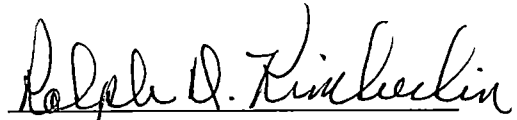


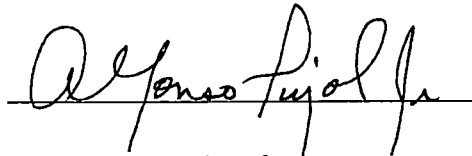
To the Graduate Council

I am submitting herewith a thesis written by Courthney Carlos Cumberbatch entitled "The effects of half delta strakes on the performance of the Piper Navajo twin engine airplane." I have examined the final copy of this thesis for form and content and recommend that it be accepted in partial fulfillment of the requirements for the degree of Master of Science, with a major in Aviation Systems

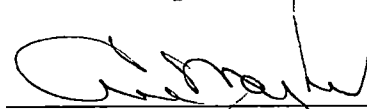


Ralph Kimberlin, Major Professor

We have read this thesis
and recommend its acceptance



Accepted for the Council



Associate Vice Chancellor and
Dean of The Graduate School

**THE EFFECTS OF HALF DELTA STRAKES ON
THE PERFORMANCE OF THE PIPER NAVAJO TWIN ENGINE
AIRPLANE**

A Thesis

Presented for the

Master of Science

Degree

The University of Tennessee, Knoxville

Courthney Carlos Cumberbatch

December 2000

To my late uncle
Carlisle Henry Cumberbatch

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ABSTRACT

The effects of mounting half delta-wing strakes onto the fuselage and nacelles of a Piper Navajo twin-engine airplane are observed. The strakes' ability to enhance the performance of the Piper Navajo is evaluated by its effects on the power required and on lift and drag characteristics of the airplane. Previous wind tunnel and flight test studies have shown that strakes can improve the performance of aircraft by strengthening the airflow over the wing's trailing edge. These studies showed that Lift coefficient values with respect to angle of attack were increased. Drag coefficient values with respect to lift coefficient were also significantly reduced. Lift to drag ratios was improved significantly. Results showed that the strakes significantly reduced the power required for the Piper Navajo, thereby increasing maximum range and endurance. Drag was significantly reduced and the maximum lift to drag ratio was effectively improved.

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NOMENCLATURE

AR	aspect ratio
b	wing span
C	wing chord
C_D	drag coefficient
C_{D0}	drag coefficient at zero lift
C_L	lift coefficient
C_P	pressure coefficient
D	drag
dC_D/dC_L^2	slope of C_D vs. C_L^2 plot
e	Oswald's efficiency factor
F_R	thrust required
Hg	mercury
H_P	instrument corrected pressure altitude
H_{PO}	observed pressure altitude
L	lift
lb.	pounds
L/D	lift to drag ratio
$L/D_{\max}$	maximum lift to drag ratio
P_A	pressure at altitude
P_R	power required
P_{SL}	pressure at standard sea level conditions
S	wing area

T_a	corrected ambient air temperature at altitude
T_C	instrument corrected ambient air temperature
T_O	observed ambient air temperature
T_{SL}	temperature at standard sea level conditions
V_C	calibrated airspeed
V_I	indicated airspeed
$V_\square$	observed airspeed
W_S	standard maximum takeoff gross weight
W_T	weight at the test point
α	angle of attack
δ	pressure ratio
η_P	propeller efficiency
Λ_W	strake sweep
π	PI
θ	temperature ratio
ρ_0	standard sea level density
σ	density ratio

Abbreviations

BHP_{IW}	brake horsepower corrected for all air densities and weights
BHP_T	brake horsepower at that instant
c g	center of gravity
FAR	Federal Aviation Regulations

FWD forward

in inches

K degrees Kelvin

kts. knots (nautical miles per hour)

M Mach number

MP instrument corrected manifold pressure

MP_O observed manifold pressure

PIW brake horsepower corrected for all air densities and weights after normalization

RPM revolutions per minute

RPM_O observed revolutions per minute

VIW airspeed corrected for all air densities and weights

CHAPTER I

INTRODUCTION

Drag is a major factor that is taken into consideration in the design of modern aircraft. Airplane designers try to design airplanes that produce the least amount of drag. In its many forms, drag inhibits the forward motion of an aircraft.

When the flow separates from the surface, drag is increased. Flow separation is caused by a significant loss of energy as the flow travels toward the trailing edge of an airplane's wings. This loss in energy can result in an aircraft stalling or losing lift when the location of the point of separation is shifted more towards the wing's leading edge. This shift in the separation point occurs when the wing's angle of attack is increased.

Designers and aerodynamicists have tried many schemes to delay flow separation at high angles of attack. This thesis discusses such a scheme, using delta strakes fastened to the fuselage and nacelles of a Piper Navajo twin-engine airplane. Studies have shown that delta strakes have the ability to reduce drag and delay flow separation by producing vortices that interact with and energize the flow of air over the wings.

Thomas Wallis did a wind tunnel study at Wichita State University for a master's thesis, in which delta strakes were placed on a half twin-engine model (Wallis, 1981). Wallis observed that the strakes were very successful in reducing interference drag at the wing-fuselage and wing-nacelle intersections.

This thesis examines the use of the strakes for reducing interference drag and delaying flow separation by showing the effect that the strakes have on the amount of

power required from the engines and on the lift and drag characteristics of the Piper Navajo. It also discusses and compares the results of the flight tests with the Piper Navajo with the wind tunnel tests conducted by Wallis.

Chapter II of this thesis gives a discussion of the properties of delta wings, which enable them to be very good drag reducers. A discussion on previous wind tunnel and flight test experiments using delta strakes, including Wallis' study is also given. The equipment used is presented in Chapter III. Chapter IV presents the flight test method used to obtain the parameters to be reduced. The data reduction methodology is presented in Chapter V. Chapter VI presents the results obtained from the experiment and Chapter VII presents conclusions realized and recommendations for further study.

CHAPTER II

BACKGROUND

Flow separation occurs when the flow of air over a wing's surface does not remain attached. This is due to a significant reduction in flow energy when the airplane reaches higher angles of attack.

Normally when air flows over a wing, the pressure decreases and it reaches a minimum as it flows over the point of maximum wing thickness. This decrease in pressure is one of the sources of lift. The pressure then gradually increases as the flow approaches the wing's trailing edge. This increase in pressure is known as an adverse pressure gradient. This adverse pressure gradient is the source of lift-induced drag. When the airplane is in normal straight and level flight, this adverse pressure gradient is low and the flow of air has enough energy to overcome this adverse pressure gradient and thus remain attached at the wing's trailing edge.

When the angle of attack is raised however, the situation changes. The pressure decreases over the point of maximum wing thickness, providing more lift, but the adverse pressure gradient increases, thus also increasing lift-induced drag. If the wing's angle of attack continues to be raised, it will eventually reach a point where the adverse pressure gradient will be too high for the energy of the flow to overcome, thus causing the flow to separate from the wing. This separation is the beginning of a stall.

Past studies have been performed to investigate ways to reduce this lift-induced drag and delay flow separation. Flow separation must be controlled in order to increase system performance with a savings in energy. Referring to basic aerodynamics, at a

given distance from a surface, the velocity of a fluid element in a turbulent boundary layer is higher than in a laminar boundary layer. Thus there is more kinetic energy in the flow nearer to the surface and hence the flow is less inclined to separate. This shows, therefore, that laminar boundary layers will separate more easily than turbulent boundary layers. It is better therefore to have a turbulent boundary layer to maintain attached flow.

It was discovered that one way to make a flow turbulent was to create a vortex in the flow. As a result, numerous types of vortex generators have been designed and numerous studies on these types of vortex generators have been conducted.

The general idea of a vortex generator is to energize the flow of air passing over a wing. It does this by creating vortices that "mix" with the flow of air over a wing. The vortices contain fluid particles with high momentum in the streamwise direction. The fluid particles are swept along helical paths towards the wing surface, and it mixes with the slower moving air at the surface. The boundary layer flow is thus "energized", thereby having enough energy to overcome the adverse pressure gradient.

The United Aircraft Research Corporation used vortex generators in 1946 (Taylor, 1950). The scientists working in the United Aircraft Research Laboratories wanted to investigate increasing the maximum test velocity in the 8 foot diameter test section of their large wind tunnel. The maximum test velocity was limited by rather large pulsations in the test section flow. It was suspected that these pulsations were due to the boundary layer separating at a station downstream from the diffuser entrance. This flow separation was causing non-uniform flow in the diffuser following the test

section. Vortex generators were conceived and developed during a program that was directed towards eliminating this boundary layer separation

This first application of vortex generators turned out to be very successful in eliminating this boundary layer separation. Afterwards they determined that vortex generators could be used in other applications, such as increasing the efficiency of wind tunnel fans, decreasing the mixing length of burners, increasing the secondary flow rate of ejectors, increasing the heat transfer from a hot fluid to the surrounding medium and increasing the maximum lift of wings, while minimizing the drag, as is shown in this thesis

One such type of vortex generator, which this thesis will discuss, is strakes. Strakes are used on airplanes to control flow separation. Strakes can be applied on aircraft in two ways. They can be an integrated part of the wing, as is done on some military fighter planes, or it can be mounted parallel to the wing, as is done in this thesis. It has been shown that the best types of strakes have a delta shape planform. This chapter will first discuss a study done which reveals delta shape characteristics. Afterwards, studies done on integrated wing-strake models are discussed and finally a wind tunnel experiment in which strakes are mounted parallel to a half twin-engine model are discussed, the results of which are compared to the results of this thesis.

