INFORMATION TO USERS

This manuscript has been reproduced from the microfilm master. UMI films the text directly from the original or copy submitted. Thus, some thesis and dissertation copies are in typewriter face, while others may be from any type of computer printer.

The quality of this reproduction is dependent upon the quality of the copy submitted. Broken or indistinct print, colored or poor quality illustrations and photographs, print bleedthrough, substandard margins, and improper alignment can adversely affect reproduction.

In the unlikely event that the author did not send UMI a complete manuscript and there are missing pages, these will be noted. Also, if unauthorized copyright material had to be removed, a note will indicate the deletion.

Oversize materials (e.g., maps, drawings, charts) are reproduced by sectioning the original, beginning at the upper left-hand corner and continuing from left to right in equal sections with small overlaps. Each original is also photographed in one exposure and is included in reduced form at the back of the book.

Photographs included in the original manuscript have been reproduced xerographically in this copy. Higher quality 6" x 9" black and white photographic prints are available for any photographs or illustrations appearing in this copy for an additional charge. Contact UMI directly to order.

UMI
University Microfilms International
A Bell & Howell Information Company
300 North Zeeb Road, Ann Arbor, MI 48106-1346 USA
313/761-4700  800/521-0600
An experimental investigation of upper vortex flaps on a 60° delta wing

Erol, Hamdi, M.S.
The University of Texas at Arlington, 1993
AN EXPERIMENTAL INVESTIGATION OF UPPER VORTEX FLAPS
ON A 60° DELTA WING

The members of the Committee approve the masters
thesis of Hamdi Erol

Donald D. Seath
Supervising Professor

Frank K. Lu

L. F. Heath
AN EXPERIMENTAL INVESTIGATION OF UPPER VORTEX FLAPS
ON A 60° DELTA WING

by
HAMDI EROL

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

THE UNIVERSITY OF TEXAS AT ARLINGTON
May 1993
ACKNOWLEDGEMENTS

I would like to thank my supervising Professor Donald D. Seath very much for his support and patience during the course of my thesis. His critiques were more than welcome and contributed greatly to the end product of my work.

I am also grateful to my grandfather James Clarkson who was very helpful in the making of the test models as well as with his enlightening comments. My appreciation goes further to my neighbor Mr. Buck Rooney a former photographer whose expertise and efforts led to the photographic results.

I also thank my wife, Dilek, very much for her patience and support throughout my studies.

December 9, 1992
ABSTRACT

AN EXPERIMENTAL INVESTIGATION OF UPPER VORTEX FLAPS
ON A 60° DELTA WING

Publication No. ________

Hamdi Erol
The University of Texas at Arlington, 1992

Supervising Professor: Donald D. Seath

Low-speed wind tunnel tests were conducted to investigate the effects of upward deflected constant chord leading-edge flaps on the performance of a 60° delta wing. Smoke flow visualization showed a growing vortex pattern on the upper surfaces with increasing angle of attack and decreasing flap deflections stemming from leading-edge flap separation. Flow reattachment and secondary separation lines could be observed with surface flow visualization.

Upper surface pressure distribution measurements on wing and flaps revealed a significant drop in suction levels over the wing with the least pressure drop occurring on the smallest flap deflection of 120°. Large suction levels on the upper surfaces of the flaps were achieved for all flap deflections tested.

Longitudinal force and moment measurements with a pyramidal balance system indicated that lift and drag were increased significantly by the use of
upward deflected flaps and that the pitching moment about the aerodynamic center showed that the wing was stable for all configurations tested.
TABLE OF CONTENTS

ACKNOWLEDGEMENTS .......................................................................................... iii
ABSTRACT ............................................................................................................. iv
LIST OF FIGURES .............................................................................................. viii
LIST OF TABLES ................................................................................................. xi
LIST OF SYMBOLS ............................................................................................. xii
CHAPTER 1 - INTRODUCTION ......................................................................... 1
CHAPTER 2 - BACKGROUND ........................................................................... 5
CHAPTER 3 - RESEARCH MODEL ................................................................. 8
CHAPTER 4 - WIND TUNNEL FACILITY .................................................. 12
CHAPTER 5 - FLOW VISUALIZATION
   5.1 Flow Visualization ..................................................................................... 17
   5.1.1 Smoke-Wire Flow Visualization ....................................................... 17
   5.1.1.1 Electrical Setup for Heating the Smoke Wire ............................. 18
   5.1.1.2 Smoke Wires .................................................................................. 21
   5.1.1.3 Coating Material and Procedure .............................................. 22
   5.1.1.4 Photographis Technique and Lighting ................................. 22
   5.1.1.5 Test Setup and Conditions ......................................................... 23
   5.1.1.6 Results and Discussion .............................................................. 28
   5.2 Oil-Dot Technique ................................................................................... 42
   5.2.1 Test Setup for the Oil-Dot Technique ............................................ 42
   5.2.2 Results and Discussion ..................................................................... 43
CHAPTER 6 - PRESSURE DISTRIBUTION MEASUREMENTS
   6.1 Test Setup and Conditions ................................................................. 57
   6.2 Results and Discussion ......................................................................... 58
CHAPTER 7 - FORCE BALANCE MEASUREMENTS

7.1 Test Setup and Conditions ...................................... 68
7.2 Results and Discussion ........................................... 69

CHAPTER 8 - CONCLUSIONS AND RECOMMENDATIONS ...... 79

REFERENCES ........................................................................ 81
LIST OF FIGURES

Figure 1: Cross-sectional view of vortex flap types............................................... 3
Figure 2: Vortex pattern caused by upper vortex flap (UVF) in a plane
normal to the leading edge...................................................................................... 6
Figure 3: Geometry of model for pressure measurements........................................ 8
Figure 4: Starboard side flap box used for pressure measurements...................... 9
Figure 5: Wing box used for pressure measurements............................................. 10
Figure 6: Model geometry for flow visualization and force measurements............. 11
Figure 7: UTA 30 x 30 low-speed wind tunnel..................................................... 12
Figure 8: Dynamic pressure calibration curve for UTA 30-inch by
30-inch low-speed wind tunnel.............................................................................. 14
Figure 9: Sketch of closed return wind tunnel....................................................... 15
Figure 10: Dynamic pressure calibration curve for UTA 24-inch
by 36-inch low-speed wind tunnel........................................................................ 16
Figure 11: Electrical circuit for resistive heating of the smoke wire...................... 18
Figure 12: Frame used to integrate smoke wire into the test section.................... 19
Figure 13: Test setup for smoke-wire flow visualization in UTA's
30" x 30" low-speed wind tunnel........................................................................... 20
Figure 14: Photographic results of the smoke-wire flow visualization
for an angle of attack of 0° for the 135° UVF wing covering
a time span of 0.267 seconds. \( v_\infty = 13.5 \) ft/s, \( Re_w = 46.5 \).................. 24
Figure 15: Plane in which the smoke flow acts on test model................................ 27
Figure 16: Photographic results of smoke-wire flow visualization
for planar delta wing without flaps at angles of attack
from 0° to 20°. \( v_\infty = 13.5 \) ft/s, \( Re_w = 46.5 \)................................................... 30
Figure 17: Photographic results of smoke-wire flow visualization for the delta wing with 150° UVF at angles of attack from 0° to 20°. \( v_\infty = 13.5 \text{ ft/s}, \, Re_w = 46.5 \).

Figure 18: Photographic results of smoke-wire flow visualization for the delta wing with 135° UVF at angles of attack from 0° to 20°. \( v_\infty = 13.5 \text{ ft/s}, \, Re_w = 46.5 \).

Figure 19: Photographic results of smoke-wire flow visualization for the delta wing with 120° UVF at angles of attack from 0° to 20°. \( v_\infty = 13.5 \text{ ft/s}, \, Re_w = 46.5 \).

Figure 20: Photographic results of the oil-dot technique for the delta wing with 165° UVF at angles of attack from 0° to 20°. \( v_\infty = 87.7 \text{ ft/s}, \, Re_{MAC} = 323,000 \).

Figure 21: Photographic results of the oil-dot technique for the delta wing with 150° UVF at angles of attack from 0° to 20°. \( v_\infty = 87.7 \text{ ft/s}, \, Re_{MAC} = 323,000 \).

Figure 22: Photographic results of the oil-dot technique for the delta wing with 135° UVF at angles of attack from 0° to 20°. \( v_\infty = 87.7 \text{ ft/s}, \, Re_{MAC} = 323,000 \).

