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Aerodynamic investigation of diamond planforms

Elbers, Wayne Keith, M.S.

The University of Texas at Arlington, 1991
AERODYNAMIC INVESTIGATION OF DIAMOND PLANFORMS

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AERODYNAMIC INVESTIGATION OF DIAMOND PLANFORMS

by

WAYNE KEITH ELBERS

Presented to the Faculty of the Graduate School of
The University of Texas at Arlington in Partial Fulfillment
of the Requirements
for the Degree of

MASTER OF SCIENCE IN AEROSPACE ENGINEERING

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I would like to thank Dr. Don Seath of The University of Texas at Arlington for his advisement and patience during my pursuit of this project. I also would like to thank Dr. Don Wilson, also of UTA, for aiding in the completion of the test. I greatly appreciate the efforts of Don Miller in fabricating the wing models and Scott Stuessy in setting up and running the test and improving the data reduction capability of the system. Lastly, I thank Charles Smith and Art Sheridan of General Dynamics Fort Worth Division for supporting and funding the project.

November 26, 1991
ABSTRACT

AERODYNAMIC INVESTIGATION OF DIAMOND PLANFORMS

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Wayne Keith Elbers, M.S.
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Supervising Professor: Donald Seath

A wind tunnel test was conducted to investigate the longitudinal aerodynamics, primarily lift curve slope, of delta and diamond wings. The test was performed at Mach numbers 0.75, 0.90, and 1.25. Fifteen half-span flat plate wings with sharp leading edges and trailing-edge flaps deflectable to 10 degrees were tested: aspect ratios of 1.0, 2.0, and 3.0 with five cutout factors from zero (delta wing) to -infinity at each aspect ratio. Test results were compared to predictions from a linear theory panel code.

Lift-curve slope was higher for the diamond wings than the delta wings. Maximum slopes occurred at cutout factors between -0.67 and -1.5. At subsonic speeds, the trailing-edge flap was more effective for the delta wings. Predictions and test results did not compare well concerning lift-curve slope magnitudes but did compare closely concerning lift-curve slope trends with respect to cutout factor.
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SYMBOLS AND ABBREVIATIONS

\( a \)  
wing planform geometric cutout factor; \( = \tan \Delta_{le}/\tan \Delta_{le} \)

\( a.c. \)  
aerodynamic center

\( AR \)  
wing aspect ratio; \( = b^2/S \)

\( b \)  
w ing span, inches

\( c_{mac} \)  
wing geometric mean aerodynamic chord, inches

\( C_{\text{root}} \)  
root chord, inches

\( C_{\text{tip}} \)  
tip chord, inches

\( C_A \)  
axial force coefficient

\( C_D \)  
drag force coefficient

\( C_{D0} \)  
zero-lift drag coefficient

\( C_{DL} \)  
drag-due-to-lift coefficient; \( = C_D-C_{D0} \)

\( C_L \)  
lift coefficient

\( C_{l\alpha} \)  
two-dimensional lift curve slope, per degree or per radian

\( C_{L\alpha} \)  
lift curve slope, per degree or per radian

\( C_{L\alpha} \)  
lift curve slope, per degree or per radian

\( C_{L\delta} \)  
delta lift coefficient due to flap deflection, per degree

\( C_{L\delta} \)  
delta lift coefficient due to flap deflection, per degree

\( C_M \)  
stability axis pitching moment coefficient, referenced to 0.5 \( C_{\text{root}} \)

\( C_{M\delta} \)  
delta pitching moment coefficient due to flap deflection, per degree

\( C_{M\delta} \)  
delta pitching moment coefficient due to flap deflection, per degree

\( C_N \)  
normal force coefficient

\( e \)  
span efficiency factor

\( M \)  
Mach number

\( q \)  
dynamic pressure, psf
\( R_e \)  \quad \text{unit Reynolds number, per foot}

\( S \)  \quad \text{wing area, in}^2

\( \alpha \)  \quad \text{angle of attack, degrees}

\( \delta \)  \quad \text{flap deflection angle, measured in x-z plane, positive trailing-edge down, degrees}

\( \varepsilon \)  \quad \text{complement angle to leading-edge sweep angle, degrees}

\( \Delta_{le} \)  \quad \text{leading-edge sweep angle, degrees}

\( \Delta_{te} \)  \quad \text{trailing-edge sweep angle, degrees}

\( \lambda \)  \quad \text{taper ratio; } = C_{tip}/C_{root}
INTRODUCTION

In the past, the diamond planform wing has received very little attention for military or commercial vehicle application. Some aircraft have had delta wings with moderately forward-swept trailing edges such as the B-58, F-102, and Saab Viggen. Recently, the YF-22 and YF-23 have flown and have wings with large forward trailing-edge sweeps. However, much of their aerodynamics are still classified.

Generic wind tunnel data on diamond planform wings are scarce, especially at transonic and supersonic speeds. One set of tests of arrow, delta, and diamond wings at various Mach numbers is documented in NASA TN D-7631.¹ The diamond wing in these tests had a leading-edge sweep of 74 degrees and a trailing-edge sweep of -35 degrees (cutout factor, $a = \tan \Delta_{le}/\tan \Delta_{le} = -0.202$).

In theory, the diamond planform is predicted to have significant advantages over the delta planform. For example, for wings of equal span and planform area (hence, equal aspect ratio), the diamond lift-curve slope may be 10 percent higher at subsonic speeds and as much as 30 percent higher at moderate supersonic speeds. Also, the diamond planform will produce a lower wave drag at low supersonic speeds than the delta. However, the trailing-edge flap control effectiveness of the diamond planform may be significantly less than that of the delta wing.

A wind tunnel test was conducted to investigate the transonic longitudinal aerodynamics, primarily the lift-curve slope, of sharp-leading-edge delta and
diamond planform wings. The effect of aspect ratio and cutout factor on the transonic longitudinal characteristics were determined. Fifteen flat plate delta and diamond wings were tested in the High Reynolds Number Transonic Wind Tunnel at the University of Texas at Arlington. Each wing included a trailing-edge flap deflectable to 10 degrees to investigate the planform effects on flap effectiveness.

A linear theory Carmichael finite-element lifting surface code was used to predict the aerodynamics of these planforms. The predictions are compared to the wind tunnel test results to determine the usefulness of the method for these wings at low angles of attack.
MODEL AND BALANCE DESCRIPTION

Fifteen half-span wing models of equal planform area were tested. The
wing geometries varied over three aspect ratios (1.0, 2.0, and 3.0) and included
five cutout factors (0.0, -0.667, -1.0, -1.5, and -∞) for each aspect ratio. The
wing geometry parameters are listed in Table 1 and shown in Figures 1, 2, and
3. The wing models are 0.25-inch thick flat plates made of heat treated AISI
Type 2 steel. The wing leading edges are beveled to a sharp edge. Geometric
planform dimensions are given in Table 1.

Each wing included trailing-edge flaps, sized to 12% of the wing area, of 0-
and 10-degrees deflection. The inboard flap chord is at 15% half-span and the
outboard flap chord is at 70% half-span. The inboard flap chord is 15% of the
local chord in length and the outboard flap chord is 30% of the local chord in
length. Each trailing-edge flap is attached to the wing with four flat-head socket
cap screws.

A cylindrical end-plate adapter attaches the wing models to a five-
component balance that was designed and fabricated specifically for the UTA
High Reynolds Number Transonic Wind Tunnel. Only three components were
used on this test to obtain normal force, axial force, and pitching moment. The
cylindrical balance is installed in the plenum chamber external to the test
section. A ring adapter with an O-ring seal attaches the balance to the outside
of the tunnel, extending the balance inside the plenum as far as the test section
wall. A sketch of the balance, including a listing of its load capacities, and the
end-plate adapter are shown in Figure 4. Balance accuracies from the balance
calibration are approximately equivalent to one-half of one percent of the balance design loads, which equate to $0.013 \ C_N$, $0.0014 \ C_A$, and $0.0004 \ C_m$ when nondimensionalized by $q=25$ psi, $S=10.656$ square inches, and $mac=6.16$ inches (Wings 31-35 at about 0.9 Mach number).

