 25$ and 15 degrees show good agreement with estimated data from -3 to 0 to 2 degrees angle of attack. The damping levels drop rapidly with increasing angle of attack above 0 to 2 degrees. This phenomenon is associated with the onset of airflow separation over the wing as the angle of attack for stall is approached. Force data obtained on other B-1 models indicate the onset of separation at somewhat higher angles of attack than those at which the roll damping begins to deteriorate. This is probably because the force models were tested at higher Reynolds numbers than the rotary derivative model.
CONCLUSIONS

Data obtained with the wing panels swept 15 and 25 degrees showed positive damping in pitch, that generally remained linear with angle of attack for angles below about six degrees. The damping in roll data obtained showed positive damping at angles of attack from -2 to +2 degrees; above these angles of attack the damping deteriorated rapidly. This phenomenon is closely associated with the onset of separation at these sweeps and angles of attack.

Data obtained with the wing panels swept 55 and 65 degrees showed positive damping in pitch and roll and remained linear with angle of attack for angles below about six degrees at subsonic and transonic Mach numbers and about 10 degrees supersonically. No supersonic damping in roll data was obtained.

The aerodynamic damping in pitch and roll measured in these tests appear reasonable, and are substantiated by estimates for the B-1 configuration.

The measured aerodynamic damping in yaw showed some unusual results. The damping due to the body was larger than estimated, but the measured damping due to the vertical tail was less than estimated, and sometimes even negative at Mach numbers between .7 and .8. Component buildup showed the vertical tail input to be strongly affected by aerodynamic interference between the fuselage afterbody and the horizontal and vertical tails. Since the model shape is significantly different than the B-1 air vehicle in this area, the yaw damping characteristics measured on the model are probably not representative of the B-1 air vehicle.

RECOMMENDATIONS

In order to obtain data that can be used during design to define the flight dynamics of the B-1 air vehicle, additional testing must be accomplished. The model lines should be modified to more closely approximate B-1 afterbody lines so that the questionable validity of the initial test data can be clarified. Since the horizontal tail planform was changed subsequent to the construction of this model, this component should also be updated. Also since roll damping was not attainable above $M = 1.2$, additional testing for this parameter should be accomplished over the complete Mach range.
DIMENSIONAL DATA

Wing

Model

Reference Area, Ft.² 1.120
Span, Ft. 3.272
Aspect Ratio 9.560
Taper Ratio .351
Chords, In.:
  Root (B.P. 0.0) 6.075
  Tip (B.P. 19.634) 2.135
  M.A.C. (B.P. 8.247) 4.420
Fus. Sta. of 0.25 M.A.C., In. 23.705
Fus. Sta. of Wing Pivot (B.P. 3.480), In. 23.256

Leading Edge Sweepback Angle, Deg. 14.992°
Dihedral, Deg. -1.940°
Incidence, Deg. 0.0°

Wing hood

Data for One of Two Sides

Model

Area, Ft² .065
Corresponding Wing Area, ft² 1.120
Wing Sweep, Deg. 15.

Body

Model

Length, Ft. 3.40
Max. Width (F.S. 22.8-40.8), In. 3.6
Max. Depth (F.S. 22.8-40.8), In. 3.6
Max. Cross-Sectional Area (F.S. 22.8-40.8), Ft² .071
Fineness Ratio 11.333

B60 B51 with Ogive Forebody

B61 B51 with Filler Blocks in Channel Between Horizontal Tail and Fuselage (see page 32)

B62 B51 with Vertical Tail Mounting Plate Faired Forward to Blend More Smoothly into Fuselage
TR 19  Boundary Layer Transition Grit  
No. 100 Carborundum Grit  
Wing, Horizontal, Vertical (Both Sides)  
Width = .05 In.  
.5 In Aft of L.E.  
Fuselage Forebody  
Width = .05 In  
1.5 in Aft. of L.E.  

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### \( C_{12} \) Structural Model Control Vane

#### Total
- **Area**, \( \text{Ft}^2 \): 0.027
- **Span (Equiv.)**, Ft.: 0.25
- **Aspect Ratio**: 2.298
- **Taper Ratio**: 0.100

#### Chords, Inc.
- **Root (B.P. 0.0)**: 2.053
- **Tip (B.P. 1.499)**: 0.206
- **M.A.C. (B.P. 0.545)**: 1.381

#### Fus. Sta. of 0.25 M.A.C. (W.P. 0.417), In.
- **Dihedral Angle, Deg.**: -30.0
- **Incidence Angle, Deg.**: 0
- **Sweepback Angle, Deg.**: 0
- **Leading Edge**: 35.022°
- **0.25 Chord Element**: 23.458°