Figure 23: Photographic results of the oil-dot technique for the delta wing with 120° UVF at angles of attack from 0° to 10°. \( v_\infty = 87.7 \text{ ft/s}, \, Re_{MAC} = 323,000 \).

Figure 24: An interpretation of the flow about the 60° delta wing with 165° UVF at 5° angle of attack.

Figure 25: Spanwise pressure distribution on 60° planar delta wing at 64.2 percent of the root chord.
Figure 26: Spanwise pressure distribution on a 60° delta wing with
165° UVF at 64.2 percent of the root chord.............................. 61
Figure 27: Spanwise pressure distribution on a 60° delta wing with
150° UVF at 64.2 percent of the root chord.............................. 62
Figure 28: Spanwise pressure distribution on a 60° delta wing with
135° UVF at 64.2 percent of the root chord.............................. 63
Figure 29: Chordwise pressure distribution on a 165° upwards
deflected flap at a distance of 29.2 percent of flap span
from the wing apex............................................................... 64
Figure 30: Chordwise pressure distribution on a 150° upwards
deflected flap at a distance of 29.2 percent of flap span
from the wing apex............................................................... 65
Figure 31: Chordwise pressure distribution on a 135° upwards
deflected flap at a distance of 29.2 percent of flap span
from the wing apex............................................................... 66
Figure 32: Effect of vortex flap deflection on the location of the
aerodynamic center.............................................................. 73
Figure 33: Lift coefficient versus angle of attack with flap upwards
deflections as parameter....................................................... 74
Figure 34: Effect of UVF deflections on drag polar........................ 75
Figure 35: Effect of UVF deflections on the lift-to-drag ratio............. 76
Figure 36: Effect of UVF deflections on the pitching moment referenced
to the aerodynamic center of each flap configuration............... 77
LIST OF TABLES

Table 1: Test matrix for pressure model configuration ......................... 58
Table 2: Test matrix for force balance measurement model ................... 71
**LIST OF SYMBOLS**

<table>
<thead>
<tr>
<th>Symbol</th>
<th>Definition</th>
</tr>
</thead>
<tbody>
<tr>
<td>AR</td>
<td>Aspect ratio ( (b^2/S) )</td>
</tr>
<tr>
<td>a.c.</td>
<td>Aerodynamic center location ( (x_{ac}/MAC) )</td>
</tr>
<tr>
<td>b</td>
<td>Half-span of wing ( \text{(in.)} )</td>
</tr>
<tr>
<td>( c_D )</td>
<td>Measured drag coefficient</td>
</tr>
<tr>
<td>( c_{D0} )</td>
<td>Profile drag coefficient</td>
</tr>
<tr>
<td>( c_{D_s} )</td>
<td>Separation drag coefficient</td>
</tr>
<tr>
<td>( c_L )</td>
<td>Measured lift coefficient</td>
</tr>
<tr>
<td>( c_{MAC} )</td>
<td>Pitching moment coefficient about the aerodynamic center</td>
</tr>
<tr>
<td>( c_{mtr} )</td>
<td>Measured pitching moment coefficient about the balance moment center</td>
</tr>
<tr>
<td>( c_p )</td>
<td>Pressure coefficient</td>
</tr>
<tr>
<td>( c_r )</td>
<td>Root chord of the 60° delta wing ( \text{(in.)} )</td>
</tr>
<tr>
<td>( \Delta h )</td>
<td>Difference in manometer fluid heights ( \text{(in.)} )</td>
</tr>
<tr>
<td>d</td>
<td>Wire diameter ( \text{(mm)} )</td>
</tr>
<tr>
<td>D</td>
<td>Measured drag force ( \text{(lb)} )</td>
</tr>
<tr>
<td>( D_{str} )</td>
<td>Drag force caused by the strut ( \text{(lb)} )</td>
</tr>
<tr>
<td>( g )</td>
<td>Gravity constant ( (32.2 \text{ ft/s}^2) )</td>
</tr>
<tr>
<td>k</td>
<td>Wake blockage ratio.</td>
</tr>
<tr>
<td>L</td>
<td>Measured lift force ( \text{(lb)} )</td>
</tr>
<tr>
<td>( L_{str} )</td>
<td>Lift force caused by the balance strut ( \text{(lb)} )</td>
</tr>
<tr>
<td>MAC</td>
<td>Mean aerodynamic chord (aka mean geometric chord) ( \text{(in.)} )</td>
</tr>
<tr>
<td>( M_{str} )</td>
<td>Pitching moment caused by the balance strut ( \text{(in.-lb)} )</td>
</tr>
<tr>
<td>p</td>
<td>Local static pressures ( \text{(lb/ft}^2) )</td>
</tr>
<tr>
<td>( p_{\infty} )</td>
<td>Free-stream static pressure ( \text{(lb/ft}^2) )</td>
</tr>
<tr>
<td>Symbol</td>
<td>Definition</td>
</tr>
<tr>
<td>---------</td>
<td>-----------------------------------------------------------------------------</td>
</tr>
<tr>
<td>$p_{amb}$</td>
<td>Ambient pressure (lb/ft$^2$)</td>
</tr>
<tr>
<td>$q$</td>
<td>Dynamic pressure (lb/ft$^2$)</td>
</tr>
<tr>
<td>$q_{\infty}$</td>
<td>Free-stream dynamic pressure (lb/ft$^2$)</td>
</tr>
<tr>
<td>$q_c$</td>
<td>Corrected dynamic pressure (lb/ft$^2$)</td>
</tr>
<tr>
<td>$q_u$</td>
<td>Uncorrected dynamic pressure (lb/ft$^2$)</td>
</tr>
<tr>
<td>$Re_w$</td>
<td>Reynolds number based on wire diameter</td>
</tr>
<tr>
<td>$S$</td>
<td>Projected surface area of model (ft$^2$)</td>
</tr>
<tr>
<td>$S/C$</td>
<td>Blockage ratio (model surface area divided by wind tunnel test section cross section area)</td>
</tr>
<tr>
<td>$t$</td>
<td>Thickness of pressure model both of wing and flaps (in.)</td>
</tr>
<tr>
<td>$t_r$</td>
<td>Balance moment center location ($x_r$/MAC)</td>
</tr>
<tr>
<td>$t_w$</td>
<td>Thickness of the delta wing used as force and flow visualization model (in.)</td>
</tr>
<tr>
<td>UVF</td>
<td>Upper vortex flap</td>
</tr>
<tr>
<td>$x_{ac}$</td>
<td>Distance of aerodynamic center from wing leading edge</td>
</tr>
<tr>
<td>$x_{tr}$</td>
<td>Distance of balance moment center from wing leading edge</td>
</tr>
<tr>
<td>$v_{\infty}$</td>
<td>Free-stream velocity (ft/s)</td>
</tr>
<tr>
<td>$\alpha$</td>
<td>Angle of attack of wing (°)</td>
</tr>
<tr>
<td>$\alpha_l$</td>
<td>Local angle of attack between the free-stream velocity vector and the upper vortex flap chord (°)</td>
</tr>
<tr>
<td>$\beta$</td>
<td>Manometer board slant angle (°)</td>
</tr>
<tr>
<td>$\delta_{UVF}$</td>
<td>Flap deflection angle measured perpendicular to leading edge (°)</td>
</tr>
<tr>
<td>$\epsilon$</td>
<td>Angle between wing chord and flap leading edge in the flow plane (°)</td>
</tr>
<tr>
<td>$\gamma$</td>
<td>Specific gravity of manometer fluid</td>
</tr>
</tbody>
</table>
CHAPTER 1
INTRODUCTION

Aircraft designed for supersonic flight performance are usually fitted with thin, highly swept delta wings (also categorized as slender wings which is a term used for thin low aspect ratio wings) to reduce supersonic wave drag. At high subsonic speeds these wings also provide the advantages of high drag divergence Mach number. High-g maneuverability, due to increased lift generated by stable leading-edge vortices, is also possible. These vortices, stemming from leading-edge flow separation and subsequent flow reattachment on the upper wing surface, enhances aerodynamic performance at high angles of attack.

However, the aerodynamic behavior of highly swept delta wings is considerably different from conventional wings at subsonic speeds, in a way that the wing exhibits poor performance at low subsonic speeds. At low-to-moderate angles of attack, these wings experience a significant drop in lift compared to conventional wings, which has to do with the absence of a strong leading-edge vortex. More deficiencies of highly swept delta wings are confronted at high angles of attack in the low-speed flight regime arising from leading-edge separation, which can result in severe drag penalties, longitudinal instabilities, and lateral control deficiencies (Ref. 1).