Characteristics of Delta Shapes in Drag Reduction

The strakes applied in this thesis are basically delta shapes. A picture of a delta planform showing flow visualization of the vortices it produces is shown in Figure 2.1. Studies were done on the characteristics of delta shapes by Dr. William Wentz in which

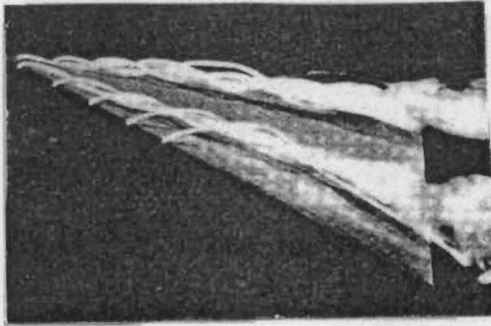


Figure 2.1 – Delta Shape Planform with Flow Visualization of Vortices

he investigated the points where vortices generated by sharp-edged delta shapes would breakdown (Wentz, 1971). The glass wind tunnel setup for Wentz's study is shown in Figure 2.2 and the model installation is shown in Figure 2.3. These investigations showed that the angle of attack where the vortex would breakdown increases with increasing sweep angle for sweep angles less than 75 degrees. For sweep angles greater than 75 degrees, the breakdown occurs at a constant angle of attack. The investigation thus showed that the optimum sweep angle for sharp-edged delta shapes is 75 degrees. This effect of sweep on the vortex breakdown point on the delta shapes is shown in Figure 2.4.

This high leading edge sweep angle enables the strakes to produce vortices that flow over the inboard portion of the aircraft wing, closer to the wing-fuselage junctions, thus alleviating flow separation due to interference drag. Strakes swept at lower leading edge sweep angles are not as effective because the vortices produces flow over more of the outboard portion of the aircraft wing, not having as much effect on interference drag.

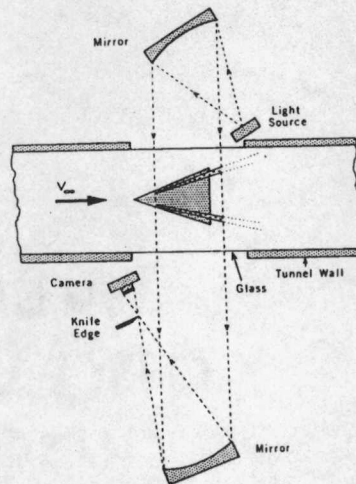


Figure 2.2. Glass Wind Tunnel Setup for Wentz' Delta Planform Study

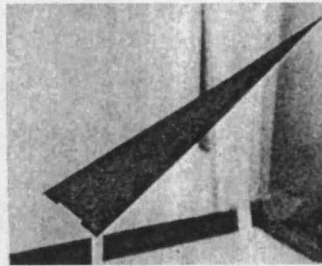


Figure 2.3. Model Installation in Wind Tunnel for Wentz' Study

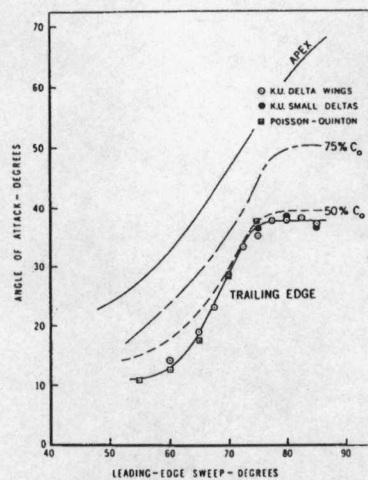


Figure 2.4. Effect of Sweep on Trailing Edge Breakdown Angle of Attack of Delta Shapes in Wentz' Study

Strakes Built Integrated into Wing

Studies were done on models with strake-integrated wings. A study was done at The Beijing Institute of Aeronautics and Astronautics in which tests on low and high-speed wing-strake and wing-alone configuration models were performed (Liu et al, 1979). The tests showed that having the strake added to the wing resulted in large increments in lift at moderate angles of attack. At an angle of attack of about 20 degrees, at Mach 0.1, the wings without strakes had a lift coefficient of about 0.8, compared to the wings with strakes, which had a lift coefficient of about 1.4. This is shown in Figure 2.5.

The strake vortex had great effects on the outboard wing, increasing its lift. At angles of attack of 15 and 18 degrees, there was a drastic increase in lift on the mid to outboard wing due to the addition of the strake. This is shown in Figure 2.6 where the pressure coefficients are plotted against the positions on the wing with and without strake. At the mid outboard wing, the pressure coefficient decreases drastically to a value above -2 as a result of the strake added to the wing.

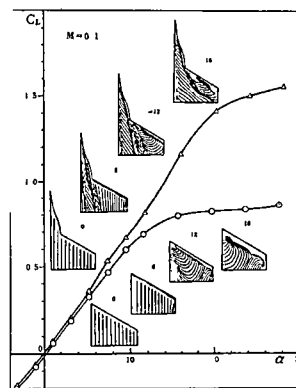


Figure 2.5 Lift Curves for Wing with and without Strakes According to Liu's Speed Tests

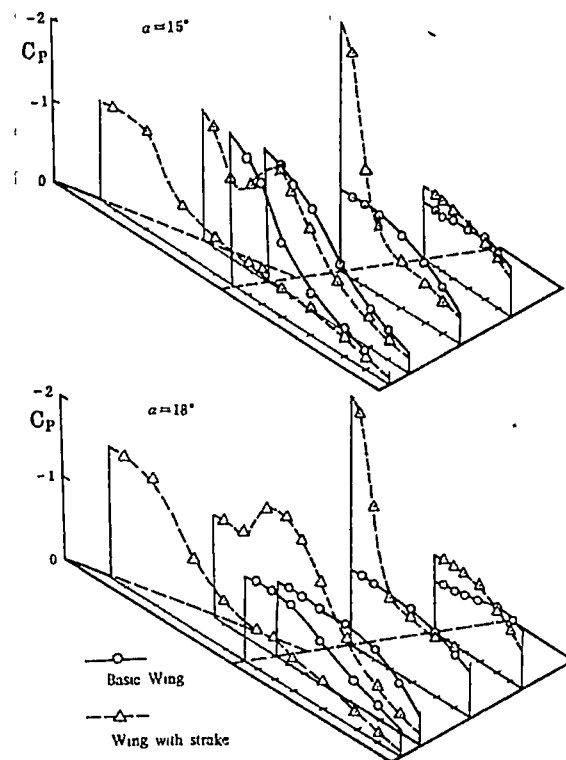


Figure 2.6. Effect of Strake on Pressure Distribution at Low Speed for Liu's Tests

The strakes also have had an effect on the lift to drag ratios in Liu's tests. The lift to drag ratio at high lift coefficients were improved. This is shown in Figure 2.7.

A study was done at NASA Langley Research Center in Hampton, Virginia by James Luckring, which showed that strakes provide substantial lift increments at maneuver angles of attack on fighter models, but had little influence on lift at low angles of attack (Luckring, 1979). Luckring also showed that strake span and wing sweep have an increasing effect on lift and drag characteristics. This is shown in Figure 2.8. Two sweep angles of 44 and 60 degrees are shown. More lift is generated with increasing strake span at sweep angles of 44 and 60 degrees. The strake swept at 60 degrees generates more lift with increasing strake span than the strake swept at 44