Deflecting the model to angle of attack is achieved by rotating the balance/model with respect to the tunnel centerline. A machine-polished aluminum bar and a bubble level were used to set the desired angle of attack. Based on the normal force results, angle of attack accuracy was approximately $\pm 0.5$ degrees.
Table 1  Model Wing Geometry

<table>
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<tr>
<th>Wing No.</th>
<th>Aspect Ratio (AR)</th>
<th>Cutout Factor (c)</th>
<th>Area (in.²)</th>
<th>Leading Edge Sweep (deg.)</th>
<th>Trailing Edge Sweep (deg.)</th>
<th>Root Chord (in.)</th>
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Figure 1 Aspect Ratio 1.0 Wings
Figure 2  Aspect Ratio 2.0 Wings
Figure 3  Aspect Ratio 3.0 Wings
BALANCE DESIGN LOADS

<table>
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</tr>
<tr>
<td>AXIAL FORCE (R₃)</td>
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</tr>
<tr>
<td>PITCHING MOMENT (R₅)</td>
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</tr>
<tr>
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<tr>
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<tr>
<td>ROOT BENDING MOMENT (R₄)</td>
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<tr>
<td>DUE TO AXIAL FORCE</td>
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Figure 4 Sketch of Wind Tunnel Balance
TUNNEL DESCRIPTION AND TEST CONDITIONS

This investigation was conducted in the UTA High Reynolds Number Transonic (HIRT) Wind Tunnel. A drawing of the tunnel from Reference 2 is shown in Figure 5. The HIRT tunnel is a Ludwieg tube tunnel capable of simulating Reynolds numbers from $10^7$ to over $10^8$ per foot at Mach numbers from 0.5 to 1.3. Ludwieg tube tunnels are based on an unsteady expansion wave concept for acceleration of high-pressure air stored in a charge tube to transonic Mach numbers, similar to the flow within the driver tube of a conventional shock tube. Operation of the tunnel can be summarized as follows: (1) the entire tunnel system is pressurized to the desired charge tube level, (2) the sliding sleeve start valve is opened rapidly, generating an unsteady expansion wave that propagates upstream into the charge tube and initiating flow towards the open valve at the downstream end of the system, (3) the expansion wave clears the nozzle, resulting in steady flow in the test section for the time it takes for the expansion wave to travel the length of the charge tube, reflect off the end, and return through the test section, and (4) subsequent wave reflections pass through the tube until the tube pressure is back to ambient. A steady flow period of approximately 100 milliseconds can be achieved in the test section through the proper operation of the starting mechanism.

The test section, which is enclosed in a plenum chamber, measures 7.3 by 9.15 inches in cross section and is 25.4-inches long. The tunnel contains porous test section walls, which are used to alleviate wall interference effects,
attain low supersonic Mach number operation and prevent shock wave reflections back onto the model at the low supersonic speeds.

The test was conducted at nominal Mach numbers of 0.75, 0.90, and 1.25 at a Reynolds number of approximately 24 million per foot. Test Mach number was accurate to ±0.03 and dynamic pressure was accurate to ±1.0 psi. At the start of the test, data were acquired at 0, 1, 5, and 10-degrees angle of attack. However, axial force errors were discovered. Further study found that the axial force gages had debonded from the balance. The cause was the increasing amplitude of the dynamic loads as the angle of attack was increased to 10 degrees. After the balance was regaged and recalibrated, the test was restarted with an angle of attack limit of 5 degrees.
Figure 5  Sketch of UTA HIRT Wind Tunnel
DATA REDUCTION

The UTA HIRT Wind Tunnel produces primarily unsteady flow in the test section throughout the run with a period of steady flow in the middle of the run. The unsteady test section conditions, including those during the period of steady flow, and associated model forces and moments are measured every 0.5 milliseconds from the start of the run through 502 milliseconds. The readings are communicated via an IEEE 488 bus to either an IBM XT or HP VECTRA microprocessor. Two files are created per tunnel run to store the readings: a pressure file and a balance file. The balance data file is processed through a fifth-order Butterworth filter program to compute steady-state information at each 0.5-millisecond point. The new steady-state (filtered) model force and moment data are stored in a separate data file.

To determine the time period during each run over which steady flow is maintained, the Mach number and test section total pressure are scanned. Steady flow start and stop times are recorded by hand. A data reduction program is run on the microprocessor to average normal force, axial force, pitching moment, Mach number, and total pressure over the steady flow time period. The averaged values are then written to a data file.

The axial force produced by the end plate which mounts the wings to the balance was separately measured and is subtracted from wing-installed axial force to obtain the true wing axial force. Normal force and pitching moment produced by the end plate are very small and considered negligible for this study. Test section dynamic pressure ($q$) is calculated from Mach number and
total pressure. No test section wall or blockage corrections were employed in
the data reduction procedure. Normal force, axial force, and pitching moment
are nondimensionalized by $q$, $S$, and $mac$ to compute coefficients. Pitching
moment is referenced to 50% of the root chord. Lift and drag coefficients are
calculated by rotating normal force and axial force coefficient through angle of
attack. The standard equations used are:

\[
\begin{align*}
C_N &= NF/(q^*S) \\
C_A &= (AF-\text{AFplate})/(q^*S) \\
C_{M,5cr} &= PM/(q^*S*mac) \\
C_L &= C_N^*\cos\alpha - C_A^*\sin\alpha \\
C_D &= C_A^*\cos\alpha + C_N^*\sin\alpha.
\end{align*}
\]
DISCUSSION OF RESULTS

Lift Coefficient and Lift Curve Slope

Test Data

Lift coefficients were acquired on all fifteen planforms at Mach numbers of 0.75, 0.90, and 1.25 at angles of attack of 0, 1, and 5 degrees. Straight lines are drawn through the plotted points to obtain the lift-curve slopes. Figure 6 presents lift-curve slope per degree as a function of cutout factor for aspect ratios 1.0, 2.0, and 3.0 at all Mach numbers tested. Five wings were retested to check the repeatability of the data with respect to the test setup and to attempt to improve upon the initial test results. Wings 14, 15, and 21 were rerun at Mach numbers of 0.75, 0.90, and 1.25 at angles of attack of 0, 1, and 5 degrees. Wings 23 and 24 were rerun at Mach numbers of 0.75 and 0.90 at all three angles of attack.

The lift coefficient data for the 1.0 aspect ratio wings (11 through 15) are plotted in Figures 7 through 11. At 0.75 Mach number, the delta wing (Wing 11, a=0) produced a higher lift-curve slope than any of the diamond-shaped wings. At Mach numbers 0.90 and 1.25, the diamond wings produced higher lift-curve slopes with the maximum occurring at a=-0.67. Lift-curve slope as a function of Mach number is presented for all five wings in Figure 12. These data show similar magnitudes of lift-curve slopes between 0.75 and 0.90 Mach number.

The lift coefficient data for the 2.0 aspect ratio wings (21 through 25) are presented in Figures 13 through 17. Lift-curve slopes are plotted with respect to cutout factor in Figure 6. At all three Mach numbers tested the diamond-planform
LIFT CURVE SLOPE COMPARISON
WIND TUNNEL DATA

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Figure 6 Lift Curve Slope Comparison for Aspect Ratios 1.0, 2.0, 3.0
Figure 7 Wing11 (AR=1.0, a=0.0) Lift Coefficient Data
Figure 8: Wing12 (AR=1.0, a=0.67) Lift Coefficient Data

LIFT COEFFICIENT

AR=1.0, WING12
WIND TUNNEL DATA

M=1.25

M=0.90

M=0.75

ANGLE OF ATTACK (deg.)