One method of alleviating drag and stability problems at low subsonic speeds is through the control of flow attachment. Through flow attachment a leading-edge suction force results from high local velocities and the
accompanying low pressure region, which occur when air flows from the stagnation point on the lower surface of the wing around the leading edge to the upper surface. Maintaining leading-edge suction provides a drag reduction through the thrust component in this suction force. But to achieve flow attachment a leading-edge camber is required. A fixed cambered leading edge provides flow attachment up to high angles of attack, but causes severe drag penalties at supersonic cruise speed and is quite impractical in the combat environment for fighter type aircraft where it is difficult to maintain attached flow at high load factor maneuvers (Ref. 2). Variable leading-edge camber in comparison can achieve the required performance levels but adds an unacceptable level of weight and mechanical complexity which makes it undesirable.

On the other hand leading-edge separation and the consequent leading-edge vortex formation is an important part of the lift produced on highly swept slender wings and therefore should be promoted rather than eliminated. Thus, an alternative to the conventional approach of maintaining attached flow and the subsequent leading-edge suction for drag reduction is the use of leading-edge vortex flaps (Ref.2) deflected downwards from the wing leading edge and conforming to the wing surface when retracted (see Fig.1.a). As the name of the 'vortex flap' implies, the flow is forced to separate at the flap leading edge which produces a spiral vortex with its reattachment ideally at the wing-flap hinge line for maximum efficiency. The vortex generated by the downward deflection of this flap as depicted in Fig.1.a. results in a forward-tilted suction force. The components of this suction force parallel with and perpendicular to the freestream flow direction are lift and thrust component, respectively. Of course the leading-edge flap concept is also going to introduce increased weight and
mechanical complexity as in the case of attached flow variable camber design. But it is believed that the advantages of the vortex flap overcome these deficiencies.

![Diagram showing vortex flap and retraction positions]

- **Vortex flap**: downward deflection gives thrust component for drag reduction.
- **S**: Separation
- **R**: Reattachment
- **$v_\infty$**: Free-stream velocity (fps)

- **Upward deflection causes vortex suction contributing to lift and drag**

Figure 1: Cross-sectional view of:
(a) - Conventional-type vortex flap.
(b) - Upwards deflected vortex flap.

The vortex flap in its conventional form as mentioned above gained some popularity as a research subject in the 1980's (References 1,2,4-8). In its variations, the possible usage of it with deflection upward rather than downward as illustrated in Fig.1.b. have been proposed (Refs. 8-9). These 'upper vortex flaps' (UVF) deflect upward relative to the wing chord line and cause flow separation at the flap leading edge as in the conventional vortex flap type, only
at much lower angles of attack, because the incoming flow sees the UVF at a higher incidence than the wing itself. Consequently, the resulting vortex induces a suction force on the upper surface of the flap which is tilted backward thus increasing drag and lift.
CHAPTER 2
BACKGROUND

Conventional type vortex flaps as described in Chapter 1 have been the focus of research in improving the subsonic aerodynamic characteristics of highly swept slender wings in the past 12 years (Refs.1,2,5-7). This area has been extensively explored by NASA's Langley Research Center (Refs.1,2,5,6). Figure 1a shows the principal concept behind the conventional vortex flap. However, there are relatively few investigations on the concept of an upwards deflected vortex flap (UVF). It was studied by Marchman (Ref. 8) who used 150° 'inverted' (referring to upward deflections of flaps) leading-edge vortex flaps on a 60° delta wing. The 150° deflection is measured from the wing chord on the upper surface of the wing in the counterclockwise direction (see Fig.3). He determined the lift and drag coefficients with varying angles of attack for his model and concluded that there is a significant increase in lift as well as in drag.

Tingas (Ref.4) has likewise investigated the effect of upward deflected vortex flaps on the aerodynamic characteristics by employing a 150° flap and essentially came up with the same conclusions as Marchman (Ref 8).

Later Rao (Ref.9) studied the low-speed performance of upper vortex flaps in a range from 30° to 70° flap deflection angle on a 74° swept flat plate delta wing. According to his findings, an inboard vortex forms at low to moderate angles of attack (Fig. 2a) associated with the free-stream attachment on the flap surface and an additional leading edge vortex at high angle of attack when the free-stream attachment moves to the lower wing surface (Fig. 2b). This particular UVF
Figure 2: Vortex pattern caused by upper vortex flap (UVF) in a plane normal to the leading-edge. (a) & (b) are from Ref. 9.

configuration leads to increases for $c_L$ at low angles of attack, a drag increase beneficial for the landing and approach phase with improved longitudinal stability by eliminating pitch up.

One of the latest contributions in this area was done by Londenberg (Ref. 3) who experimentally investigated the performance of a 60° delta wing equipped with UVF deflections of 120°, 135°, and 150°. Trailing-edge flaps were used to
trim his configuration. His conclusions were that the aerodynamic center moves forward and an increase in lift and drag were achieved as UVF span and chord increase, and finally that the efficiency of the flaps increases for trimmed conditions as UVF deflection increases.

The author notes that the effect of upwards deflected vortex flaps (from $120^\circ$ and up) on the upper surface flow have not been well investigated. Thus, flow visualization should give a better understanding of the complicated flow structure caused by the UVF/wing configuration.

The smoke-wire technique quite often used in the past, has been found to be adequate for the visualization of the flow around a delta wing with leading-edge extensions because of two reasons. For one it is very economical and on the other hand it can give high resolution pictures, indicative of the flow structure, even though the present flow structure is purely three dimensional.

Because a better understanding of the flow phenomena associated with upper vortex flaps on delta wings was desired, an experimental investigation was conducted to:

1) Measure pressure distributions restricted to the upper surfaces of the wing and UVF.
2) Measure lift and drag forces and the pitching moment.
3) Visualize flow with smoke lines.
4) Visualize surface flow with oil dot visualization.

The present research investigates full-span upward deflected flaps with force balance measurements and pressure distribution measurements as well as with flow visualization techniques to determine the behavior of the vortices generated by such configurations and their effect on the longitudinal aerodynamic forces.
CHAPTER 3
RESEARCH MODEL

For the current experimental investigation two different wind tunnel models were used; a pressure model and a smaller model for flow visualization and force measurements.

\[ C_T = 12.99 \text{ in}^2 \]
\[ S = 97.43 \text{ in}^2 \]
\[ AR = 2.309 \]
\[ t = 0.3925 \text{ in} \]
\[ MAC = 8.66 \text{ in} \]

Figure 3: Geometry of model for pressure measurements.

The pressure model (see Figs.3-5) was a flat plate, delta wing with full-span, 2-inch-wide flaps and was constructed out of 0.040-inch thick aluminum sheet metal for the wing and flap skins and 5/16-in.-square cross-section aluminum bar for the frame. Upper surface flap skins were bent to the desired deflection angles of 135°, 150°, 165° and fastened to the flap box frame (see Fig.4). Likewise the wing upper surface was attached to the wing box frame (see Fig. 5). Lower surface flap and wing skins were riveted to flap and wing frames,
respectively. Connection between wing and flaps is provided by screws. It should be noted in Fig.3 that at each of the wing-flap junctions a gap appears on the lower surface which starts at the wing apex and ends at the wing tip due to the way the model was constructed. Hence, to avoid air from being trapped in this area during testing, this gap was sealed with a tape. Twenty-four pressure orifices along the upper surface of half of the wing span on the right hand side at 64.2% of the root chord from the wing apex and fifteen pressure orifices on the upper surface of the starboard flaps along the flow direction at 29.2% from the apex were placed on the pressure model. The hollow inside of the wing and flaps allows for the pressure orifice tubing to be placed inside the model and the screw-on type connection makes it easy to repair tubes and pressure orifices that might malfunction while testing.

Figure 4: Starboard side flap box used for pressure measurements.
Figure 5: Wing box used for pressure measurements.

The second (shown in Fig.6) model, used for the flow visualization and force measurements was previously used in reference 3 to investigate 120°, 135°, and 150° full span flap effects on a 60° flat plate delta wing. An additional flap was constructed for a flap deflection of 165°. This model was constructed out of 0.250-inch thick aluminum sheet metal for the planar delta wing and 0.040-inch thick aluminum sheet metal for the flaps.
Figure 6: Model geometry for flow visualization and force measurements.