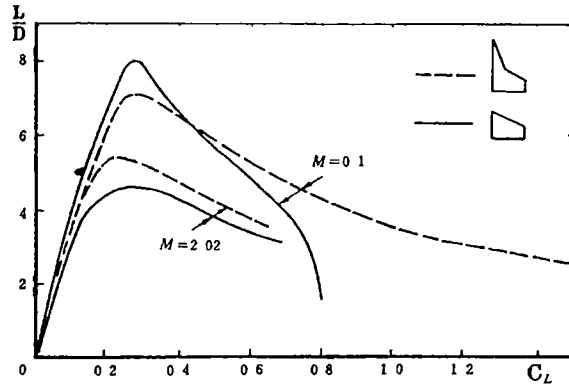


Figure 2.7 Lift to Drag Ratio of Liu's Study

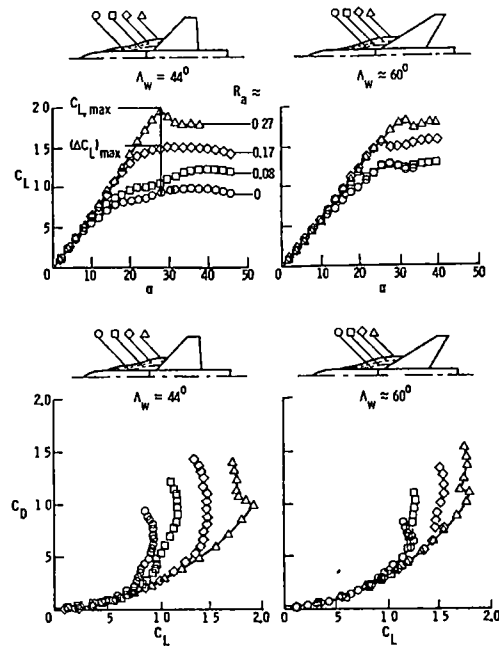


Figure 2.8. Effect of Strake Span and Wing Sweep on Lift and Drag Characteristics of Luckring's Study

degrees. The strake swept at 60 degrees also generates more drag than the strake swept at 44 degrees. Neal Frink of North Carolina State University and John Lamar of the NASA Langley Research Center in Hampton, Virginia analyzed the effects of different sizes and shapes of strakes (Frink and Lamar, 1981). A 16-x 24-inch water tunnel was used. The results showed that the strake vortex breakdown angle of attack increased with strake area.

Studies Done of Strakes Mounted Parallel to Wings

The studies just mentioned were performed with the strakes as an integral part of the wing. There were some studies performed in which the strakes were separate from the aircraft wing and were mounted parallel to it. This is illustrated in Figure 2.9. This figure shows the strakes being mounted parallel to the wings and the vortices produced which improves the flow in the typical regions of separated flow shown.

Thomas Wallis, while a student at Wichita State University, performed a study on a model of half of a twin-engine airplane using delta wings strakes in different overlap and gap configurations (Wallis, 1981). A sketch of this model is shown in

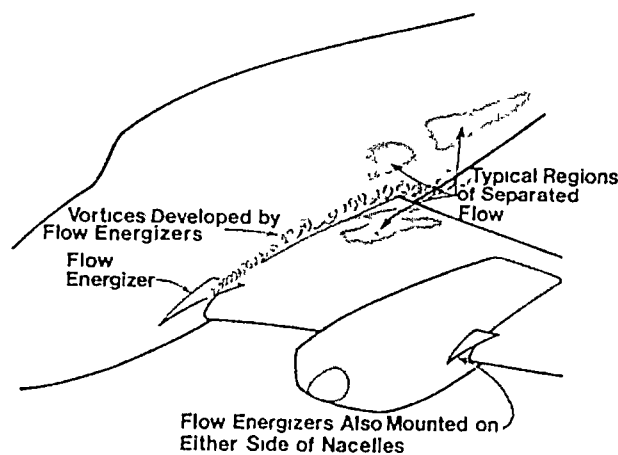


Figure 2.9 Diagram Showing Strakes Mounted Parallel to Wing of Twin Engine Airplane

Figure 2.10 Specifications for this model are shown in Appendix A A sketch of a delta strake used in the wind tunnel study that was used in this thesis is shown in Figure 2.11.

Overlap is how much of the strake extended aft of the wing airfoil leading edge Overlap is a very important factor because it controls where along the wing horizontally the vortex flow and local air flow mix. Gap is the distance from the strake's trailing edge to a point on the wing surface perpendicular to the chord line of the airfoil. Gap is an important factor because it controls how much of the vortex flow interacts with the local airflow over the wing A diagram illustrating overlap, gap and convergence ratio is shown in Figure 2.12.

Convergence ratio is the ratio of the gap height to the distance from the leading edge of the wing to the bottom of the strake Wind tunnel and flight tests of parallel mounted strakes were done at Texas A&M University, in which the best results came with low convergence ratios (Ward, 1985, 1987)

The studies in which Wallis performed also showed that the best results came when the strakes were cambered to align with the airflow Figure 2.13 shows a side

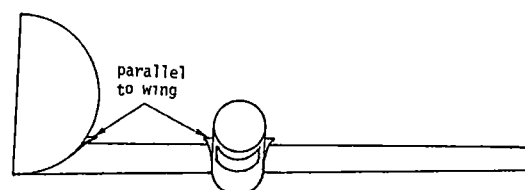


Figure 2.10. Half Twin-Engine Model with Strakes Mounted in Wallis' Study

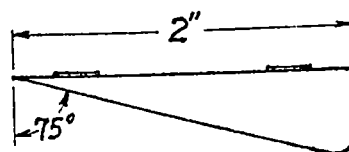


Figure 2.11. Picture of a Delta Strake used in Wallis' Study

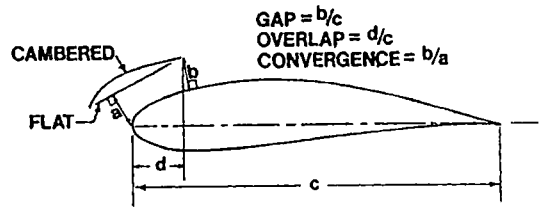


Figure 2.12. Picture Illustrating Overlap, Gap, and Convergence Ratio in terms of Wing Chord

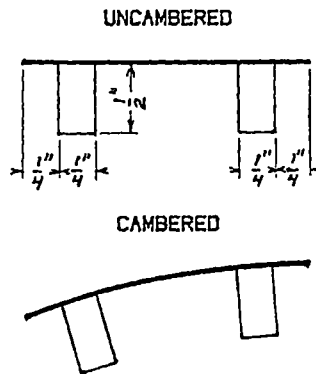


Figure 2.13 Side View of Cambered and Uncambered Strakes

view of cambered and uncambered strakes used in Wallis' study. Both overlap and gap were expressed in percent local wing chord. It was concluded that the two best configurations used were cambered strakes with 18% overlap, 2% gap and 12% overlap, 5% gap.

The strakes improved the lift and drag characteristics on the half twin engine model. The plot showing the drag coefficients versus lift coefficients for the baseline, 18% overlap, 2% gap and 12% overlap, 5% gap configurations is shown in Figure 2.14. Looking at this figure, it can be seen that the 18% overlap, 2% gap configuration did a slightly better job of improving the drag characteristics of the model than the 12% overlap, 5% gap configuration. The strakes enabled the model to reach a wider range of

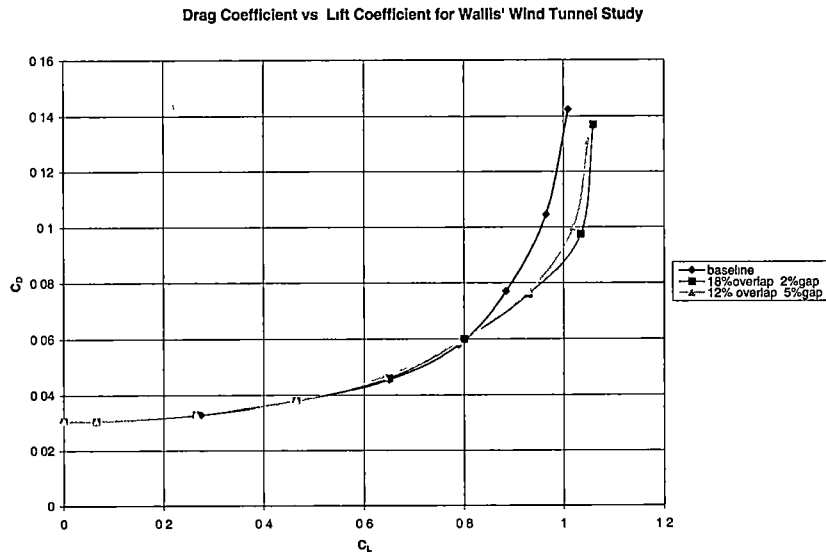


Figure 2.14 Drag Polars for the Configurations in Wallis' Study

lift coefficients than the baseline configuration. The 12% overlap, 5% gap configuration enabled the model to reach a maximum lift coefficient of 1.1 and the 18% overlap, 2% gap enabled it to reach a maximum lift coefficient of 1.12, whereas the model only reached a maximum lift coefficient of 1.02 in the baseline configuration.

At a lift coefficient of 1.0, the baseline drag coefficient was 0.13. The 18% overlap 2% gap drag coefficient was 0.086 at this lift coefficient, reducing the drag produced by 33.85%. The 12% overlap, 5% gap drag coefficient was 0.091 at this lift coefficient, reducing the drag produced by 30%.