CL
Figure 9: Wing13 (AR=1.0, α=1.0) Lift Coefficient Data
Figure 10  Wing14 (AR=1.0, α=-1.5) Lift Coefficient Data
Figure 11 Wing15 \((AR=1.0, a=\infty)\) Lift Coefficient Data

LIFT COEFFICIENT

\(AR=1.0\)

WIND TUNNEL DATA
LIFT CURVE SLOPE COMPARISON

WIND TUNNEL DATA
ASPECT RATIO = 1.0

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<tr>
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<tr>
<td>√</td>
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</tbody>
</table>

Figure 12 1.0 Aspect Ratio Lift Curve Slope Comparison with Respect to Mach Number
Figure 13 Wing21 (AR=2.0, a=0.0) Lift Coefficient Data
Figure 14 Wing22 (AR=2.0, α=-0.67) Lift Coefficient Data
Figure 15: Wing23 (AR=2.0, a=1.0) Lift Coefficient Data
Figure 16 Wing24 (AR=2.0, a=1.5) Lift Coefficient Data
Figure 17 Wing25 (AR=2.0, a=∞) Lift Coefficient Data
wings produced higher slopes than the delta wing (Wing 21, \(a=0\)). The highest lift-curve slope is shown by the \(a=-1.5\) wing. It is suspected that the maximum lift-curve slope should occur at \(a=-1.0\) even though this wing was retested. Lift-curve slope is plotted with respect to Mach number for the aspect ratio 2.0 wings in Figure 18. These data show higher lift-curve slopes at 1.25 Mach number.

Lift coefficient data for the 3.0 aspect ratio wings (Wings 31 through 35) are plotted in Figures 19 through 23. These data are also plotted with respect to cutout factor in Figure 6. At all three Mach numbers the diamond-planform wings produced higher slopes than the delta wing (Wing 31, \(a=0\)), with the maximum at \(a=-1.0\) at Mach 0.75 and 1.25 and at \(a=-0.67\) at Mach 0.90. Lift-curve slope is plotted as a function of Mach number in Figure 24. The results are very similar to the 2.0 aspect ratio results from Figure 18 which show higher slopes at 1.25 Mach number.

The lift coefficient data from the retested wings did not repeat well (see Figures 10, 11, 13, 15, and 16). These undesirable results form a basis for data uncertainty evaluation (see Appendix).

**Comparison to Previous Test Data**

A summary of lift-curve-slope test data for delta wings is given in Figure 25. The data are plotted as \(C_L/AR\) per radian with respect to the function \(\sqrt{1-M^2} \cdot \tan \varepsilon\) for subsonic speeds and \(\sqrt{M^2-1} \cdot \tan \varepsilon\) for supersonic speeds. \(\varepsilon\) is the complement angle to the leading-edge sweep (\(\varepsilon = 90^\circ - \Delta \theta\)). Data from the current study delta wings, Wings 11 (AR=1.0), 21 (AR=2.0), and 31 (AR=3.0), are also represented in the figure.

The 1.0 aspect ratio delta wing subsonic data compare well with the previous test data. At 1.25 Mach number, the measured lift-curve slope is 0.17 per radian
LIFT CURVE SLOPE COMPARISON

WIND TUNNEL DATA
ASPECT RATIO = 2.0

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Figure 18 2.0 Aspect Ratio Lift Curve Slope Comparison with Respect to Mach Number
Figure 19 Wing31 (AR=3.0, a=0.0) Lift Coefficient Data
Figure 20 Wing32 (AR=3.0, a=-0.67) Lift Coefficient Data
Figure 21 Wing33 (AR=3.0, a=-1.0) Lift Coefficient Data
LIFT COEFFICIENT
AR=3.0 WING34
WIND TUNNEL DATA

Figure 22 Wing34 (AR=3.0, α=-1.5) Lift Coefficient Data
Figure 23 Wing35 (AR=3.0, a=∞) Lift Coefficient Data
LIFT CURVE SLOPE COMPARISON

WIND TUNNEL DATA
ASPECT RATIO = 3.0

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</table>

Figure 24  3.0 Aspect Ratio Lift Curve Slope Comparison with Respect to Mach Number
below the previous "mean" curve. The 2.0 and 3.0 aspect ratio delta wing lift-curve slopes from the current test are low by as much as 29 percent compared to the previous data. For 2.0 aspect ratio, the worst point is at 1.25 Mach number; lift-curve slope is 0.78 per radian below previous test data. At 0.90 Mach number it is 0.68 per radian low. For 3.0 aspect ratio, the worst point is at 0.90 Mach number; lift-curve slope is 0.84 per radian below previous test data. At 1.25 Mach number it is 0.69 per radian low.

This comparison indicates that the magnitudes of the lift-curve slopes from the current test are in error. The differences are primarily due to angle-of-attack measurement inaccuracies and balance normal force inaccuracies. An uncertainty analysis, which is presented in the Appendix, provides an estimation of the amount of uncertainty in the test lift-curve slope data. However, it does not fully account for the differences in the delta wing lift-curve slope comparison and does not explain why the current test slopes are all low.

One possible explanation for the differences is that angle of attack was set consistently lower than the estimated uncertainty when rotating the model to 5 degrees. Another possible explanation is that the filter program erroneously computed low steady-state values. It is also possible that the model-to-balance end-plate adapter grounded on the test section wall due to model and balance deflection under normal force loads. The result would be low measured normal forces. Also, since the 2.0 and 3.0 aspect ratio delta wings produced larger normal forces than the 1.0 aspect ratio delta wing and, thus, they deflected more, it is plausible that grounding occurred only at the higher load conditions. This might explain why the 1.0 aspect ratio delta wing data compared well and the 2.0 and 3.0 aspect ratio delta wing data was low. Blockage of the test section flow by the model can cause errors in the experimental measurements. The model blockage
at 5 degrees angle of attack is approximately 1.4% of the test section area. This is a small amount of blockage and has a negligible effect on the results.

However, it is the author's belief that the trends of lift-curve slope with respect to cutout factor from the current test are reasonable and actual. These trends are substantiated using test data of delta and diamond wings at subsonic and supersonic speeds. The wings have a leading-edge sweep of 74 degrees. The delta wing aspect ratio and cutout factor are 1.102 and 0.0. The diamond wing aspect ratio and cutout factor are 0.956 and -0.202. The computed lift-curve slopes are divided by the respective wing aspect ratio to compare data at an equal aspect ratio of 1.0 and are plotted with respect to Mach number in Figure 26. The diamond wing produces a higher lift-curve slope than the delta wing; the slope is as much as 19 percent higher at 1.20 Mach number. Data from the current test are also shown in the figure and show very similar results.

Comparison to Linear Theory Predictions

The lift-curve slopes of all fifteen wings were predicted using linear theory. The Carmichael finite-element lifting surface method was employed. The lift-curve slopes computed from the wind tunnel data are compared to the linear theory predictions in Figures 27 through 29. The wing planforms were modeled as shown in Figures 30 through 32.

At 1.0 aspect ratio the lift-curve slope test data compare favorably to linear theory. The linear theory lift-curve slope trends with respect to cutout factor produce a maximum at approximately $a=-1.0$. The test data indicates a maximum slope at $a=-0.67$.

The 2.0 aspect ratio test lift-curve slopes are low compared to linear theory predictions. This result concurs with the previously described comparison of the
LIFT CURVE SLOPE COMPARISON
WIND TUNNEL DATA
ASPECT RATIO = 1.0

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Figure 26 Lift Curve Slope per AR Comparison with Previous Test Data for Delta and Diamond Wings
Figure 27 Lift Curve Slope Test-To-Theory Comparison for 1.0 Aspect Ratio
LIFT CURVE SLOPE TEST-TO-THEORY COMPARISON
ASYMETRY RATIO = 2.0

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<td></td>
<td>Linear Theory Prediction</td>
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Figure 28 Lift Curve Slope Test-To-Theory Comparison for 2.0 Aspect Ratio
LIFT CURVE SLOPE TEST-TO-THEORY COMPARISON

ASPECT RATIO = 3.0

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<td>Linear Theory Prediction</td>
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Figure 29  Lift Curve Slope Test-To-Theory Comparison for 3.0 Aspect Ratio
Figure 30  Linear Theory Panel Models for Aspect Ratio 1.0 Wings
Figure 31  Linear Theory Panel Models for Aspect Ratio 2.0 Wings
Figure 32  Linear Theory Panel Models for Aspect Ratio 3.0 Wings
test data to previous test data for delta wings. The linear theory and test slopes vary more with respect to cutout factor than the 1.0 aspect ratio wings, especially at 1.25 Mach number. The maximum lift-curve slope predicted by linear theory occurs at -0.67 cutout factor at all three Mach numbers. The test slopes are highest at cutout factors of -1.5 or lower. It was previously mentioned that the subsonic lift-curve slopes of the -1.0 cutout-factor wing (Wing 23) are suspect. It is believed that the maximum test subsonic lift-curve slopes will occur at -1.0 cutout factor. The conclusion is that the test data optimize at more negative cutout factors than linear theory predicts for 2.0 aspect ratio wings.