An important factor in wind tunnel testing is that the maximum frontal area of the model during testing does not exceed 7.5% of the test section cross-section area (Ref.10). This so called wake blockage ratio $k$ for the present models is:

- Pressure model: $k = 4.94\%$
- Force model: $k = 3.51\%$
CHAPTER 4
WIND TUNNEL FACILITY

For smoke-wire flow visualization and pressure measurements, UTA's 30-in x 30-in open-circuit low-speed wind tunnel capable of speeds up to 100 ft/s was used. The air is drawn by an electrically-powered fan into the inlet fitted with a honeycomb and a screen and passes through a contraction section prior to reaching the test section. Wind tunnel speed is controlled via a variable resistance potentiometer and determined with a calibrated slant manometer. A schematic of this wind tunnel is depicted in figure 7.

Figure 7: UTA 30x30 Low-Speed Wind Tunnel.

For the wind tunnel speed calibration, a Pitot-static tube was inserted into the test section where the test model apex was to be located at $0^\circ$ angle of
attack. The Pitot-static tube was connected to a multimanometer board. The
dynamic pressure was determined from the difference in the fluid levels, \( \Delta h \), for
the Pitot and static pressures as indicated on the manometer board. Simultaneous readings were made on a wind tunnel slant manometer which indicated the pressure inside the test section. The corresponding dynamic pressure has been determined from this calibration using the formula given in reference 10:

\[
q = \Delta h \gamma \rho g \sin \beta
\]  

(1)

where the gravity constant, \( g \), was 32.14 ft/s². Manometer board slant angle, \( \beta \),
was determined as 40.3° from the horizontal and the specific gravity of the
manometer fluid, \( \gamma \), was 0.823. The resulting dynamic pressure versus slant
manometer reading curve is shown in figure 8.

Due to the fact that the 30 x 30 wind tunnel became inoperative shortly after
the smoke-wire flow visualization tests, UTA's closed return low-speed wind
tunnel was utilized for oil-dot flow visualization and force measurements. It can
obtain test section speeds up to approximately 160 ft/s. A rectangular test
section, 36-in. x 24-in. in cross section, is available for the researcher. Figure 9
shows a sketch of the wind tunnel. The dynamic pressure of this tunnel was
previously calibrated by Baker (Ref.11) with a Pitot-static tube against a
pressure transducer (Setra 261-1). This transducer, connected with an interface
to a computer, was then used to read the dynamic pressure during testing.
However, a Pitot-static tube calibration was also performed for the tunnel to
confirm the test section speed obtained via the transducer. Manometer board
slant angle was 46.2° from horizontal and specific gravity of the manometer fluid
Figure 8: Dynamic pressure calibration curve for UTA 30-inch by 30-inch low-speed wind tunnel.
used was 0.812. The corresponding dynamic pressure calibration curve is shown in figure 10.

Reference 12 should be consulted for more information about the closed return low speed wind tunnel.

Figure 9: Sketch of closed circuit low-speed wind tunnel (dimensions in inches) taken from reference 12.
CHAPTER 5
FLOW VISUALIZATION

The experiments to investigate the effects of upper vortex flap deflection and angle of attack on the delta wing configuration can be divided into two main parts. The first part consisted of flow visualization tests to get a better understanding of the flow field and aid in the interpretation of the pressure distribution and balance measurements in the second part of the experiments. Thus, this chapter will focus on flow visualization, while chapter 6 deals with pressure distribution measurements and chapter 7 with the force balance measurements.

5.1 Flow Visualization

Two types of flow visualization were used; the 'smoke-wire' flow field visualization and the 'oil-dot' surface flow technique. Both types will be briefly introduced and their present implementation described followed by test results and a discussion.

5.1.1 Smoke-Wire Flow Visualization

This method utilizes a very fine wire (d = 1/10 mm) inserted into the flowfield, coated with oil, and heated by passing an electrical current through the wire. Oil beads form on the wire from which smoke filaments originate when the wire is heated. In actuality, the "smoke" generated in this way are small liquid particles and should be classified as vapor-condensation aerosol (Ref. 13).
Though it is not smoke in the strict definition, it will be referred to as such in this research. References 13-15 can be consulted for more information about the smoke-wire technique.

5.1.1.1 Electrical Setup for Heating the Smoke Wire

A very simple electrical circuit as shown in fig. 11 was used for resistive heating of the smoke wire. Power was supplied by a POWERSTAT variable transformer (type 116B), which has an output a.c. range from 0 to 140 Volts for an input a.c. of 120 Volts. A power switch was also integrated into the circuit which allowed the user to set the transformer dial to the required output voltage and to switch on the power as needed during testing.

![Electrical Circuit Diagram]

Figure 11: Electrical circuit for resistive heating of the smoke wire.

A rectangular frame (see Fig. 12) was built out of phenolic to mount the smoke wire and this makes it possible to traverse the wire across the test
section in order to introduce smoke into the region of interest (i.e., where the vortex flap is located). This also prevented excessive drilling of holes into the tunnel.

![Diagram of frame used to integrate smoke wire into the test section.](image)

Figure 12: Frame used to integrate smoke wire into the test section.

An often encountered problem with the smoke-wire technique is that when the wire is heated, it expands and sags. Therefore the wire should be pre-stressed. This is accomplished with a simple pulley system mounted on the top of the tunnel stressing the smoke-wire with weights hooked onto it (see Fig.13).

Several aspects have to be taken in consideration for the amount of power needed when heating the wire. Among the most important are the test section temperature and wind tunnel velocity. Low temperatures require a higher output voltage. Likewise the temperature is directly affected by wind tunnel velocity.
Increased velocity causes the wire to cool off faster and thus requires more power to the circuit. Of course the most important aspect is to produce the right amount of smoke. Depending upon the current through the wire, the beads can

Figure 13: Test setup for smoke-wire flow visualization in UTA's 30''x30'' low-speed wind tunnel.
be vaporized very rapidly or, for a lower current, continuous filaments with 2 seconds duration can be produced. If the current is too low the streaklines may become too faint and be inadequate for photography.

5.1.1.2 Smoke Wires

The selection of wire size was limited by the Reynolds number based on the wire diameter. Ideally it is recommended that one should maintain $Re_d < 40$ for the smoke-wire technique to obtain optimum results (Ref. 13). To obtain such low Reynolds numbers two parameters, namely wind tunnel speed and wire diameter have to be kept small. The wind tunnel speed should be kept very low, ideally at idle since the upper speed limit for the usefulness of the smoke-wire technique is about 13 ft/s. The dynamic pressure calibration curve (see Fig.8) for UTA's 30-in x 30-in low-speed wind tunnel reveals no detailed information about the tunnel speed in its idle operation regime, since it represents the start of this curve. Therefore measurements were taken with a micromanometer connected to a Pitot-static tube inserted into the test section. This micromanometer contained unity oil and was sensitive to readings of one thousands of an inch of H$_2$O. Ten readings were taken while the tunnel was operating at idle conditions and averaged to a value of 0.045-in. of H$_2$O. This value converted to a wind tunnel speed at idle of 13.5 ± 0.5 ft/s. According to several references (Refs.13-15) this is about the upper limit for useful smoke-wire flow visualization. But another main factor as mentioned earlier is that Reynolds number is directly proportional to the wire diameter.

A set of stainless steel wires with wire diameters of 0.11, 0.17, 0.22, 0.35, and 0.40 mm have been tested. Preliminary results have shown that the smoke generated with wire diameters greater than 0.17 mm caused broken filaments
not suitable for flow visualization because the wire acts just like a cylinder set into the flow. A laminar vortex street forms behind the wire at these wire diameters. The 0.11 mm diameter wire can not endure the current long enough to allow time for photography (= 2 - 4s) and broke quite easily. The 0.17 mm diameter wire proved to be the most adequate for visualization. A Reynolds number calculation based on this wire diameter at idle conditions yields a value of 46.5. Even though this is above the recommended limit of 40, it is close enough for our purpose.

5.1.1.3 Coating Material and Procedure

A variety of different liquids was tested as a wire coating material for smoke production. It was found that a commercial mineral oil (SQUIBB mineral oil) gave the best results in producing efficient smoke. This selection proved to be also good from a safety point of view since it is non-toxic.