The lift to drag ratio plot of the wind tunnel study also shows that the strake improved the performance characteristics. The lift to drag ratios of the baseline, 12% overlap, 5% gap and 18% overlap, 2% gap configurations also show that the strakes enabled the model to attain higher lift coefficients even though the maximum lift to drag ratio is decreased. Figure 2.15 shows that above a lift coefficient of about 0.82, the model

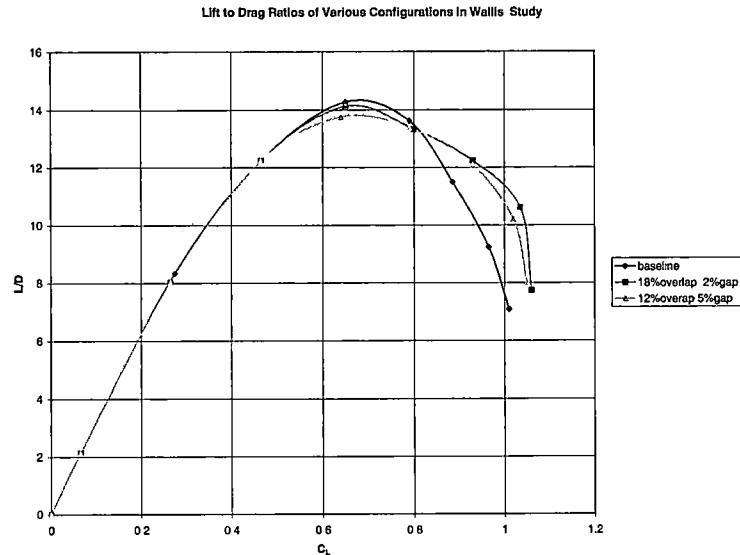


Figure 2.15 Lift to Drag Ratios for the Configurations in Wallis' Study

maintained a higher lift to drag ratio with the strakes attached than in the baseline configuration.

This plot also shows that the 18% overlap, 2% gap maintained a slightly higher lift to drag ratio than the 12% overlap, 5% gap. This further verifies that this is the best configuration for the model.

As a result of Wallis' study, the designers of the Cessna T303 Crusader twin engine airplane decided to use strakes with the 12% overlap, 5% gap configuration despite the wind tunnel study's showing that the 18% overlap, 2% gap configuration was better. This thesis discusses the effect of using delta shaped strakes in the 18% overlap 2% gap configuration as in Wallis' study in flight test on a Piper Navajo twin engine airplane.

CHAPTER III

EQUIPMENT USED

Aircraft

The aircraft used for this experiment was the Piper Navajo. A picture of this aircraft is shown in Figure 3.1. The Piper Navajo is a twin-engine airplane with a trapezoidal wing planform the span of 40.67 feet. It has a wing area of 229 square feet. The root and tip chords are 9.17 feet and 8.09 feet respectively. It has a mean aerodynamic chord of 5.63 feet. Specifications for the Piper Navajo are shown in Appendix A.

Strakes

Six delta shaped strakes were made for this experiment. A diagram of the specifications of the strakes is shown in Figures 3.2a-c. These strakes were made out of



Figure 3.1. Piper Navajo Twin-Engine Airplane

2024-0 aluminum, 0.063 inches thick. The strakes, when mounted on the airplane, were cambered in order to become aligned with the streamlined flow. This is per the specifications of the optimum strakes used in the study by Wallis. The only difference is that in Wallis' study; the aluminum was 2024 aluminum, 0.02 inches thick. This was the aluminum used on the model. Since this is a full-scale airplane in flight test thicker aluminum was used. Pictures of the strakes after being cut from the sheets of aluminum metal are shown in Figures 3.3a-c.

The design parameters called for the strake lengths to be 18% of the local wing chord. The chord lengths of the root, inboard nacelle-wing junction and outboard nacelle-wing junction were measured with measuring tape and came out to be 9.42, 7.17 and 6.21 feet respectively. The lengths of the strakes therefore were 1.7 and 1.29 feet for the root, inboard nacelle-wing junction and 1.12 feet for the outboard nacelle-wing junction strakes respectively.

The 2024-0 aluminum sheets were cut using a Sears Craftsman 12-inch belt drive band saw. Referring to the figure 3.2a, the strakes, which were mounted above the intersection of the wing and the fuselage, were cut out leaving 7 tabs for fastening onto the fuselage. These tabs, which were bent 90 degrees for fastening onto the airplane, extended perpendicular to the strakes. The tabs are 1 3/4 inches long, 1 1/2 inches wide. The trailing edges of these strakes were 6 inches wide.

The specifications of the strakes, which were mounted above the intersection of the wings and nacelles, are also referred to in figure 3.2b. These strakes were cut out with 4 tabs for fastening onto the left and right nacelles. The first 3 tabs were 1 1/4