The 3.0 aspect ratio lift-curve slopes are low compared to linear theory. This also concurs with the delta wing current-to-previous test data comparison. The variation of lift-curve slope with respect to cutout factor is similar to the results for 2.0 aspect ratio. The maximum slope predicted by linear theory is at approximately -0.67 cutout factor at subsonic speeds and at -1.0 at 1.25 Mach number. The test data coincide with the linear theory trends except at 0.75 Mach number, where the maximum test slope occurs at $a=-1.0$.

Another comparison can be made between the current test delta wing lift-curve slopes and theoretical values. The two-dimensional lift-curve slope of a flat plate is given by
\[ C_{l\alpha} = \frac{2\pi}{\sqrt{1-M^2}} \quad \text{for } M<1, \]
\[ C_{l\alpha} = \frac{4\sqrt{M^2-1}}{M} \quad \text{for } M>1. \]

The finite (three-dimensional) wing lift-curve slope is
\[ C_{L\alpha} = C_{l\alpha}/[1+(C_{l\alpha}/(\pi e*AR))], \]
where $e=1$ for an elliptical lift distribution.
The theoretical lift-curve slopes for the current test delta wing models are computed as

<table>
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<th>Aspect Ratio</th>
<th>CL₀/AR (per radian) at M=0.75</th>
<th>M=0.90</th>
<th>M=1.25</th>
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<td>1.58</td>
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</table>

These values are much higher than those of the current or previous test results. This is due to non-elliptical lift distribution. The theoretical lift-curve slopes would match the subsonic previous test data at an e of approximately 0.55. At supersonic speeds, they would match at an e of approximately 0.87.
Pitching Moment Coefficient and Aerodynamic Center Location

Test Data

Pitching moment coefficients, referenced about the 0.5 root chord location, were obtained at 0.75, 0.90, and 1.25 Mach numbers at angles of attack of 0, 1, and 5 degrees. They are plotted with respect to lift coefficient in Figures 33 through 47. Straight lines are drawn through these data points to compute the slope of the pitching moment with respect to lift for each wing. Combined with moment reference center, the aerodynamic center (a.c.) location in the stability axes, presented in Figure 48, is computed as follows:

\[ \text{a.c. (\%mac) = } -\frac{dC_M}{dC_L} + ((0.5*\text{C}_{\text{root}}) - (\tan(\Delta\alpha)*(b/2))/\text{mac}) \]

In general, the a.c. moved forward as a percentage of the mac as cutout factor varied from zero (delta wing) to higher negative values. This result is primarily due to lower wing leading-edge sweep and the fact that the location of the wing tip moves forward as a percentage of the mac as cutout factor becomes more negative.

Aerodynamic center location is plotted as a function of Mach number in Figures 49 through 51. The data show very little movement of a.c. between 0.75 and 0.90 Mach number; in some cases the a.c. was more forward for the higher Mach number. The a.c. generally moved aft as Mach number increased from 0.90 to 1.25. The magnitude of the a.c. shift increased as aspect ratio increased and as cutout factor decreased to \(-\infty\).

The repeat data for Wings 14 and 15 (Figures 36 and 37) did not compare well to the original runs. However, the retested data for Wings 21, 23, and 24 (Figures 38, 40, and 41) provided good comparisons to the original runs.
Figure 33 Wing 11 (AR=1.0, a=0.0) Pitching Moment Coefficient Data

CL vs. CM

M = 1.25

M = 0.90

M = 0.75
Figure 34: Wing12 (AR=1.0, α = 0.67) Pitching Moment Coefficient Data
Figure 35 Wing13 (AR=1.0, a=-1.0) Pitching Moment Coefficient Data
Figure 36  Wing14 (AR=1.0, α=-1.5) Pitching Moment Coefficient Data
Figure 37: Wing15 (AR=1.0, a=45°) Pitching Moment Coefficient Data
Figure 38 Wing21 (AR=2.0, a=0.0) Pitching Moment Coefficient Data

CL vs. CM (AR=2.0, WIND TUNNEL DATA)

M = 0.75
M = 0.90
M = 1.25
Figure 40 Wing23 (AR=2.0, α=-1.0) Pitching Moment Coefficient Data
Figure 41 Wing24 (AR=2.0, a=-1.5) Pitching Moment Coefficient Data
Figure 42: Wing25 (AR=2.0, a=∞) Pitching Moment Coefficient Data
Figure 43 Wing31 (AR=3.0, a=0.0) Pitching Moment Coefficient Data

CL VS. CM
AR=3.0, WIND TUNNEL DATA

M=1.25

M=0.90

M=0.75

0.40 0.30 0.20 0.10 0.00
-0.10 -0.2 -0.1 0.0 0.1 0.2
Figure 44 Wing32 (AR=3.0, α=0.67) Pitching Moment Coefficient Data
Figure 46 Wing34 (AR=3.0, a=1.5) Pitching Moment Coefficient Data

$C_L$ vs. $C_M$

$M = 1.25$

$M = 0.90$

$M = 0.75$
Figure 47: Wing35 (AR=3.0, a=∞) Pitching Moment Coefficient Data
AERODYNAMIC CENTER COMPARISON
WIND TUNNEL DATA

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</table>

M=0.75  M=0.90  M=1.25

a (Cutout Factor)  a (Cutout Factor)  a (Cutout Factor)

Figure 48  Aerodynamic Center Comparison for Aspect Ratios 1.0, 2.0, 3.0
AERODYNAMIC CENTER COMPARISON

WIND TUNNEL DATA
ASPECT RATIO = 1.0

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Figure 49 1.0 Aspect Ratio Aerodynamic Center Comparison with Respect to Mach Number
Figure 50 2.0 Aspect Ratio Aerodynamic Center Comparison with Respect to Mach Number
AERODYNAMIC CENTER COMPARISON
WIND TUNNEL DATA
ASPECT RATIO = 3.0

<table>
<thead>
<tr>
<th>SYM</th>
<th>Cutout Factor</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>0.8</td>
</tr>
<tr>
<td></td>
<td>0.87</td>
</tr>
<tr>
<td></td>
<td>-1.00</td>
</tr>
<tr>
<td></td>
<td>-1.50</td>
</tr>
<tr>
<td></td>
<td>-INFINITY</td>
</tr>
</tbody>
</table>

Figure 51 3.0 Aspect Ratio Aerodynamic Center Comparison with Respect to Mach Number
The current test 1.0 aspect ratio a.c. locations are compared to previous delta and diamond wing test data in Figure 52. The delta wing a.c. locations compare closely. The diamond wing a.c. locations trend correctly: as cutout factor becomes more negative, the a.c. moves forward with respect to mac.