There are a number of methods to coat the wire. For example, a pressurized gravity feed (Ref.13) for vertical wires can be used, with the disadvantage that large droplets blow off the wire and wet the model surface, or the 'Windshield Wiper' (Ref.13) device coating the wire automatically but causing flow perturbation by being in the flow field. The best method after all was coating the wire manually. Access to the tunnel test section was provided through a rear door. Between each use of the wire, this door was opened and the wire coated before power was given to the circuit.

5.1.1.4 Photographic Technique and Lighting

An investigation of the literature regarding vortex flaps (Refs.1-9) has shown that the flow phenomena dealt with has a steady nature until the wing
stalls. However, to indicate that this is true a video camera, a General Electric CG-9806 Camcorder (30 frames/s) was utilized. It should be noted here, however, that in reality any unsteadiness encountered at these low Reynolds numbers can only be detected with a high speed motion camera with at least 1000 frames/s. The film obtained using this camcorder was converted into photographs by using a still camera, a Minolta SRT101 with a MC Rokkor-PF (f1:1.4) lens, and a JVC HR-D820U video cassette recorder (VCR) with frame freezing and frame-by-frame advancement capabilities. A series of photographs 0.066-s apart was obtained in this way (see Fig.14) of a smoke-wire flow visualization run with 135° UVF at a side-view (model is yawed to 40° to the port side) and at 0° angle of attack. Photographs in Fig.14 show very little change in the flow patterns. For still photography, a Minolta SRT101 camera with a MC Rokkor-PF (f1:1.4) lens was used. Experimenting with various Black & White films showed that Tri-X pan with 400 filmspeed gave good results.

For lighting, two 1000-Watt flood lights were utilized. Experiments to achieve optimum lighting conditions made it necessary to put the flood lights on top of the tunnel test section where a glass door was located. The entire setup for this experiment is illustrated in Fig.13.

5.1.1.5 Test Setup and Conditions

Test conditions for the smoke-wire flow visualization were as follows. Tunnel speed was 13.5 ft/s which yielded a Reynolds number based on wire diameter of 46.5. The Reynolds number based on the mean geometric chord commonly known and termed as the mean aerodynamic chord (MAC) of 7.3 inches of the test model was approximately 51,000. The vertical smoke wire was positioned 5 inches from the wing apex upstream of the model at a spanwise
Figure 14: Photographic results of the smoke-wire flow visualization for the delta wing with $135^\circ$ UUF at $0^\circ$ angle of attack covering a time span of 0.267 seconds. $v_\infty = 13.5$ ft/s, $Re_w = 46.5$. 

(a) $t = 0$ s

(b) $t = 0.066$ s
(e) t = 0.267 s

Figure 14: (continued)
station of 2.25". The plane in which the smoke flow acted on the test model is illustrated in figure 15. In figure 15, a local angle of attack ($\alpha_1$) is introduced to represent the effective angle between the flap leading edge and the freestream velocity vector.

Angle of attack of the model was varied with the help of a turnbuckle on the pyramidal balance on which the model was mounted and set to the desired angle with an electronic level indicator with digital display (Smart Level).

Figure 15: Plane in which the smoke flow acts on test model.
5.1.1.6 Results and Discussion

The photographic results of the smoke-wire flow visualization can be seen in figures 16 through 19. These pictures show that for the planar 60° delta wing with no flaps and for the 60° delta wing with 120°, 135°, and 150° UVF the changing flow pattern in the smoke-wire flow plane as shown in figure 15 for an angle of attack range of 0° to 20°.

The pictures in figure 16 visualize the flow behaviour around a planar 60° delta wing. At $\alpha = 0^\circ$ a small separation bubble occurs at the leading edge which reattaches a short distance thereafter inboard of the wing (Fig.16.a). This separation bubble increases in magnitude as $\alpha$ increases to 5° (Fig.16.b) with reattachment assumed to be further inboard of the wing. As $\alpha$ is increased to 10° the separation bubble has grown. In the next picture for $\alpha = 15^\circ$ the flow structure has changed significantly (Fig.16.d). The incoming flow is separated at the fairly thin (0.250 in. thickness) leading edge of the delta and forms a rolled-up vortex sheet. At 20° incidence (Fig.16.e) this vortex sheet has grown in magnitude.

Smoke-wire flow visualization results of the 150° UVF-equipped delta wing are shown in figure 17. The UVF gives the flap surface an angle of attack that is higher than the nominal angle of attack of the wing, i.e., when the wing is at $\alpha = 0^\circ$, the incoming flow sees a leading-edge flap at $\alpha = 14.5^\circ$ (defined as $\alpha_l$ in figure 15) and the leading-edge vortex is formed accordingly. This is seen in figure 17.a. The pictures in figure 17.b - 17.d show the growth of the rolled-up leading-edge vortex on the wing upper surface fairly well. A visual comparison of pictures at the same $\alpha$ in figures 16 and 17 indicates larger vortex formations with increased suction levels for the flapped wing configuration. Unfortunately these pictures (Fig.17) do not reveal detailed information of what is happening
behind the vortex flap (which obstructed the camera's view) to the separated flow due to a three-dimensional model with a two-dimensional flow visualization approach. Nevertheless, in an attempt to see what was occurring behind the vortex flap, the model was yawed about 40° to the port side in order to see an end view of the flap on the starboard side. Unfortunately, results of this attempt showed that the smoke particles were getting into a turbulent region (leading-edge vortex formation) and were diffusing with the downstream flow and besides the yaw angle applied to the model was not characteristic of a flow around a 60° delta with UVF anymore. Examples of this have been shown previously in figure 14.

Pictures in figures 18 and 19 for the 60° delta with 135° UVF and 120° UVF, respectively, reveal in general the same information about the vortex formation as in figure 17 for the 150° UVF case with the difference that the vortex grows with increasing angle of attack and decreasing UVF deflection angle. A visual comparison of these pictures with each other implies that. One further distinguishing characteristic was observed for the 120° case at \( \alpha = 15° \) and \( \alpha = 20° \) in figures 19.d and 19.e, respectively. The spiral vortex induced by the leading-edge separation in these pictures seems to have grown so much that it is not able to reattach on the wing and is carried downstream. A hypothesized pattern of this phenomena is shown in figure 19.f.

To obtain further visual information about the flow structures caused by UVF on the 60° delta wing the 'Oil-Dot' technique has been employed. Section 5.2 introduces these techniques with subsequent results and discussions.
Figure 16: Photographic results of smoke-wire flow visualization for planar delta wing without flaps at angles of attack from $0^\circ$ to $20^\circ$. $v_\infty = 13.5$ ft/s, $Re_w = 46.5$.
(c) Planar delta, $\alpha = 10^\circ$

(d) Planar delta, $\alpha = 15^\circ$

Figure 16: (continued)
(e) Planar delta, $\alpha = 20^\circ$

Figure 16: (continued)
Figure 17: Photographic results of smoke-wire flow visualization for the delta wing with 150° UVF at angles of attack from 0° to 20°. $v_\infty = 13.5$ ft/s, $Re_w = 46.5$. 

(a) 150° UVF, $\alpha = 0°$

(b) 150° UVF, $\alpha = 5°$
Figure 17: (continued)
(e) $150^\circ$ UVF, $\alpha = 20^\circ$

Figure 17: (continued)
Figure 18: Photographic results of smoke-wire flow visualization for the delta wing with 135° UVF at angles of attack from 0° to 20°. $v_\infty = 13.5$ ft/s, Re$_w$ = 46.5.
(c) $135^\circ$ UVF, $\alpha = 10^\circ$

(d) $135^\circ$ UVF, $\alpha = 15^\circ$

Figure 18: (continued)
(e) $135^\circ$ UVF, $\alpha = 20^\circ$

Figure 18: (continued)
Figure 19: Photographic results of smoke-wire flow visualization for the delta wing with 120° UVF at angles of attack from 0° to 20°. $v_\infty = 13.5$ ft/s, $Re_w = 46.5$. 