#### Airfoil Section
- **L.E.R.**: \((0.051)\) (Local Chord)
- **T.E.R.**: Knife Edge

#### Exposed
- **Area, \( \text{Ft}^2 \)**: 0.0065
- **Span (Equiv.), Ft.**: 0.110
- **Aspect Ratio**: 1.867
- **Taper Ratio**: 0.202

#### Chords, In.
- **Root (B.P. 0.837)**: 1.021
- **Tip (B.P. 1.499)**: 0.206
- **M.A.C. (B.P. 1.095)**: 0.704

#### Fus. Sta of 0.25 M.A.C. (W.P. 0.0999), In.
- **Fus. Sta of 0.25 M.A.C. (W.P. 0.0999), In.**: 5.548
### Horizontal Tail

**Total**

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**Exposed**

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### Vertical Tail

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REFERENCES


2. NA-70-550-2, TP/RS 0221-1-002, "Pretest Information for B-1 0.024-Scale Air Vehicle Rotary Derivative Tests in Langley 4 Foot UPWT", Dated 19 April 1971.


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TABLE III
LANGLEY 8' TPT TEST NO. 596 (CONT'D)
ACTUAL RUN SCHEDULE - YAW OSCILLATION

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*Runs 87, 88 and 89 Made with Pitch Oscillation Rig with Model at \( \phi = 270° \)

**Runs 90, 91 and 92 Made with 2 more Cables added to the Yaw Oscillation Rig to increase System Stiffness.
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<td>W47h58B51N54C12D--V46 TR19</td>
<td>65°</td>
<td>—</td>
<td>.75</td>
<td>—</td>
</tr>
<tr>
<td>W47h58B51N54C12D3H163V46 TR19</td>
<td>55°</td>
<td>0°</td>
<td>.75</td>
<td>—</td>
</tr>
</tbody>
</table>
### TABLE V (CONT'D)

<table>
<thead>
<tr>
<th>CONFIGURATION</th>
<th>( \Lambda )</th>
<th>( \delta_H )</th>
<th>C.G.POS. ( %C_W )</th>
<th>MACH NUMBER/RUN NUMBER</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td></td>
<td></td>
<td>( .20 )</td>
<td>( .55 )</td>
</tr>
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<td>w47h58b51n54c12d3h163v46 TR19</td>
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<td>0°</td>
<td>.75</td>
<td>—</td>
</tr>
<tr>
<td>w47h58b61n54c12d3--v46 TR19</td>
<td>65°</td>
<td>—</td>
<td>.75</td>
<td>—</td>
</tr>
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<td>w47h58b51n54c12d3h163v46 TR19</td>
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<td>+10°</td>
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</tr>
<tr>
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<td>0°</td>
<td>.75</td>
<td>—</td>
</tr>
<tr>
<td>w47h58b61n54c12d3h163v46 TR19</td>
<td>25°</td>
<td>0°</td>
<td>.45</td>
<td>—</td>
</tr>
<tr>
<td>w47h58b61n54c12d3--v46 TR19</td>
<td>25°</td>
<td>—</td>
<td>.45</td>
<td>—</td>
</tr>
<tr>
<td>w47h58b51n54c12d3--v46 TR19</td>
<td>25°</td>
<td>—</td>
<td>.45</td>
<td>—</td>
</tr>
</tbody>
</table>

*Runs 12, 64, 65, and 68 are Check Runs

**Runs 53, 54 and 55 were made at \( \Lambda = 40° \) only with Horizontal Steel Rod Stiffness Instead of Cable

**LANGLEY 7 x 10' HST TEST NO. 936**

**ACTUAL RUN SCHEDULE - PITCH OSCILLATION**

<table>
<thead>
<tr>
<th>CONFIGURATION</th>
<th>( \Lambda )</th>
<th>( \delta_H )</th>
<th>C.G.POS. ( %C_W )</th>
<th>MACH NUMBER/RUN NUMBER</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td></td>
<td></td>
<td>( .20 )</td>
<td>( .55 )</td>
</tr>
<tr>
<td>w47h58b51n54c12d3h163v46 TR19</td>
<td>25°</td>
<td>0°</td>
<td>.45</td>
<td>—</td>
</tr>
<tr>
<td>w47h58b61n54c12d3h163v46 TR19</td>
<td>25°</td>
<td>0°</td>
<td>.45</td>
<td>—</td>
</tr>
</tbody>
</table>
.024 SCALE ROTARY DERIVATIVE MODEL

Looking Forward

CHANNEL FILLER BLOCKS, B61
DATA IN INR OZ CENEC B)  

ANALYSIS OF FINISHES