Comparison to Linear Theory Predictions

The Carmichael lifting surface method\(^7\) was also used to predict aerodynamic center location. The linear theory aerodynamic center locations are compared to those computed from the test data in Figures 53 through 55. The subsonic test a.c. locations compare well with predictions except for the \( \rightarrow \) cutout factor wings (Wings 15, 25, and 35). The test a.c. locations produced by these wings occurred from 4 to 10 percent mac forward of the locations predicted by linear theory. At 1.25 Mach number the test a.c. locations generally occurred forward of the predicted locations.
AERODYNAMIC CENTER COMPARISON

WIND TUNNEL DATA

<table>
<thead>
<tr>
<th>SYM</th>
<th>Aspect Ratio</th>
<th>Cutout Factor</th>
<th>Data Source</th>
</tr>
</thead>
<tbody>
<tr>
<td>O</td>
<td>1.102 (Delta)</td>
<td>0.0</td>
<td>Reference 1</td>
</tr>
<tr>
<td>D---</td>
<td>0.956 (Diamond)</td>
<td>-0.202</td>
<td>Reference 1</td>
</tr>
<tr>
<td>D---</td>
<td>1.0 (Delta)</td>
<td>0.0</td>
<td>Current Test</td>
</tr>
<tr>
<td>D---</td>
<td>1.0 (Diamond)</td>
<td>-0.6</td>
<td>Current Test</td>
</tr>
</tbody>
</table>

_Figure 52_ Aerodynamic Center Comparison with Previous Test Data of Delta and Diamond Wings
AERODYNAMIC CENTER TEST-TO-THEORY COMPARISON
ASPECT RATIO = 1.0

<table>
<thead>
<tr>
<th>SYM</th>
<th>Source</th>
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<tbody>
<tr>
<td></td>
<td>Wind Tunnel Data</td>
</tr>
<tr>
<td></td>
<td>Linear Theory Prediction</td>
</tr>
</tbody>
</table>

Figure 53 1.0 Aspect Ratio / Aerodynamic Center Test-To-Theory Comparison
AERODYNAMIC CENTER TEST-TO-THEORY COMPARISON
ASPECT RATIO = 2.0

<table>
<thead>
<tr>
<th>SYM</th>
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<tbody>
<tr>
<td></td>
<td>Wind Tunnel Data</td>
</tr>
<tr>
<td></td>
<td>Linear Theory Prediction</td>
</tr>
</tbody>
</table>

Figure 54 2.0 Aspect Ratio / Aerodynamic Center Test-To-Theory Comparison
AERODYNAMIC CENTER TEST-TO-THEORY COMPARISON

ASPECT RATIO = 3.0

<table>
<thead>
<tr>
<th>SYM</th>
<th>Source</th>
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<tbody>
<tr>
<td></td>
<td>Wind Tunnel Data</td>
</tr>
<tr>
<td></td>
<td>Linear Theory Prediction</td>
</tr>
</tbody>
</table>

Figure 55 3.0 Aspect Ratio / Aerodynamic Center Test-To-Theory Comparison
Drag Coefficient

Test Data

Drag coefficients were acquired on all fifteen planforms at all test conditions. The drag coefficients are plotted with respect to the square of the lift coefficients. The plots for Wings 11 through 15 (1.0 aspect ratio) are presented in Figures 56 through 60.

The subsonic drag levels are similar over the range of cutout factors tested considering drag coefficient uncertainties of more than ±0.001. The drag levels are slightly lower at cutout factors of -0.67, -1.0, and -1.5. The drag coefficients at Mach number 1.25 decrease as cutout factor becomes more negative until a large drag increase at a = -∞. The lowest drag levels occur at a cutout factor of -1.5. These results are consistent with the theory that wave drag predominates low-CL supersonic drag and that, at relatively high leading-edge sweeps, the cross-sectional area curve with the lowest slopes produces lower wave drag. The zero and -∞ cutout factor wings (Wings 11 and 15) have area curves with higher slopes than the other wings because of zero trailing-edge and leading-edge sweep, respectively.

The drag coefficients of Wings 21 through 25 (2.0 aspect ratio) are plotted in Figures 61 through 65. The subsonic drag coefficient characteristics produce the same trends as the 1.0 aspect ratio data: the drag levels are similar over the range of cutout factors with slightly lower levels at -0.67, -1.0, and -1.5. At 1.25 Mach number the lowest drag coefficients occur at a cutout factor of zero (delta wing). The drag levels at -0.67, -1.0, and -1.5 cutout factors are similar to each other and higher than the delta wing. The drag at a = -∞ cutout factor is higher yet. These trends differ from those for the 1.0 aspect ratio wings. The reason the 2.0 aspect
Figure 56 Wing 11 (AR=1.0, a=0.0) Drag Coefficient Data
Figure 57: Wing12 (AR=1.0, a=0.67) Drag Coefficient Data
Figure 58 Wing13 (AR=1.0, a=-1.0) Drag Coefficient Data
Figure 59 (AR = 1.0, a = 1.5) Drag Coefficient Data
Figure 61 Wing2 (AR=2.0, α=0.0) Drag Coefficient Data
Figure 62: Wing22 (AR=2.0, a=0.67) Drag Coefficient Data

DRAG COEFFICIENT

M=1.25

M=0.90

M=0.75

0.080
0.064
0.048
0.032
0.016
0.000

0.08
0.06
0.04
0.02
0

CD

CL

AR=WING22

WIND TUNNEL DATA
Figure 64 Wing24 (AR=2.0, a=1.5) Drag Coefficient Data

AR=2.0
WING24
WIND TUNNEL DATA

M=1.25

M=0.90

M=0.75

CL

CD
Figure 6. Wing 25 (AR=2.0, α=∞) Drag Coefficient Data

DRAG COEFFICIENT

AR=2.0, WING25
WIND TUNNEL DATA

M=1.25

M=0.90

M=0.75

C_D

0.080 0.064 0.048 0.032 0.016 0.000

0.02 0.04 0.06 0.08

0 0.02 0.04 0.06 0.08
ratio delta wing has a lower supersonic drag level, even though it has a worse area
curve shape, is the leading-edge sweeps of the other wings are sufficiently lower
and adversely affect wave drag.

The drag coefficients of Wings 31 through 35 (3.0 aspect ratio) are plotted in
Figures 66 through 70. The subsonic drag characteristics are the same as the 1.0
and 2.0 aspect ratio data. At 1.25 Mach number the drag trends resemble those of
the 2.0 aspect ratio data; the zero cutout factor delta wing produced the lowest
drag. The reason is also the same; the lower leading-edge sweeps of the negative
cutout factor wings adversely affect wave drag.
Figure 66 Wing31 (AR=3.0, a=0.0) Drag Coefficient Data
Figure 67 Wing32 (AR=3.0, α=-0.67) Drag Coefficient Data
Figure 68 Wing33 (AR=3.0, a=-1.0) Drag Coefficient Data

Drag Coefficient

AR=3.0, WING33
WIND TUNNEL DATA

M=1.25

M=0.90

M=0.75
Figure 69 Wing94 (AR=3.0, a=1.5) Drag Coefficient Data

DRAG COEFFICIENT

\[ M = 0.75 \]

\[ C_D \]

\[ C_L \]

\[ M = 0.90 \]

\[ C_D \]

\[ C_L \]

\[ M = 1.25 \]

\[ C_D \]

\[ C_L \]
Figure 70 Wing35 (AR=3.0, α=∞) Drag Coefficient Data
Effect of Trailing-Edge Flap Deflection

Test Data

Lift, drag, and pitching moment coefficients were obtained on all fifteen planforms with the trailing-edge flap deflected 10 degrees to measure flap effects. These data were acquired at Mach numbers of 0.75, 0.90, and 1.25 and at angles of attack of 0, 1, and 5 degrees. The lift coefficients are plotted with respect to angle of attack in Figures 71 through 85. Pitching moment coefficients are plotted with respect to lift coefficient in Figures 86 through 100. Drag coefficients are presented as a function of lift coefficient squared in Figures 101 through 115.

Lift coefficient increments due to 10-degrees trailing-edge flap deflection are calculated at each test condition by subtracting clean wing data from the flap-deflected data. Delta lift coefficients per degree flap deflection are computed by dividing the increments by 10 degrees. Plots of $C_{L\delta}$ per degree as a function of cutout factor are presented in Figures 116, 117, and 118 for aspect ratios 1.0, 2.0, and 3.0, respectively.

At subsonic speeds delta lift coefficient generally decreases with decreasing cutout factor. The delta wings ($a=0$) produce the highest increments except for a couple of points. Note that the difference in $C_{L\delta}$ between the delta wing and the diamond-shaped wings decreases as aspect ratio increases.