(a) 120° UVF, $\alpha = 0°$

(b) 120° UVF, $\alpha = 5°$
(c) $120^\circ$ UVF, $\alpha = 10^\circ$

(d) $120^\circ$ UVF, $\alpha = 15^\circ$

Figure 19: (continued)
(e) $120^\circ$ UVF, $\alpha = 20^\circ$

(f) $120^\circ$, $\alpha = 15^\circ$

Figure 19: (continued)
5.2 Oil-Dot Technique

The oil-dot technique is a flow visualization technique employing discrete dots of an indicator fluid, i.e., an artist's oil mixed with thinner, on the surface of the model configuration to be investigated. The model is then mounted in the tunnel. After the tunnel has been run, oil streaks result from surface shear stresses and give the direction of the limiting surface streamlines. These streaklines will indicate regions of separated flow and flow reattachment, if any occur. Flow attachment lines are identified by limiting streamline divergence whereby flow separation lines are identified by limiting streamline coalescence with the oil-dot method. More information about the oil-dot technique can be obtained from references 16 and 17.

5.2.1 Test Setup for the Oil-Dot Technique

The oil-dot tests were conducted in UTA's closed return low-speed wind tunnel at a wind tunnel velocity of approximately 87.7 feet/second representing a dynamic pressure of 8.7 lb/ft². Average test section temperature and density for all runs was at 76° F and 0.002262 slug/ft³, respectively. The average Reynolds number based on the mean aerodynamic chord length of 7.3 inches was determined as approximately 323,000. A higher Reynolds number for the oil-dot flow visualization was selected in comparison to the smoke wire flow visualization because of the resistance of the oil mixture at low tunnel speeds. Angle of attack was changed through a turnbuckle on the balance system on which the model has been mounted and set to the desired angle with the help of a propeller protractor.

During testing, discrete dots of the oil mixture (Grumbacher artist's oil and paint thinner) were applied on the model mounted in the tunnel at several wing-
span stations starting at a station close to the wingtip and going downstream on the model. After the tunnel was run the model was taken out of the tunnel and the changed local surface streamline pattern photographed with a Minolta SRT101 still camera. The model was then cleaned off and ready for the next angle of attack run.

For the oil-dot technique an additional constant chord UVF with 165° deflection angle was used besides the three UVF with 120°, 135°, and 150° deflection angles available.

5.2.2 Results and Discussion

The photographic results of the oil-dot technique are summarized in figures 20 - 23 for UVF deflections of 165°, 150°, 135°, and 120°, respectively. Figure 20(a) for the 165° UVF case represents the surface shear stress structure for α=0°. This particular picture shows a symmetric flow on the upper surface of this configuration. There is a secondary flow separation line on the flaps (primary flow separation takes place at the flap leading edges) which starts close to the flap leading edge and moves inboard with the downstream flow. Also on this picture (Fig.20.a) a flow reattachment of the separated flow from the flap leading edge appears to occur close to the wing-flap hinge line on the flap. This reattachment moves inboard downstream of the flow but remains at the wing-flap junction. The flow on the remainder of the wing seems to be progressing uniformly in the incoming flow direction. As α is increased to 5° in the next picture (Fig.20.b), some changes in the flow structure occur. The secondary separation line on the flap has moved further outboard. The flow reattachment line did move further inboard on the flap. This reattachment line which starts on the flap moves on to the wing further downstream on the model. Unfortunately,
this picture does not reveal detailed information about the flow structure close to the wing apex. Therefore more discrete dots of the oil mixture have been applied in this region.

A better result of the oil-dot technique is pictured in the $\alpha = 10^\circ$ case (Fig.20.c). The secondary flow separation line and flow reattachment line are also featured in this configuration. Flow reattachment starts on the flap close to the wing apex, but at a short distance downstream it moves on to the wing. The flow structure close to the wing apex indicates also a flow separation line moving towards the flap leading edge apex meaning that the air in this region is drawn forward on the wing. In the other direction (flow direction) of this separation is the start of the flow reattachment line mentioned above. The next picture of the $\alpha = 15^\circ$ case (Fig.20.d) reveals more information about the flow separation structures. As expected the new separation line recognized in Fig.20.c is the start of the secondary separation line. In this picture (Fig.20.d) it is clearly visible how the air close to the wing apex is drawn forward in a converging manner to form the start of the secondary separation line at the junction of the two flaps. Flow reattachment is still initiated on the flap but moves on to the wing a short distance downstream. At $\alpha = 20^\circ$ (Fig.20.e) the above mentioned features are also encountered with secondary flow separation taking place closer to the flap leading edges and flow attachment line moving to the wing center.

An examination of the photographic results for the remaining UVF configurations in figures 21 - 23 of 150°, 135°, and 120°, respectively, reveals in general the same global flow structure as for the 165° UVF case (Fig.20) discussed in detail above. Generally secondary separation lines move inboard with the downstream flow and outboard as $\alpha$ increases. Flow reattachment lines
on the other hand tend to move inboard for increasing $\alpha$ where finally at a
certain $\alpha$ (varies from one UVF deflection to the other) they seem to be
converging to the root chord. Also the onset of the above mentioned features
taking place on these configurations starts at an earlier stage with decreasing
UVF deflections.
Figure 20: Photographic results of oil-dot technique for the delta wing with $165^\circ$ UVF at angles of attack from $0^\circ$ to $20^\circ$. $v_\infty = 87.7$ ft/s, $Re_w = 323,590$. 
(c) $165^\circ$ UVF, $\alpha = 10^\circ$

(d) $165^\circ$ UVF, $\alpha = 15^\circ$

Figure 20: (continued)
(e) $165^\circ$ UVF, $\alpha = 20^\circ$

Figure 20: (continued)
Figure 21: Photographic results of oil-dot technique for the delta wing with 150° UVF at angles of attack from 0° to 20°. \( v_\infty = 87.7 \text{ ft/s}, \) \( \text{Re}_w = 323,590. \)
(c) $150^\circ$ UVF, $\alpha = 10^\circ$

(d) $150^\circ$ UVF, $\alpha = 15^\circ$

Figure 21: (continued)
(e) $150^\circ$ UVF, $\alpha = 20^\circ$

Figure 21: (continued)
(a) 135° UVF, $\alpha = 0^\circ$

(b) 135° UVF, $\alpha = 5^\circ$

Figure 22: Photographic results of oil-dot technique for the delta wing with 135° UVF at angles of attack from 0° to 15°. $v_{in} = 87.7$ ft/s, $Re_w = 323,590$. 
(c) 135° UVF, $\alpha = 10^\circ$

(d) 135° UVF, $\alpha = 15^\circ$

Figure 22: (continued)
(a) 120° UVF, $\alpha = 0^\circ$

(b) 120° UVF, $\alpha = 5^\circ$

Figure 23: Photographic results of oil-dot technique for the delta wing with 120° UVF at angles of attack from 0° to 10°. $v_\infty = 87.7$ ft/s, $Re_w = 323,590$. 
(c) $120^\circ$ UVF, $\alpha = 10^\circ$

Figure 23: (continued)
Figure 24: An interpretation of the flow about the 60° delta wing with 165° UVF at 5° angle of attack.

As a result of the photographic experiments a possible interpretation of the flow field about a 60° delta wing with 165° UVF is illustrated as in figure 24. In this figure the behavior of the flow close to wing apex is not sketched because of the complexity of the flow field in this region which could not be completely determined with the smoke-wire and oil-dot flow visualization techniques.
CHAPTER 6
PRESSURE DISTRIBUTION MEASUREMENTS

Upper surface pressure distribution measurements were made in UTA's 24-in x 36-in closed return low-speed wind tunnel for a 60° delta wing (see Fig.3-5) employing 135°, 150°, and 165° UVF. The pressure distribution was measured in the spanwise direction on the wing (perpendicular to the free-stream flow direction) and in the chordwise direction on the flaps (parallel to the free-stream flow direction). The pressure data obtained for the wing from these measurements were compared to the pressure data of the planar delta without flaps with the aim of obtaining an indication of the suction levels generated by the UVF vortex system and to see the relative effects of flap deflection angle and angle-of-attack variations.

6.1 Test Setup and Conditions

There are 24 pressure orifices along half of the wing span on the right hand side at 64.2% of the root chord from the wing apex on the delta wing and 15 pressure orifices on the starboard flap along the flow direction at 29.2% of the flap span from the apex, restricted to the upper surfaces of wing and flap only (see Fig.3). The pressure measurements were taken with the help of two multi-manometer banks with 20 ports each. Both were held at a 40° slant angle measured from the horizontal and contained alcohol with a specific gravity of 0.812 as the working fluid. The test section velocity for all runs was kept at approximately 87.7 feet/second corresponding to a dynamic pressure of 8.7
lb/ft². The average temperature and density were 76° F and 0.002262 slug/ft³, respectively. This converted to a test Reynolds number based on the 8.66-inch MAC of the pressure model of approximately 384,000. The model was mounted on a pyramidal balance and angle of attack was changed through a turnbuckle on the balance system and set to the desired angle with a propeller protractor.