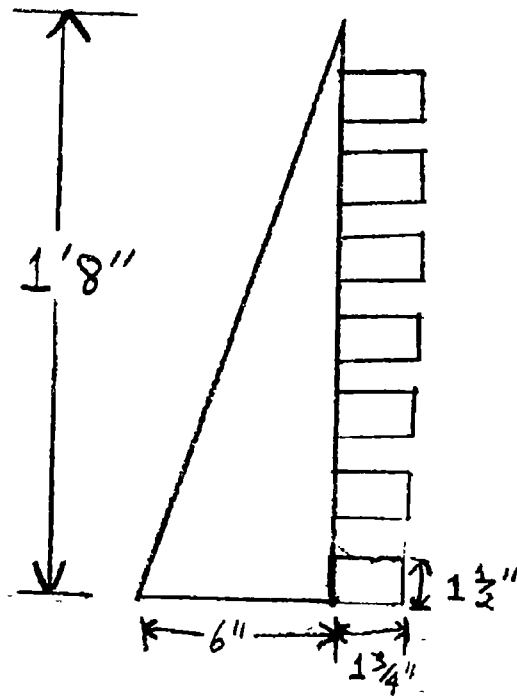


Figure 3.2a Specifications of Fuselage Strake for Piper Navajo

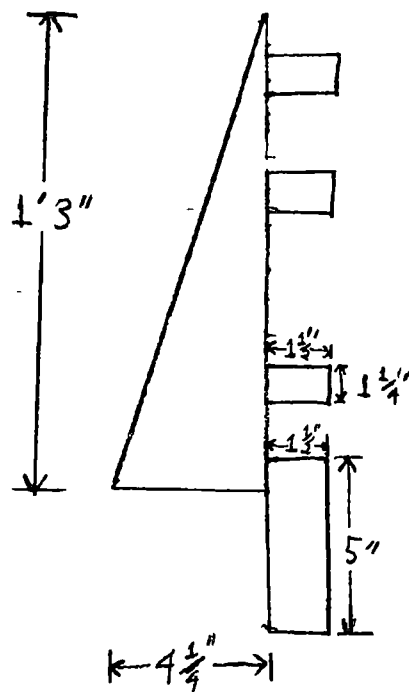


Figure 3.2b Specifications of Inboard Nacelle Strake for Piper Navajo

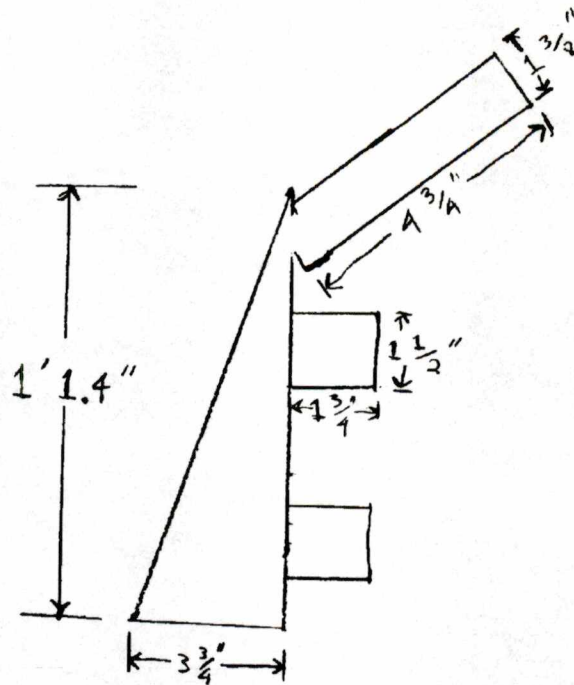


Figure 3.2c. Specifications of Outboard Nacelle Strake for Piper Navajo

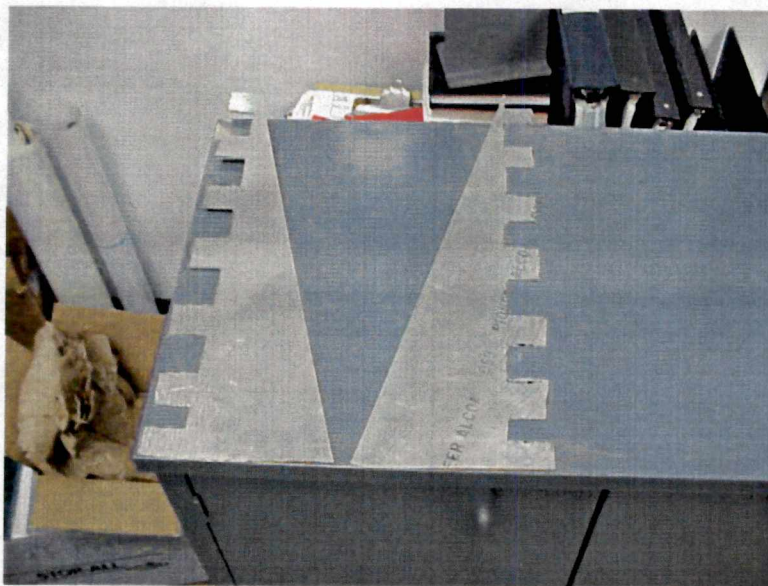


Figure 3.3a – Fuselage-Wing Intersection Strakes

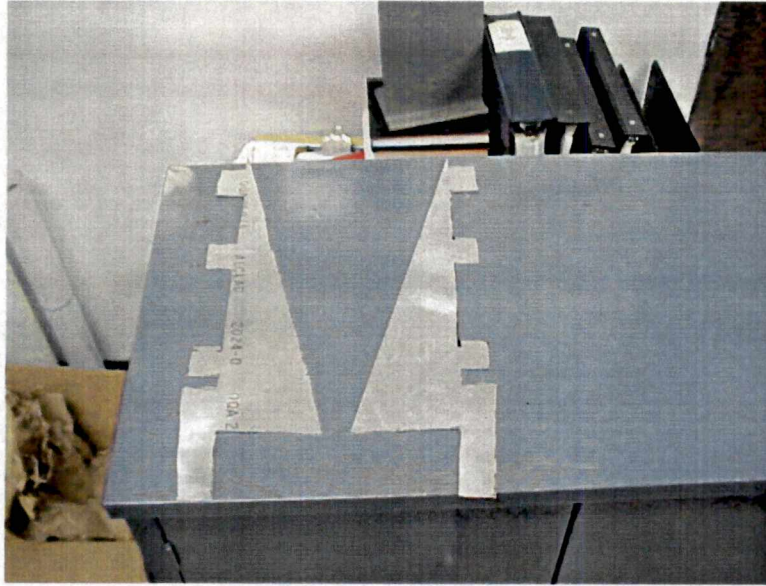


Figure 3.3b. Inboard Nacelle-Wing Intersection Strakes



Figure 3.3c. Outboard Nacelle-Wing Intersection Strakes

inches long, 1 1/2 inches wide, and they protruded perpendicular to the strakes. The fourth tab was longer than the first 3 and it protruded parallel to the strake, unlike the first 3 that protruded perpendicular to the strake. They were cut out this way in order to reach the closest screw behind where the strake was mounted. This tab was 5 inches long, 1 1/2 inches wide. The trailing edges of the strakes were mounted above the inboard nacelle-wing intersections and are 4 1/4-inches wide.

The specifications of the strakes that were mounted above the intersection of the wings and the outboard nacelles are shown in Figure 3.2c. These strakes were cut out with 3 tabs for fastening onto the left and right nacelles. The first 2 tabs were 1 1/2 inches long, 1 3/4 inches wide, and they protruded perpendicular to the strakes. There was one long tab, which was cut to diagonally protrude from alongside the front of the strake. This long tab's purpose was to reach the closest screw ahead of where the strake was mounted. This long tab was 4 3/4 inches long, 1 3/8 inches wide. The trailing edges of the strakes were mounted above the left and right outboard nacelle-wing intersections and were 3 3/4-inches wide.

The strakes were mounted on the airplane in order to make a gap height that is 2% of the local wing chord. The strakes, which were mounted to the fuselage above the wing root, were therefore mounted with the strake trailing edges 2.2 inches above the wing surface. The strakes, which were mounted to the inboard nacelles, were mounted with the strake trailing edges 1.72 inches above the wing surface. The strakes, which were mounted to the right and left outboard nacelles, were mounted with the strake trailing edges 1.49 inches above the wing surface.

Tufts

Tufts made of black yarn were taped to the wing between the fuselage and the left and right nacelles. Removable duck tape was used to attach the tufts. Tufting was done so that the patterns of the flow over the wings could be seen visually. The tufts of yarn were spread 6 inches apart. They were lined in rows of five and the rows started from the leading edge of the wings and went back to the wings' trailing edge. This is shown in Figures 3.4a-b.

Measurement Instruments

Instruments in the aircraft were used to measure the test parameters needed. The instruments used to measure the test parameters are located on student consoles, installed by the University of Tennessee Space Institute Flight Research Group for flight test purposes.

The instruments on the student consoles were used instead of the instruments in the cockpit because they are calibrated. Each student console has an airspeed indicator, an altimeter, and manifold pressure indicator, digital RPM indicators for the left and right engines, a digital ambient air temperature indicator, and a voltmeter, which works with a switch to provide various other parameters. The instruments on the center left student console of the Piper Navajo, with the exception of the voltmeter, were used to obtain the data in this thesis and it is shown in Figure 3.5.

1. Airspeed Indicator

The airspeed indicator measures the aircraft's speed in knots. Airspeed indicators are fixed so that they show what the airspeed would be if the aircraft was at standard sea

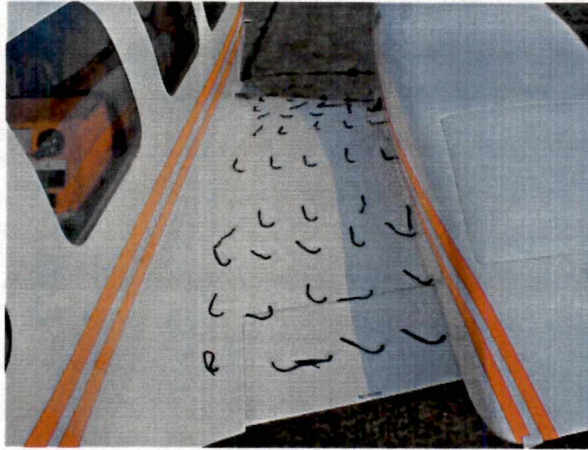


Figure 3.4a. Tufts on Right Inboard Wing between Nacelle and Fuselage



Figure 3.4b. Tufts on Left Inboard Wing between Nacelle and Fuselage



Figure 3.5. Center Left Student Console in the Piper Navajo

level conditions.

Instrument correction charts are used to account for the error in the instrument. Corrections are also made for position error. This position error is due to the position of the static ports on the aircraft. Position error charts are used along with instrument correction charts in data analysis to show the aircraft's calibrated airspeed. A picture of an airspeed indicator in the Piper Navajo is shown in Figure 3.6. Instrument and position correction charts for the center left seat in the Piper Navajo are shown in Appendix B.

2. Altimeter

The altimeter measures the altitude of the aircraft. In a flight test it is adjusted to 29.92 inches of mercury, which is standard sea level pressure. A picture of an altimeter is shown in Figure 3.7. Instrument correction charts were used along with the altimeter in data analysis and an altimeter instrument correction chart for center right seat of the Piper Navajo is shown in Appendix B.

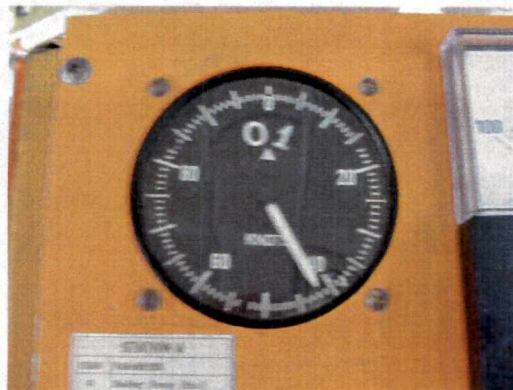


Figure 3.6. Airspeed Indicator for Center Left Console in the Piper Navajo



Figure 3.7. Altimeter for Center Left Console in the Piper Navajo

1. Manifold Pressure Indicator

The manifold pressure indicator measures the manifold pressure of the engine. A picture of the manifold pressure indicator for the cockpit of the Piper Navajo is shown in Figure 3.8. A correction chart for the manifold pressure indicator is also shown in Appendix D.

2. RPM Indicator

The digital left and right engine RPM indicators measure the rotations of the propeller per minute. The RPM, along with the manifold pressure can be used to determine the power required for an airplane. A picture of the digital RPM indicators



Figure 3.8. Manifold Pressure Indicator for Center Left Console in Piper Navajo

in the center left student console for the left and right engines of the Piper Navajo is shown in Figures 3.9a-b.

3. Fuel Quantity Indicator

The fuel quantity indicator measures how many gallons of fuel have been burned. A picture of this is shown in Figure 3.10. This enables one to either determine the aircraft's weight at a particular time; by multiplying the amount of gallons burned by six pounds per gallon and subtracting it from the aircraft beginning weight or to determine how much fuel is remaining.