At 1.25 Mach number the delta lift coefficients trend differently than at subsonic speeds. For the aspect ratio 1.0 wings, the delta wing ($a=0$) and $a=-0.67$ wing produced similar results. The rest of the diamond-shaped wings produced slightly lower increments; the levels were similar from $a=-1.0$ to $-\infty$. For the aspect ratio 2.0 and 3.0 wings, the diamond wings provided higher delta lift coefficients than the
Figure 7: Wing 1 (AR=1.0, a=0.0) Lift Coefficient Data with 10-Degree Flap Deflection.
Figure 72. Wing 12 (AR=1.0, a=0.67) Lift Coefficient Data with 10-Degree Flap Deflection
Figure 74 Wing14 (AR=1.0, $a=-1.5$) Lift Coefficient Data with 10-Degree Flap Deflection
LIFT COEFFICIENT
AR=1.0  WING15
WIND TUNNEL DATA
FLAP DEFLECTION = 10 DEGREES

M=0.75

M=0.90

M=1.25

Figure 75 Wing15 (AR=1.0, a=∞) Lift Coefficient Data with 10-Degree Flap Deflection
Figure 76  Wing21 (AR=2.0, a=0.0) Lift Coefficient Data with 10-Degree Flap Deflection
Figure 77: Wing22 (AR=2.0, a=0.67) Lift Coefficient Data with 10-Degree Flap Deflection
LIFT COEFFICIENT

AR=2.0 WING23
WIND TUNNEL DATA
FLAP DEFLECTION = 10 DEGREES

M=0.75

M=0.90

M=1.25

Figure 78 Wing23 (AR=2.0, a=-1.0) Lift Coefficient Data with 10-Degree Flap Deflection
Figure 79: Wing24 (AR=2.0, a=-1.6) Lift Coefficient Data with 10-Degree Flap Deflection
Figure 80 Wing25 (AR=2.0, a=∞) Lift Coefficient Data with 10-Degree Flap Deflection
Figure B1  Wing31 (AR=3.0, α=0.0) Lift Coefficient Data with 10-Degree Flap Deflection
Figure 82: Wing32 (AR=3.0, α=0.67) Lift Coefficient Data with 10-Degree Flap Deflection
Figure 83 Wing33 (AR=3.0, a=-1.0) Lift Coefficient Data with 10-Degree Flap Deflection
LIFT COEFFICIENT

AR=3.0 WING34
WIND TUNNEL DATA
FLAP DEFLECTION = 10 DEGREES

M=0.75

M=0.90

M=1.25

Figure 84 Wing34 (AR=3.0, \( a=-1.5 \)) Lift Coefficient Data with 10-Degree Flap Deflection
Figure 85  Wing35 (AR=3.0, a=∞) Lift Coefficient Data with 10-Degree Flap Deflection
Figure 86 Wing11 (AR=1.0, a=0.0) Pitching Moment Coefficient Data with 10-Degree Flap Deflection
Figure 87 Wing12 (AR=1.0, a=0.67) Pitching Moment Coefficient Data with 10-Degree Flap Deflection.
Figure 89 Wing14 (AR=1.0, α=-1.5) Pitching Moment Coefficient Data with 10-Degree Flap Deflection
Figure 90: Wing15 (AR=1.0, α=–) Pitching Moment Coefficient Data with 10-Degree Flap Deflection
Figure 91 Wing21 (AR=2.0, a=0.0) Pitching Moment Coefficient Data with 10-Degree Flap Deflection
Figure 92: Wing 22 (AR=2.0, α=0.37) Pitching Moment Coefficient Data with 10-Degree Flap Deflection
Figure 93  Wing23 (AR=2.0, ø=-1.0) Pitching Moment Coefficient Data with 10-Degree Flap Deflection
Figure 94. Wing24 (AR=2.0, a=1.5) Pitching Moment Coefficient Data with 10-Degree Flap Deflection
Figure 95  Wing25 (AR=2.0, a=∞) Pitching Moment Coefficient Data with 10-Degree Flap Deflection
Figure 96 Wing31 (AR=3.0, a=0.0) Pitching Moment Coefficient Data with 10-Degree Flap Deflection
Figure 97 Wing32 (AR=3.0, a=0.67) Pitching Moment Coefficient Data with 10-Degree Flap Deflection
Figure 99 Wing34 (AR=3.0, a=-1.5) Pitching Moment Coefficient Data with 10-Degree Flap Deflection
Figure 100  Wing35 (AR=3.0, a=∞) Pitching Moment Coefficient Data with 10-Degree Flap Deflection
Figure 101 Wing11 (AR=1.0, a=0.0) Drag Coefficient Data with 10-Degree Flap Deflection
Figure 102 Wing12 (AR=1.0, a=0.67) Drag Coefficient Data with 10-Degree Flap Deflection
Figure 103 Wing13 (AR=1.0, a=-1.0) Drag Coefficient Data with 10-Degree Flap Deflection
Figure 106 Wing21 (AR=2.0, a=0.0) Drag Coefficient Data with 10-Degree Flap Deflection
Figure 107  Wing 24 (AR=2.0, a=0.67) Drag Coefficient Data with 10-Degree Flap Deflection
Figure 108 Wing23 (AR=2.0, a=-1.0) Drag Coefficient Data with 10-Degree Flap Deflection
Figure 109 Wing24 (AR=2.0, a=-1.5) Drag Coefficient Data with 10-Degree Flap Deflection
Figure 110 Wing25 (AR=2.0, a=∞) Drag Coefficient Data with 10-Degree Flap Deflection
DRAG COEFFICIENT
AR=3.0 WING32
WIND TUNNEL DATA
FLAP DEFLECTION = 10 DEGREES

Figure 112  Wing32 (AR=3.0, a=-0.67) Drag Coefficient Data with 10-Degree Flap Deflection
Figure 113 Wing33 (AR=3.0, 30°) Drag Coefficient Data with 10-Degree Flap Deflection

DRAG COEFFICIENT

AR=3.0 WING33 WIND TUNNEL DATA
FLAP DELETION = 10 DEGREES

M=1.25

0.200 0.160 0.120 0.080 0.040 0.000
C_L 0.02 0.04 0.06 0.08 0.10 0.12
C_D

M=0.90

M=0.75
DRAG COEFFICIENT

AR=3.0 WING34
WIND TUNNEL DATA
FLAP DEFLECTION = 10 DEGREES

Figure 114: Wing34 (AR=3.0, a=1.5) Drag Coefficient Data with 10-Degree Flap Deflection
Figure 115 Wing35 (AR=3.0, a=∞) Drag Coefficient Data with 10-Degree Flap Deflection
Figure 116 1.0 Aspect Ratio / Delta Lift Coefficient Comparison

Delta Lift Coefficient per Flap Deflection

Wind Tunnel Data

Aspect Ratio = 1.0

SYM Angle of Attack

0 degrees
1 degrees
5 degrees

M = 1.25
M = 0.90
M = 0.75

Delta Cl (per degree)
DELTA LIFT COEFFICIENT PER FLAP DEFLECTION

WIND TUNNEL DATA
ASPECT RATIO = 2.0

<table>
<thead>
<tr>
<th>SYM</th>
<th>Angle of Attack</th>
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</thead>
<tbody>
<tr>
<td></td>
<td>0 degrees</td>
</tr>
<tr>
<td></td>
<td>1 degrees</td>
</tr>
<tr>
<td></td>
<td>5 degrees</td>
</tr>
</tbody>
</table>

Figure 117 2.0 Aspect Ratio / Delta Lift Coefficient Comparison
DELTA LIFT COEFFICIENT PER FLAP DEFLECTION

Figure 118. 3.0 Aspect Ratio / Delta Lift Coefficient Comparison

<table>
<thead>
<tr>
<th>M</th>
<th>1.25</th>
<th>0.90</th>
<th>0.75</th>
</tr>
</thead>
<tbody>
<tr>
<td>δ</td>
<td>0°</td>
<td>5°</td>
<td>10°</td>
</tr>
<tr>
<td>δ (Cutout Factor)</td>
<td>-INF</td>
<td>0</td>
<td>0</td>
</tr>
<tr>
<td>δ (Cutout Factor)</td>
<td>-INF</td>
<td>0</td>
<td>0</td>
</tr>
</tbody>
</table>
delta wings at 1.25 Mach number. The maximum $C_{L8}$ for these wings generally occurred at cutout factors of -1.0 to -1.5.