### 6.2 Results and Discussions

Testing for the pressure distribution measurements was done for the angle of attack and model configurations shown in Table 1.

Table 1: Test matrix for pressure model configuration.

<table>
<thead>
<tr>
<th>Configuration</th>
<th>Angle of Attack</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>-4°</td>
</tr>
<tr>
<td>Planar delta *</td>
<td>X</td>
</tr>
<tr>
<td>165° UVF wing</td>
<td>X</td>
</tr>
<tr>
<td>150° UVF wing</td>
<td>X</td>
</tr>
<tr>
<td>135° UVF wing</td>
<td>X</td>
</tr>
</tbody>
</table>
| * Pressure distribution measurements on wing with no flaps.

The difference in fluid levels, Δh (in), measured from the multi-manometer bank during testing was converted into pounds per square foot via Eq.1 given in chapter 4 and represent p - p_{amb} values. These results then were expressed in the nondimensionalized form of pressure coefficients (c_p) which is defined as,

\[ c_p = (p - p_{\infty}) / q_{\infty} \]
where $p_\infty$, the free-stream static pressure has been obtained previously from dynamic pressure calibration tests.

All pressure distribution measurement results are presented in graphical form for wing and flap configurations individually in figures 25 - 31. Results for the planar delta wing, 165° UVF wing, 150° UVF wing, and 135° UVF wing are shown in figures 25 - 28, respectively, and results for the 165° flap, 150° flap, and 135° flap they are shown in figures 29 - 31, respectively. All graphs are in the form of pressure coefficient versus nondimensionalized wing half-span for the wing and pressure coefficient versus nondimensionalized flap chord for the flaps.

Figure 25 for the pressure distribution over the planar delta wing with no flaps indicates that the suction peak on the upper wing surface moves inboard as $\alpha$ increases and increasing in its magnitude to approximately $c_p = 1.5$ for $\alpha = 16^\circ$. At $\alpha = -4^\circ$ and $\alpha = 0^\circ$ this graph does not reveal the location of this suction peak which seems to be lying close to the wing leading edge where pressure orifices could not be placed due to limitations of the model in this region.

The suction levels on the wing drop by more than 70% in figure 26 for the wing with 165° UVF. Again the same trend as for the planar delta are observed about the suction levels with increasing $\alpha$ with the difference that these suction peaks are closer to the wing leading edge. As the flap deflection angle is decreased to 150° (Fig.27) higher suction peaks are generated on the wing as $\alpha$ increases and they seem to move further inboard compared to the 165° case, but still showing less inboard movement than the planar delta with no flaps and the highest suction peak being 33% less than the planar delta case. The same
Figure 25: Spanwise pressure distribution on 60-degree planar delta wing at 64.2 percent of the root chord.
Figure 26: Spanwise pressure distribution on a 60-degree delta wing with 165-degree UVF at 64.2 percent of the root chord.
Figure 27: Spanwise pressure distribution on a 60-degree delta wing with 150-degree UVf at 64.2 percent of the root chord.
Figure 28: Spanwise pressure distribution on a 60-degree delta wing with 135-degree UVF at 64.2 percent of the root chord.
The wing apex deflects flap (UAF) at a distance of 29.2 percent of flap span from flap chord.

Figure 29: Chordwise pressure distribution on a 165-degree upwars flap chord.
Figure 31: Chordwise pressure distribution on a 135-degree upwards deflected flap (UVF) at a distance of 29.2 percent of flap span from the wing apex.
trends are observed for the 135° UVF case in figure 28 where the highest suction peak achieved on the wing is 25% less than that for the planar delta.

An examination of the pressure distribution curves for all UVF cases in figures 29 - 31 shows increasing suction levels on the flap upper surfaces with the tendency of such to move inboard with decreasing UVF deflections and increasing $\alpha$. In all three UVF cases an interesting aspect that is occurring in these graphs is that the curves are flattening out starting at $\alpha = 12^\circ$, which is believed to be due to flap stalling. Wing stalling in comparison is encountered at higher angles of attack. As will be seen later from force balance measurement results the flap stalling does not change the overall performance of the wing to higher angles of attack.

It is estimated that the overall error stemming from reading the pressure measurements manually, determining the specific gravity of the fluid used in the manometers, determining the manometer slant angle etc. is about 5%.
CHAPTER 7
FORCE BALANCE MEASUREMENTS

Force balance measurements were conducted at UTA's 24-in x 36-in closed return low-speed wind tunnel (see chapter 4) to determine the effect of UVF deflections and angle of attack on the longitudinal forces of a 60° delta wing employing UVF with deflection angles of 120°, 135°, 150°, and 165°.

7.1 Test Setup and Conditions

Force and moment data were obtained with the help of a pyramidal six-component strain gage balance, an HP-3497A data acquisition system (DAS) and a computer. A previously supplied computer program (Ref. 11) was used to trigger the HP-3497A to measure the output voltage from each load cell which was previously supplied with a 6-volt excitation voltage and to reduce these electrical readings into forces and moments. The computer program was programmed to control a 'General Purpose Interface Bus (GPIB)' to transfer data from the balance load cells to the DAS. This balance system was calibrated with a series of known forces and moments for lift, drag and pitching moment, since only these data were to be obtained from the balance measurements. The following calibration constants resulted from this calibration:

Lift constant = -9,750 lb/millivolt
Drag constant = -7,030 lb/millivolt
Pitching moment constant = 8,000 in-lb/millivolt
Each time the DAS was triggered by the program it took 100 readings for each load cell which were averaged by the software and an averaged bare reading (reading taken while tunnel is not running) was subtracted from this value. These average millivolt readings were then multiplied by the calibration constants to obtain lift, drag, and pitching moment.

Model angle of attack was varied with a turnbuckle on the balance system rotating the strut on which the model was mounted and set to the desired angle with a propeller protractor.

All tests were run at a test velocity of approximately 84.8 feet per second corresponding to a dynamic pressure value of 8.13 pound per square foot. Average test section static temperature and density were 78° F and 0.0022613 slug/ft³, respectively. An average Reynolds number of approximately 313,000 for the balance measurements based on the MAC of the wing of 7.3 inch was determined.

7.2 Results and Discussion

Four runs were made at each angle of attack for every configuration in the test matrix in table 2 and averaged. Strut effects were determined with a strut only run and averaged for seven runs. These values ( \( L_{str} = 0.1481 \) lb, \( D_{str} = 0.1383 \) lb, and \( M_{str} = -0.3553 \) in-lb) were then subtracted from the raw data. The raw data were then converted into coefficient data by dividing the lift and drag forces by the dynamic pressure and the projected model area. The pitching moment was transferred to the model aerodynamic center which is defined as the point on the wing at which the pitching moment coefficient does not change with the angle of attack. From this pitching moment the pitching moment coefficient referenced to the aerodynamic center was obtained through dividing
by the dynamic pressure, the projected model area, and the mean aerodynamic chord (MAC). All coefficients obtained by these calculations are based on projected wing area plus projected flap area; hence the effect of flap area is divided out, giving only the real aerodynamic effects of the flaps. These areas are given in Table 2 for all configurations. The vertical distance from the wing chord to the aerodynamic center can be neglected since the model in perspective is thin (0.250 inch). Before computing the pitching moment it was found that the balance moment resolving center which is 12\frac{1}{4} inches above the balance socket coincide with the model pivot point, eliminating the necessity of including the vertical distance between these two points in the pitching moment coefficient calculations.

The reduced coefficient data was then recalculated through a wake-blockage correction method given by Rae & Pope (Ref.10) in Eq.3 which makes corrections to the dynamic pressure.

\[ \frac{q_d}{q_u} = 1 + (S/2C) C_{D_0} + (5S/2C) C_{D_s} \]  

(3)

The profile \((C_{D_0})\) and separation \((C_{D_s})\) drag coefficients are determined with a graph of drag coefficient versus lift coefficient squared. The profile drag is the drag at zero lift and the separation drag is the difference between the measured drag coefficient (a departure from linearity is observed for separated flow in the \(C_D\) versus \(C_L^2\) curves) and the induced drag coefficient. The blockage ratio \((S/C)\) of this model was 0.08. \(q_u\) in this formula represents the uncorrected dynamic pressure values. By substituting these values into Eq.3 a corrected dynamic pressure \((q_d)\) results which was then used to recalculate all coefficient data.
Flow angularity test results conducted in reference 12 for UTA's closed return low-speed wind tunnel indicate that the flow direction in the middle of the test section cross section area is nearly unidirectional in the flow direction. Maximum flow deflections of 0.75° in sideslip angle and 0.50° from the horizontal plane are encountered in regions close to wall boundaries.