3. Ambient Temperature Indicator

The ambient temperature indicator measures the ambient air temperature outside of the aircraft. A picture of the digital ambient temperature indicator on the center left student console in the Piper Navajo is shown in Figure 3.11. The ambient air temperature must



Figure 3.9a. Left Digital RPM Indicator for Center Left Console in the Piper Navajo

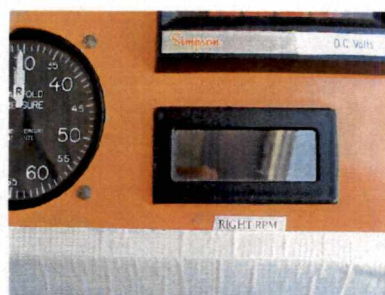


Figure 3.9b - Right Digital RPM Indicator for Center Left Console in Piper Navajo



Figure 3.10. Fuel Quantity Indicator in Cockpit of Piper Navajo

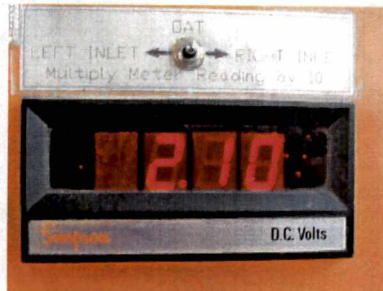


Figure 3.11. Ambient Temperature Indicator for Center Left Console in the Piper Navajo

be known in order to determine temperature and density ratios, which are used in aircraft performance analysis. The number shown in the display is multiplied by 10 and then temperature corrected using a temperature correction chart for the student consoles, shown in Appendix D.

Camera

A camera is used to take pictures of the tufts taped to the wing. Pictures are taken at various speeds in order to visualize the flow.

CHAPTER IV

FLIGHT TEST

This chapter discusses the two flight test procedures used in evaluating the strakes. Four flights were required to complete the evaluation. In this experiment, the level flight performance technique was used to evaluate the performance of the aircraft. The level flight performance method enables one to plot a power-required curve. This method also makes it possible to obtain lift and drag coefficients. These lift and drag coefficients can then be compared with the strakes on and off.

In the second procedure, black yarn tufts are taped on to the wing between the fuselage and the left and right nacelles from the point of maximum thickness to the trailing edge of the wings. This was shown in Figures 3.4a-b. The tufts were only used on the second and forth flights. No yarn was placed on the wings between the nacelles and the wing tips for two reasons. One reason is that these portions of the wings cannot be seen in flight due to being blocked by the nacelles. Secondly, it was determined that it is better to examine the flow over the wings at the wing-fuselage, wing-nacelle junctions, the regions that develop the most interference drag.

Weight and Balance

The Federal Aviation Regulations (FAR), Part 23 requires that the aircraft be loaded to certain weights and centers of gravity (c.g.) for certain tests. For performance testing, it is required that the aircraft be loaded at maximum takeoff gross weight at the most forward c.g. The certified maximum takeoff gross weight for the Piper Navajo is 6500 pounds. At this weight, the most forward c.g. is 134 inches aft of datum. Weight

and balance charts showing the actual takeoff weights of the aircraft and center of gravity positions within the envelope for the Piper Navajo for the flight tests are shown in Appendix D.

Baseline Flight Test

1. Level Flight Performance

This test was done to determine the airplane's performance in its clean baseline configuration. The data obtained during this part of the flight test was used to produce a power-required curve for the aircraft in its baseline configuration. After takeoff, the airplane was stabilized in level flight at its maximum airspeed. After the aircraft was stabilized, test parameters were recorded. These parameters were the observed airspeed, observed pressure altitude, observed outside air temperature, observed engine RPM, observed manifold pressure and fuel quantity. The power was then decreased and the same data is again recorded. This procedure was repeated until the minimum level flight speed was reached. A table showing these test parameters for the baseline configuration is shown in Appendix E.

2. Second Flight: Tufts Mounted

The second flight was performed with wing tufts mounted on the left and right wings between the fuselage and the nacelles. The reason why the tufts were not applied in the first flight was because the tufts add a slight drag to the aircraft and this will make the measurements slightly inaccurate. In this flight, the plane was trimmed and then flown at various speeds ranging from close to the stall speed to close to the maximum level

flight speed. Pictures are taken of the tufts at these various airspeeds for flow visualization.

Flight Test with Strakes Mounted

1. Mounting the Strakes

The strakes mounted on the fuselage and the nacelles are shown in Figures 4.1a-4.1f. The last two tabs before the trailing edges of the strakes to be mounted on the left and right sides of the fuselage were cut in order to enable them to adjust to the curvature of the fuselage. The portions on the top and bottom parts of the strakes that touched the airplane were taped using removable duck tape, as shown in the figures, to prevent air from leaking in the gaps between the strakes and the fuselage and nacelles during flight.

The strakes were mounted in the 18% overlap, 5% gap configuration. The distance from the strake leading edges to the aircraft wings' leading edges were recorded. In this way the convergence ratio can be known. Table 4.1 shows the overlap and gap distances in feet, the distances from the strake leading edge to the wing leading edge in feet and the convergence ratios.

2. Level Flight Performance

A level flight performance test was done to compare the airplane's performance with the strakes mounted to when it was in the baseline configuration. Comparing the power-required curves produced for the baseline aircraft and the aircraft with strake configurations did this. A table showing the test parameters obtained in the strake configuration is shown in Appendix E.



Figure 4.1a. Right Fuselage Strake Mounted to Right Fuselage (blue arrows point to strake)

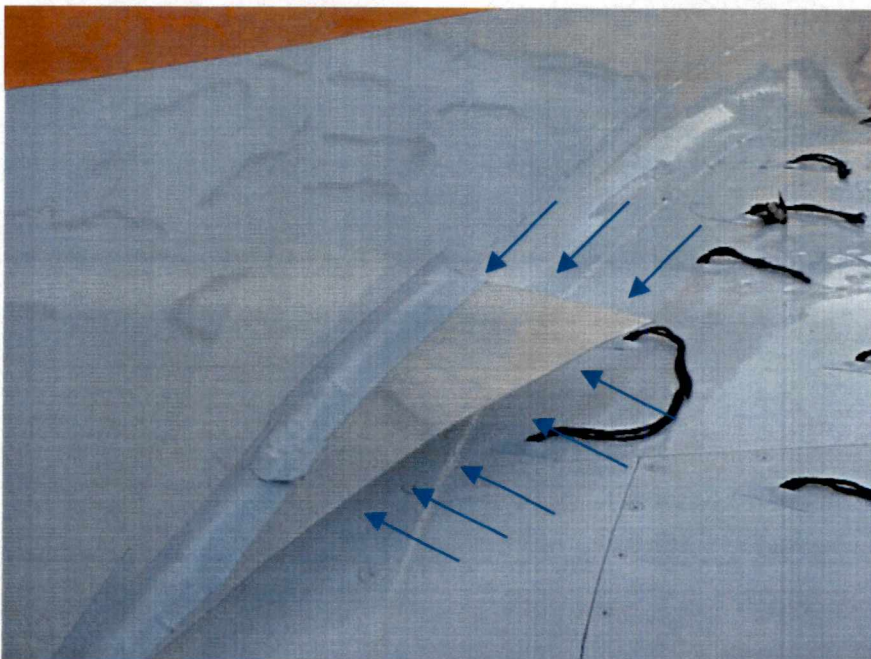


Figure 4.1b. Left Fuselage Strake Mounted to Left Fuselage (blue arrows point to strake)

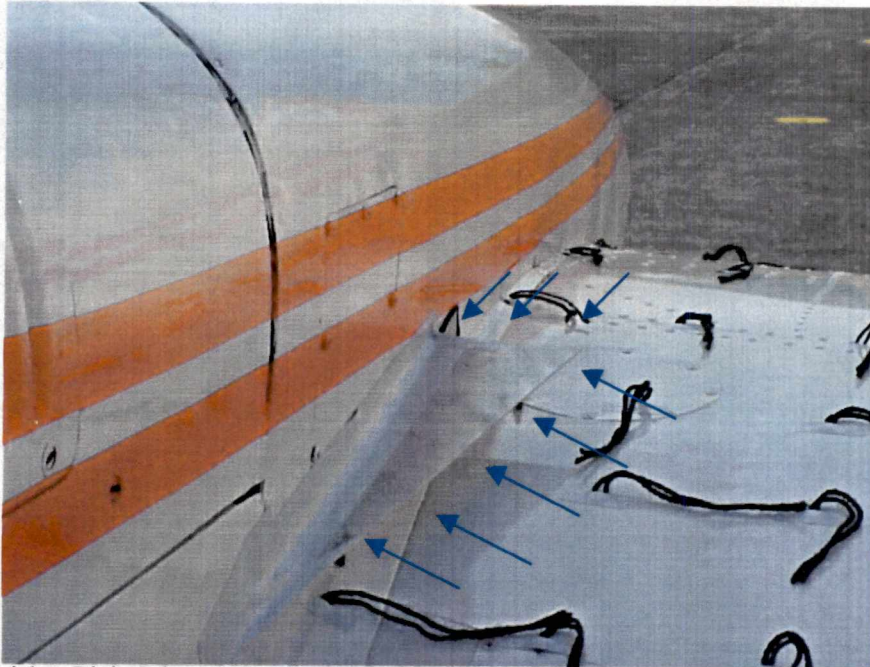


Figure 4.1c. Right Inboard Nacelle Strake Mounted to Right Inboard Nacelle (blue arrows point to strake)