Delta pitching moment coefficients per degree flap deflection are calculated as described for the lift coefficients. For this study, $C_{M8}$ per degree was computed referenced to the respective aerodynamic center location of each wing. The delta pitching moment coefficients per degree are plotted with respect to cutout factor in Figures 119, 120, and 121 for aspect ratios 1.0, 2.0, and 3.0, respectively.

These data consistently show a decrease in $C_{M8}$ as cutout factor is decreased at all three aspect ratios and Mach numbers. In all cases the delta wing produced the largest $C_{M8}$. This result is primarily due to two causes. First, the trailing-edge flap is more effective on the delta wings as seen in the subsonic $C_{L8}$ results. Second, the center of pressure of the flap lift moves forward with decreasing cutout factor. The more negative cutout factors contain high negative (forward-swept) trailing-edge sweeps which sweeps the flap forward. The center of pressure of the flap lift moves farther forward than aerodynamic center moves as cutout factor decreases.

**Comparison to Linear Theory Predictions**

The Carmichael lifting surface method\textsuperscript{7} was employed to predict trailing-edge flap effects. Comparisons of current test $C_{L8}$ data to predictions are presented in Figures 122, 123, and 124 for aspect ratios 1.0, 2.0, and 3.0, respectively. Comparisons are given at an angle of attack of 5 degrees; the comparisons are similar at other angles.

In all cases the predicted $C_{L8}$ was significantly higher than test results. Linear theory does not account for vortex flow due to sharp leading-edge separation or separated flow which may occur on the deflected flap. Thus, it is not unreasonable
DELTA CM PER FLAP DEFLECTION

WIND TUNNEL DATA
ASPECT RATIO = 1.0
Moment Reference = a.c. location of respective wing

<table>
<thead>
<tr>
<th>SYM</th>
<th>Angle of Attack</th>
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<tbody>
<tr>
<td>O</td>
<td>0 degrees</td>
</tr>
<tr>
<td>----</td>
<td>1 degrees</td>
</tr>
<tr>
<td>------</td>
<td>5 degrees</td>
</tr>
</tbody>
</table>

Figure 119 1.0 Aspect Ratio / Delta Pitching Moment Coefficient Comparison
DELTA CM PER FLAP DEFLECTION

WIND TUNNEL DATA
ASPECT RATIO = 2.0
Moment Reference = a.c. location of respective wing

<table>
<thead>
<tr>
<th>SYM</th>
<th>Angle of Attack</th>
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<tbody>
<tr>
<td></td>
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<tr>
<td></td>
<td>1 degrees</td>
</tr>
<tr>
<td></td>
<td>5 degrees</td>
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</table>

Figure 120 2.0 Aspect Ratio / Delta Pitching Moment Coefficient Comparison
Figure 121 3.0 Aspect Ratio Delta Pitching Moment Coefficient Comparison
DELTA LIFT COEFFICIENT TEST-TO-THEORY COMPARISON

ASPECT RATIO = 1.0
Angle of Attack = 5 degrees

<table>
<thead>
<tr>
<th>SYM</th>
<th>Source</th>
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<tbody>
<tr>
<td></td>
<td>Wind Tunnel Data</td>
</tr>
<tr>
<td></td>
<td>Linear Theory Prediction</td>
</tr>
</tbody>
</table>

Figure 122 1.0 Aspect Ratio / Delta Lift Coefficient Test-To-Theory Comparison

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DELTA LIFT COEFFICIENT TEST-TO-THEORY COMPARISON

ASPECT RATIO = 2.0
Angle of Attack = 5 degrees

<table>
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<th>SYM</th>
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<tbody>
<tr>
<td>Wind Tunnel Data</td>
<td>Linear Theory Prediction</td>
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</table>

Figure 123 2.0 Aspect Ratio / Delta Lift Coefficient Test-To-Theory Comparison
DELTA LIFT COEFFICIENT TEST-TO-THEORY COMPARISON

ASPECT RATIO = 3.0
Angle of Attack = 5 degrees

<table>
<thead>
<tr>
<th>SYM</th>
<th>Source</th>
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<tbody>
<tr>
<td></td>
<td>Wind Tunnel Data</td>
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<tr>
<td></td>
<td>Linear Theory Prediction</td>
</tr>
</tbody>
</table>

Figure 124 3.0 Aspect Ratio / Delta Lift Coefficient Test-To-Theory Comparison
for the predicted magnitudes to be higher. However, the trends of $C_{L5}$ with respect to cutout factor compare extremely well.

Comparisons of current test $C_{M5}$ data to linear theory predictions are presented in Figures 125, 126, and 127. These comparisons are also shown at 5-degrees angle of attack. The $C_{M5}$ predictions are significantly greater than test results. As described above, this is due to flow characteristics in the test that are not properly modeled by linear theory. However, linear theory adequately predicts the $C_{M5}$ trends with respect to cutout factor at subsonic speeds for all three aspect ratios.

At 1.25 Mach number the predicted trend compares well with test results for 1.0 aspect ratio. But for aspect ratio 2.0 and 3.0 the predicted trends compare poorly. The test data yield an increase in $C_{M5}$ as cutout factor becomes more negative; the predictions show a minimum $C_{M5}$ between -0.67 and -1.0 cutout factor. This result is probably due to supersonic flow characteristics or flap lift center of pressure location that is not modeled correctly by linear theory.
DELTA CM TEST-TO-THEORY COMPARISON

ASPECT RATIO = 1.0
Angle of Attack = 5 degrees

<table>
<thead>
<tr>
<th>SYM</th>
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<tr>
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<td>Wind Tunnel Data</td>
</tr>
<tr>
<td></td>
<td>Linear Theory Prediction</td>
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Figure 125 1.0 Aspect Ratio / Delta Pitching Moment Coefficient Test-To-Theory Comparison
DELTA CM TEST-TO-THEORY COMPARISON

ASPECT RATIO = 3.0
Angle of Attack = 5 degrees

<table>
<thead>
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<tbody>
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<td>0</td>
<td>Wind Tunnel Data</td>
</tr>
<tr>
<td></td>
<td>Linear Theory Prediction</td>
</tr>
</tbody>
</table>

M=0.75

M=0.90

M=1.25

Figure 127 3.0 Aspect Ratio / Delta Pitching Moment Coefficient Test-To-Theory Comparison
CONCLUSIONS AND RECOMMENDATIONS

A wind tunnel test was conducted at Mach numbers 0.75, 0.90, and 1.25 to investigate the longitudinal aerodynamics, primarily lift curve slope, of sharp-leading-edge diamond and delta wings. A trailing-edge flap deflectable to 10 degrees was tested to obtain flap effectiveness data. Linear theory predictions were made to compare to test results.

The diamond wings produced higher lift-curve slopes than delta wings at all conditions except one. This result corresponds to previous test data on delta and diamond wings. The delta wings of aspect ratio 2.0 and 3.0 produced lift-curve slopes lower than those from previous tests. This is primarily due to inaccuracies in angle-of-attack setting and balance normal force.

The magnitudes of the test lift-curve slopes did not compare well with linear theory predictions at 2.0 and 3.0 aspect ratio. The slope trends with respect to cutout factor compared closely between test data and predictions.

Aerodynamic center location moved forward as cutout factor varied from zero to larger negative values.

The drag levels were similar between cutout factors at subsonic speeds. At Mach number 1.25, the diamond wings have the lowest drag level for 1.0 aspect ratio. The delta wing produces the lowest drag level for 2.0 and 3.0 aspect ratios.