The test matrix for the force and moment measurements is given in Table 2.

Table 2: Test matrix for force balance measurement model.

<table>
<thead>
<tr>
<th>Configuration</th>
<th>Angle of Attack</th>
<th>$S[ft^2]$</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>-6°  -3°  0°   3°   6°   9°   12°   15°   18°   20°   22°   24°   26°</td>
<td></td>
</tr>
<tr>
<td>Planar delta *</td>
<td>X    X    X    X    X    X    X    X    X    X    X    X    0.48</td>
<td></td>
</tr>
<tr>
<td>165° UVF wing</td>
<td>X    X    X    X    X    X    X    X    X    X    X    X    0.76</td>
<td></td>
</tr>
<tr>
<td>150° UVF wing</td>
<td>X    X    X    X    X    X    X    X    X    X    X    X    0.73</td>
<td></td>
</tr>
<tr>
<td>135° UVF wing</td>
<td>X    X    X    X    X    X    X    X    X    X    X    X    0.68</td>
<td></td>
</tr>
<tr>
<td>120° UVF wing</td>
<td>X    X    X    X    X    X    X    X    X    X    X    X    0.62</td>
<td></td>
</tr>
</tbody>
</table>

* Force balance measurements on wing without flaps.

The effects of UVF deflection angles on the location of the aerodynamic center were determined from the coefficient data using a method described in Ref.10, which makes the assumption that the moment is due entirely to the lift and that the aerodynamic center is on the chord line. The following equation (Ref.10) was then used to calculate the location of the aerodynamic center:

$$ac = tr - \frac{dC_{mtr}}{dC_{L}} \quad (4)$$
where ac and tr are the distance of the aerodynamic center and the balance moment resolving center from the wing leading edge, respectively, in fractions of a mean aerodynamic chord. \( dC_{mr}/dC_L \) is the slope of the linear portion of the pitching moment coefficient about the balance moment center (which is the reduced and corrected pitching moment obtained from the balance measurements) versus lift coefficient. The results of these computations are plotted in figure 31. As expected the aerodynamic center (a.c.) does move significantly forward on the wing with the usage of UVF. A gradual forward movement of the a.c. is encountered up to UVF deflection angles of 150°. For the 165° UVF case the a.c. moves aft.

All coefficient data obtained from the force and moment measurements are plotted in graphical output format in figures 32 - 35. Figure 32 represents the lift coefficient versus angle of attack curves for all UVF configurations in comparison to the planar delta. As can be seen very well from this graph the UVF deflection angles used for this investigation do not change the lift curve slope up to their individual stall angle of attack and tend to shift the lift curve to the left. Significant increases in \( c_L \) are observed with decreasing deflection angles for all UVF's. Optimal UVF flap deflection from this graph (Fig.32) seems to be the 150° UVF case which maintains an approximate \( \Delta c_L \) of 0.20 over the entire \( \alpha \)-range until stall is reached at \( \alpha = 20° \) without sacrificing too much on the \( c_{Lmax} \) value in comparison to the planar delta.

Drag coefficient data have been graphed in figure 33 for all UVF deflection angles compared to the planar delta. As expected, drag increases are caused by the usage of UVF. The drag curve shifts to the left as UVF deflection angles decrease. At low angles of attack (\(-\alpha\)-range) the 165° UVF exhibits less drag
Figure 32: Effect of vortex flap deflection on the location of the aerodynamic center.
Note: Area effects of flaps to aerodynamic data are included.

Figure 33: Lift coefficient versus angle of attack with flap upwards deflections as parameter.
Note: Area effects of flaps to aerodynamic data are included.

Figure 34: Effect of UVF deflections on drag polar.
Note: Area effects of flaps to aerodynamic data included.

Figure 35: Effect of UVF deflections on the lift-to-drag ratio.
Note: Area effects of flaps to aerodynamic data are included.

Figure 36: Effect of UVF deflections on the pitching moment referenced to the aerodynamic center of each flap configuration.
than the planar delta, which is believed to be connected to the low local angle of attack (see Fig.15) that the incoming flow encounters at the flap leading edge.

However, to get a better interpretation of the effects of UVF deflection angles on the present configuration the efficiency (lift-to-drag ratio) versus lift coefficient was plotted in figure 34 for all model configurations available. The efficiency is dropping with the usage of UVF in comparison to the planar delta as observed in Fig.34. The optimal UVF deflection of those tested seems to be the 165° case, which sacrifices the least in the maximum attainable efficiency values.

The pitching moment about the a.c. location versus angle of attack is shown in figure 35 for all UVF deflections in comparison to the planar delta. According to this graph (Fig.35) the pitching moment behaves nearly constant for all UVF deflections up to their respective stall angle of attack with the exception of the 120° UVF case which decreases beyond an angle of attack of approximately 3°.

Since 100 readings were taken from each load cell with the data acquisition system and averaged it is believed that the error in the obtained force data is at about 1 %. However, it is estimated that the computation of lift, drag, and pitching moment coefficients from the reduced force data yielded an error margin of approximately 2.5 %. Due to the fact that the location of the aerodynamic center was determined with an approximate formula (Eq. 4) introduced an error of approximately 7.5 %.
CHAPTER 8
CONCLUSIONS AND RECOMMENDATIONS

An experimental investigation was conducted to determine the effect of upwards deflected flap deflection angles and angle of attack of constant chord upper vortex flaps (UVF) on the performance of a 60° delta wing. Flow visualization results have given an understanding of the flow structure on the upper surfaces of the present model configuration. It has been observed by use of the smoke-wire technique that the flow pattern over the UVF has a steady nature and that the vortices generated by the leading-edge flap separation are growing with increasing angle of attack and decreasing UVF deflection angles. Oil-dot tests for the 165° UVF case showed a flow reattachment on the flap close to the wing-flap hinge line at low-to-moderate angles of attack. The flow reattachment tended to move further inboard with decreasing UVF deflections and increasing angle of attack approaching the wing root chord. Secondary flow separation lines have been recognized with the tendency to move outboard with decreasing UVF deflections and increasing angle of attack.

Upper surface pressure measurements reveal that the suction levels on the wing are dropping with the usage of UVF compared to the planar delta without flaps with the largest drop of 70% encountered for the 165° UVF. High suction levels were achieved on the UVF itself with suction peaks exceeding the pressure levels reached by the wing only tests. Flap stall was noticed above an α of 8° for all UVF deflections, but this did not seem to disturb the flow over the remainder of the wing.
Finally, force balance measurements showed that the usage of UVF brought up large increments of lift and drag at low-to-moderate angles of attack compared to the planar delta for all UVF deflections tested. However, the $c_{L_{\text{max}}}$ values are dropping with decreasing UVF deflections as a penalty for the large $c_L$ increments at low-to-moderate angles of attack with a maximum reached at $c_{L_{\text{max}}} = 1.09$ and $\alpha = 20^\circ$ for the 165$^\circ$ UVF wing. At first glance it appears that the very large increase in drag at low angles of attack which is about 400% more at $\alpha = 0^\circ$ for the 120$^\circ$ UVF compared to the planar delta with no flaps would negate any advantage given by the lift increase. This is especially proven by the efficiency (L/D) results, which indicate a large drop in performance at low-to-moderate angles of attack. However, there are cases in which the combination of increased drag and lift are of advantage. Such a case, for example, is the landing of a high-speed aircraft where it is desirable to maintain sufficient engine power to abort the landing and go around. The high lift levels reached by the usage of upper vortex flaps would allow a lower approach speed at a lower angle of attack and by the same token improve the visibility of the pilot. The high drag on the other hand can brake the aircraft and provide high sink rates. Thus the right application of the UVF would result in a more responsive aircraft.

Further studies are recommended to investigate the effect of tapered chord flaps (meaning the flap chord increases linearly from 0 at the wing's apex to a determined size at the wing tip) on the aerodynamic forces of the present model configuration. Also it would be interesting to investigate the lateral stability characteristics of this wing-flap system.
REFERENCES