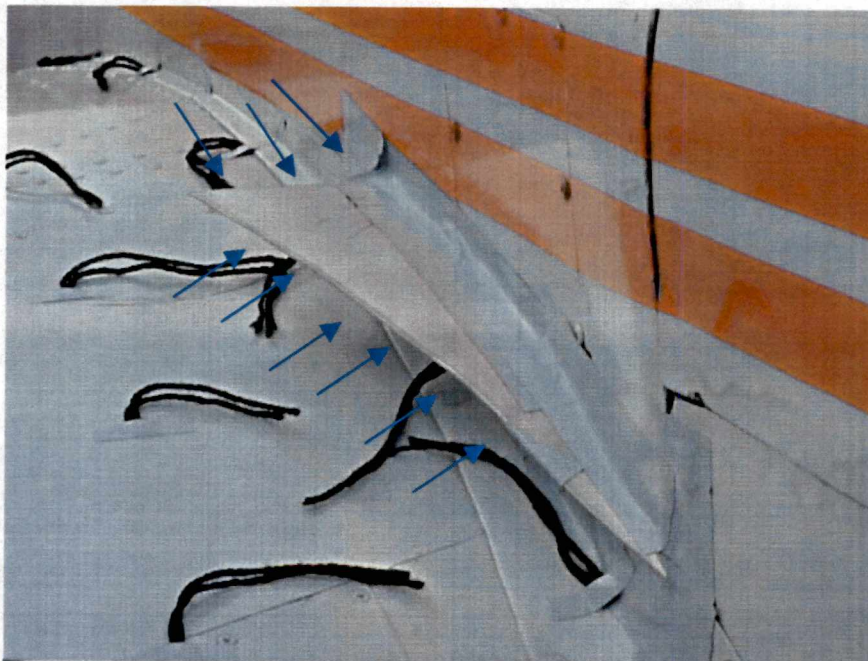


Figure 4.1d. Left Inboard Nacelle Strake Mounted to Left Inboard Nacelle (blue arrows point to strake)

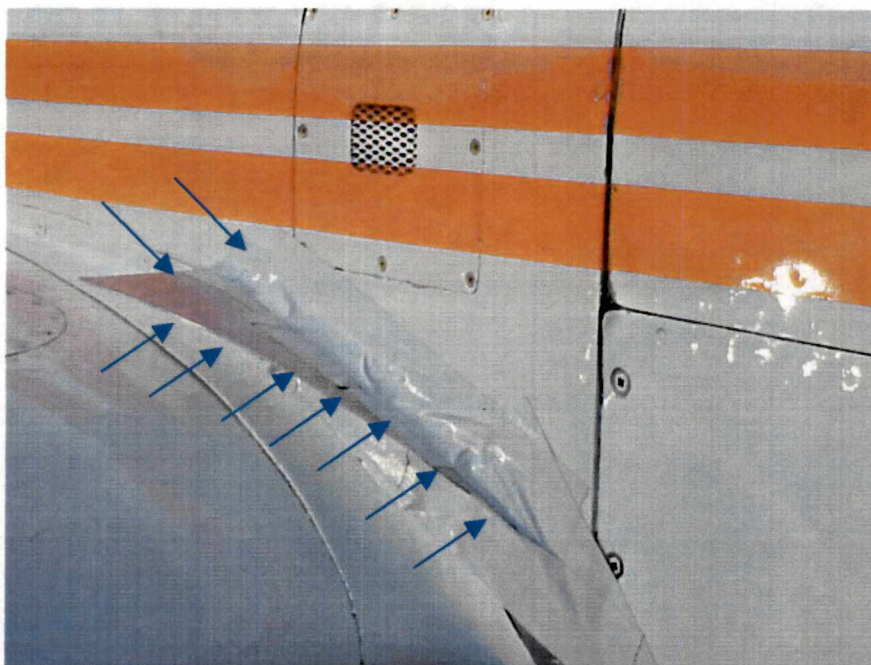


Figure 4.1e. Right Outboard Nacelle Strake Mounted to Right Outboard Nacelle (blue arrows point to strake)

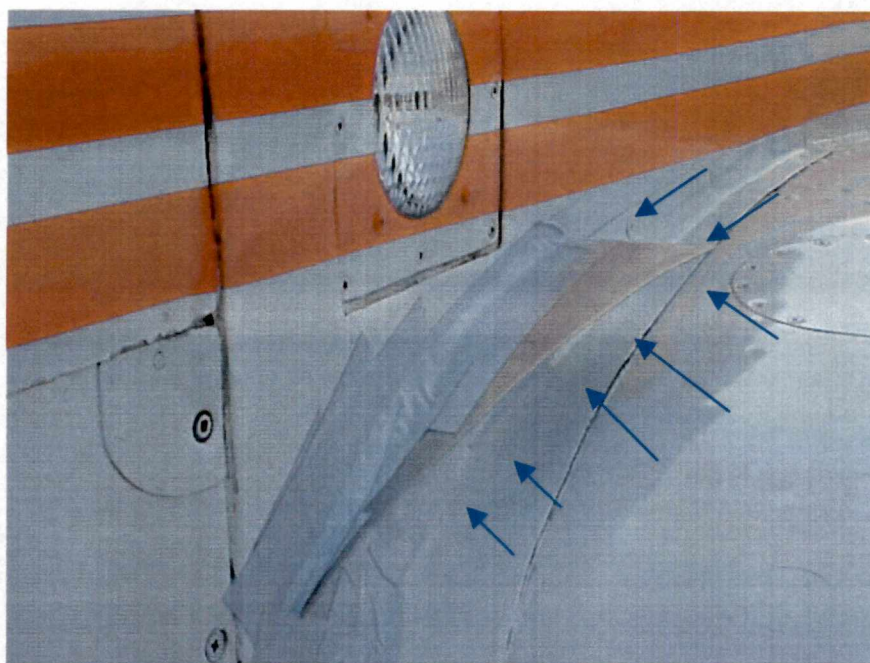


Figure 4.1f. Left Outboard Nacelle Strake Mounted to Left Outboard Nacelle (blue arrows point to strake)

Table 4.1. Mounting Distances for 18% Overlap, 5% Gap Configuration on Piper Navajo

Location	Local Wing Chord (ft.)	18% Overlap Distance (ft.)	2% Gap Distance (ft.)	Distance from Wing Leading Edge to Strake (ft.)	Convergence Ratio
Root	9.42	1.7	0.188	0.229	0.821
Inboard Nacelle	7.17	1.29	0.143	0.25	0.572
Outboard Nacelle	6.21	1.12	0.124	0.281	0.441

3. Fourth Flight: Tufts and Strakes Mounted

The fourth flight with the tufts and strakes mounted together was performed using the same procedure as the second flight with the tufts mounted. Pictures are taken of the tufts to see how the strakes affect the flow in the regions between the nacelles and the fuselage. These pictures were compared with the pictures of the tufts alone. Pictures of the regions between the left and right nacelles and the fuselage with the strakes and tufts applied are shown in Figures 4.2a-b.



Figure 4.2a. Right Wing with Tufts and Strakes Mounted



Figure 4.2b. Left Wing with Tufts and Strakes Mounted

CHAPTER V

DATA REDUCTION METHODOLOGY

Level Flight Performance-Power Required Curve

The method used in obtaining the power-required curves for the strakes on and strakes off configurations are now discussed. The power required curves are obtained using a method called the PIW-VIW method. In this method, the power required curves are single curves good for all air densities and airplane weights. Two assumptions are made with the PIW-VIW method. The first assumption is that at a given angle of attack the lift coefficient and the drag coefficient are constant. The second assumption is that the propeller efficiency is constant. A PIW-VIW plot for a Cessna U3A is shown in Figure 5.1. The equations for the PIW and VIW calculations are shown in Appendix C.