The trailing-edge flap is more effective for the delta wings at subsonic speeds. At Mach number 1.25, $C_{L5}$ is greater for the diamond wings but $C_{M5}$ is larger for the delta wings.
Several recommendations are suggested to improve the accuracy of the data and better understand the results.

1) Provide a more accurate method of setting and measuring angle of attack including deflection due to loads.
2) Increase model loads to better match balance normal force capacity through higher Reynolds number testing, higher angle of attack, or larger models.
3) Provide for indication of grounding between the model-to-balance end-plate adapter and the test section wall.
4) Decrease balance capacity to increase accuracy at lower loadings.
5) Test the wings at more angles of attack.
6) Fabricate and test additional flap deflections.
7) Conduct flow visualization testing.
APPENDIX

DATA UNCERTAINTY ANALYSIS
APPENDIX

Data Uncertainty Analysis

Based on the balance calibration, test results, repeat runs, and data analysis, large uncertainties are present in the test data. This appendix will quantify and assess the uncertainties in the measurements and calculations involved in the lift-curve slope analysis.

The uncertainties in the lift-curve slope will be determined using the method of Kline and McClintock. According to this method, the uncertainty of a variable $y=f(x_1,x_2,x_3,...)$ is

$$\sigma_y = [(\sigma_{x_1} \cdot \partial y / \partial x_1)^2 + (\sigma_{x_2} \cdot \partial y / \partial x_2)^2 + (\sigma_{x_3} \cdot \partial y / \partial x_3)^2 + ...]^{0.5}.$$ 

The lift-curve slope,

$$C_{L_\alpha} = \Delta C_L / \Delta \alpha = (C_{L_2} - C_{L_1}) / (\alpha_2 - \alpha_1)$$

$$= \left[ (NF_2 \cdot \cos \alpha_2 - AF_2 \cdot \sin \alpha_2) - (NF_1 \cdot \cos \alpha_1 - AF_1 \cdot \sin \alpha_1) \right] / \left[ q \cdot S^*(\alpha_2 - \alpha_1) \right]$$

$$= f(\alpha_1, \alpha_2, q, NF_1, NF_2, AF_1, AF_2).$$

Let $Kn = (NF_2 \cdot \cos \alpha_2 - AF_2 \cdot \sin \alpha_2) - (NF_1 \cdot \cos \alpha_1 - AF_1 \cdot \sin \alpha_1)$

and $Kd = q \cdot S^*(\alpha_2 - \alpha_1),$

then $C_{L_\alpha} = Kn / Kd.$
The uncertainty in $C_{L\alpha}$ is

$$ \sigma_{C_{L\alpha}} = [(\sigma_{\alpha_1} \cdot \partial C_{L\alpha} / \partial \alpha_1)^2 + (\sigma_{\alpha_2} \cdot \partial C_{L\alpha} / \partial \alpha_2)^2 + (\sigma_q \cdot \partial C_{L\alpha} / \partial q)^2 \\
+ (\sigma_{NF_1} \cdot \partial C_{L\alpha} / \partial NF_1)^2 + (\sigma_{NF_2} \cdot \partial C_{L\alpha} / \partial NF_2)^2 \\
+ (\sigma_{AF_1} \cdot \partial C_{L\alpha} / \partial AF_1)^2 + (\sigma_{AF_2} \cdot \partial C_{L\alpha} / \partial AF_2)^2 ]^{0.5}. $$

The partial derivatives are

$$ \partial C_{L\alpha} / \partial \alpha_1 = [Kn / (\alpha_2 - \alpha_1) + (NF_1 \cdot \sin \alpha_1 + AF_1 \cdot \cos \alpha_1)] / Kd $$

$$ \partial C_{L\alpha} / \partial \alpha_2 = [-Kn / (\alpha_2 - \alpha_1) - (NF_2 \cdot \sin \alpha_2 + AF_2 \cdot \cos \alpha_2)] / Kd $$

$$ \partial C_{L\alpha} / \partial q = -Kn / (q \cdot Kd) $$

$$ \partial C_{L\alpha} / \partial NF_1 = -\cos \alpha_1 / Kd $$

$$ \partial C_{L\alpha} / \partial NF_2 = \cos \alpha_2 / Kd $$

$$ \partial C_{L\alpha} / \partial AF_1 = \sin \alpha_1 / Kd $$

$$ \partial C_{L\alpha} / \partial AF_2 = -\sin \alpha_2 / Kd. $$

The uncertainties of the pertinent variables from the test are

$$ \sigma_q = 1.0 \text{ psi} $$

$$ \sigma_{\alpha_1} = \sigma_{\alpha_2} = 0.00873 \text{ radians (0.5 degrees)} $$

$$ \sigma_{NF_1} = \sigma_{NF_2} = 3.50 \text{ lbs} $$

$$ \sigma_{AF_1} = \sigma_{AF_2} = 0.375 \text{ lbs}. $$

Two representative cases will be evaluated; (1) Wing21 (AR=2.0, $a=0$) at 1.25 Mach number and (2) Wing31 (AR=3.0, $a=0$) at 0.90 Mach number.

For Case 1, the values of the pertinent variables are

$$ S = 10.656 \text{ in}^2 \quad , \quad q = 30.96 \text{ psi} $$

$$ \alpha_1 = 0.0 \text{ radians} \quad , \quad \alpha_2 = 0.08727 \text{ radians (5 degrees)} $$

$$ NF_1 = 0.93 \text{ lbs} \quad , \quad NF_2 = 57.51 \text{ lbs} $$
\[ AF_1 = 8.34 \text{ lbs}, \quad AF_2 = 8.72 \text{ lbs}. \]
\[ \sigma_{CL_\alpha} = [(0.1957)^2 + (-0.1973)^2 + (0.0624)^2 + (-0.1216)^2 + (0.1211)^2 \]
\[ + (0)^2 + (-0.0011)^2]^{0.5}. \]

The uncertainty in the lift-curve slope, \( \sigma_{CL_\alpha} = 0.333 \) per radian. The test lift-curve slope for this case, \( CL_\alpha = 1.94 \) per radian. Therefore, the uncertainty is 17.2\% relative to the test lift-curve slope.

For Case 2, the values of the pertinent variables are
\[ S = 10.656 \text{ in}^2, \quad q = 24.59 \text{ psi} \]
\[ \alpha_1 = 0.0 \text{ radians}, \quad \alpha_2 = 0.08727 \text{ radians (5 degrees)} \]
\[ NF_1 = -0.94 \text{ lbs}, \quad NF_2 = 65.30 \text{ lbs} \]
\[ AF_1 = 7.83 \text{ lbs}, \quad AF_2 = 5.50 \text{ lbs}. \]
\[ \sigma_{CL_\alpha} = [(0.2894)^2 + (-0.2907)^2 + (-0.1165)^2 + (-0.1531)^2 + (0.1525)^2 \]
\[ + (0)^2 + (-0.0014)^2]^{0.5}. \]

The uncertainty in the lift-curve slope, \( \sigma_{CL_\alpha} = 0.478 \) per radian. The test lift-curve slope for this case, \( CL_\alpha = 2.92 \) per radian. Therefore, the uncertainty is 16.4\% relative to the test lift-curve slope.

These results indicate that the angle of attack and normal force uncertainties dominate the uncertainty in the lift-curve slope. The uncertainty results for other conditions are similar to the cases presented.
REFERENCES


3 Ludwig, H., "Tube Wind Tunnel--a Special Type of Blowdown Tunnel."
   Paper presented at the eleventh meeting of the AGARD Wind Tunnel and Model Testing Panel, Scheveningen, Holland, July 1957.


5 Whitfield, Jack D., Schueler, C.J. and Starr, Roger F., "High Reynolds Number Transonic Wind Tunnels - Blowdown or Ludwig Tube ?;"

6 Kulakowski, L.J., "The Lift and Drag due to Lift of Delta Wing Configurations, Including The Effects of Reynolds Number Variations."
   Consolidated Vultee Report FZA-4-083, July 1953.