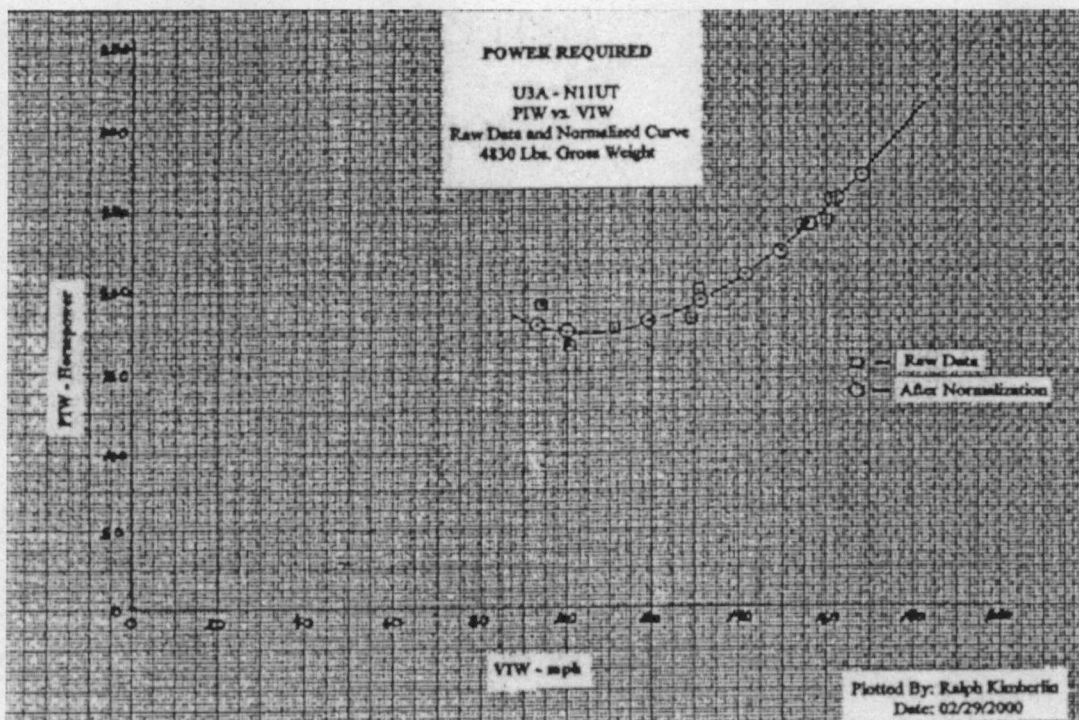


Figure 5.1. PIW-VIW Plot for a Cessna U3A

To obtain the power required curves the first step is to reduce the level flight performance data to PIW and VIW values using the equations in Appendix C. The PIW values are then plotted versus the VIW values. PIW times VIW versus VIW^4 is then plotted and a straight line is faired through the plot. This is a normalization technique that makes it easier to fair the scattered data. A normalization plot for the Cessna U3A is shown in Figure 5.2. The straight line acquired is then reduced to values of PIW and VIW and these values are plotted on the PIW versus VIW plot. Looking at this data the maximum range speed and maximum endurance speed can then be determined.

The maximum range speed is the speed that corresponds to the maximum lift to drag ratio. The maximum range speed is first found by extending a straight line from the origin to a point tangent to the PIW-VIW curve. The corresponding VIW speed to this point is the maximum range speed. The maximum endurance speed is the speed

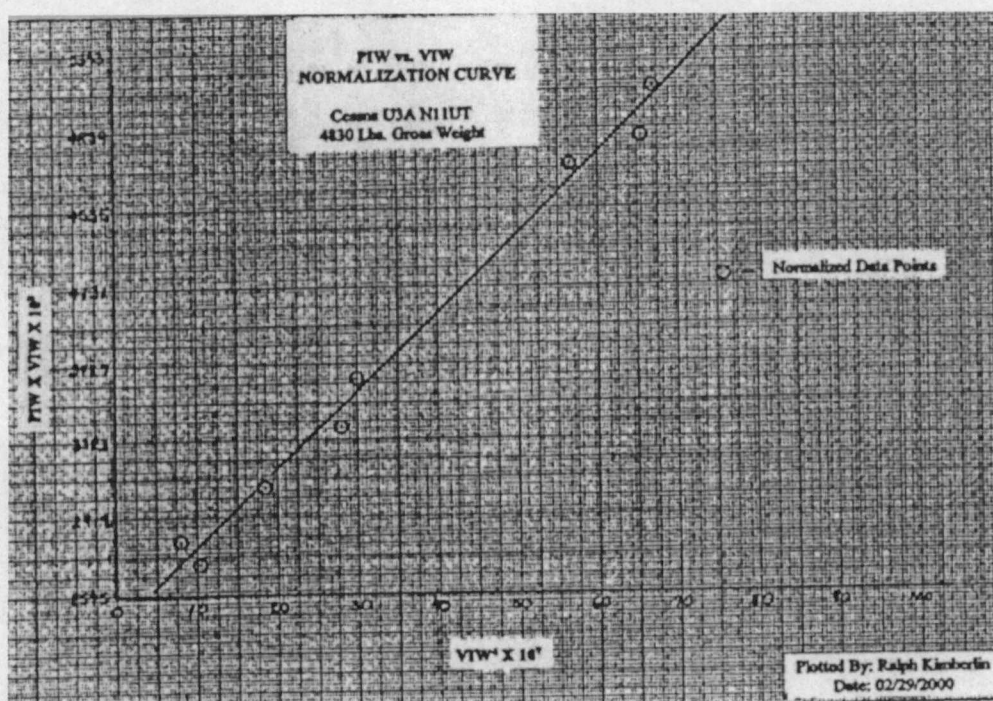


Figure 5.2. Normalization Plot for Cessna U3A

that corresponds to the minimum power required on the PIW-VIW curve. The minimum power required is obtained by drawing a straight-line tangent to the bottom portion of the PIW-VIW curve. The corresponding speed is the maximum endurance speed. These procedures for finding the maximum range and maximum endurance speeds are illustrated in Figure 5.3.

Drag and Lift Coefficients

In obtaining both the lift and drag coefficients, the values obtained previously from the PIW-VIW plot are used. The method for obtaining the lift and drag coefficients are shown in Appendix C. The lift and drag coefficient values are then used to produce plots of drag coefficient versus lift coefficient (drag polar), drag coefficient versus C_L^2 and lift to drag ratio versus lift coefficient.

The drag polar shows how the drag varies with lift. The drag coefficients are simply plotted versus their corresponding lift coefficients. Typically the plot is a polar curve. A typical drag polar curve is shown in Figure 5.4. A C_D versus C_L^2 curve is also plotted

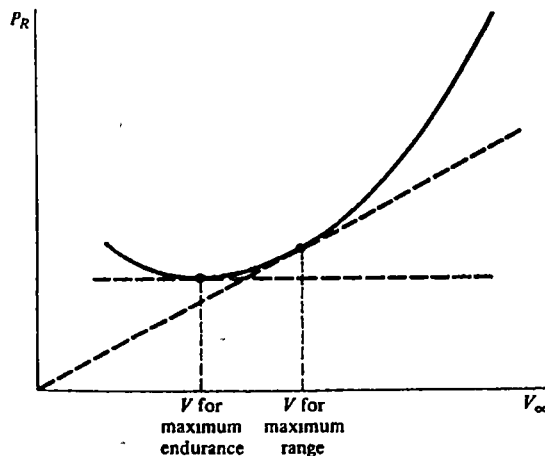


Figure 5.3. Points of Maximum Range and Endurance on a Power Required Curve of a Propeller Airplane

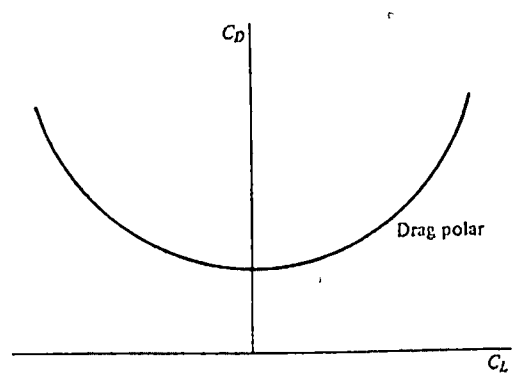


Figure 5.4. Typical Drag Coefficient versus Lift Coefficient Plot (Drag Polar)

to show how the drag varies linearly with lift. One of the things that this plot reveals is C_{D0} , the drag coefficient at zero lift. This plot also reveals Oswald's efficiency factor, e , which is the inverse of the slope of the line multiplied by the aspect ratio and by the mathematical constant π . The equation to compute this is shown in Appendix C. A plot of C_D versus C_L^2 for a Cessna U3A is shown in Figure 5.5.

Lift to Drag Ratio

The lift to drag ratio versus lift coefficient plot is obtained by dividing the lift coefficients by their corresponding drag coefficients. These are then plotted versus lift coefficient. Sample lift to drag curves for commercial aircraft in different flight configurations are shown in Figure 5.6. The speed corresponding to the maximum lift to drag on the plot should be the same as or very close in value to the speed where a straight line from the origin on the $PIW-VIW$ curve plot tangentially touches the $PIW-VIW$ curve. This speed is the speed for maximum lift to drag or better known as the speed for maximum range.

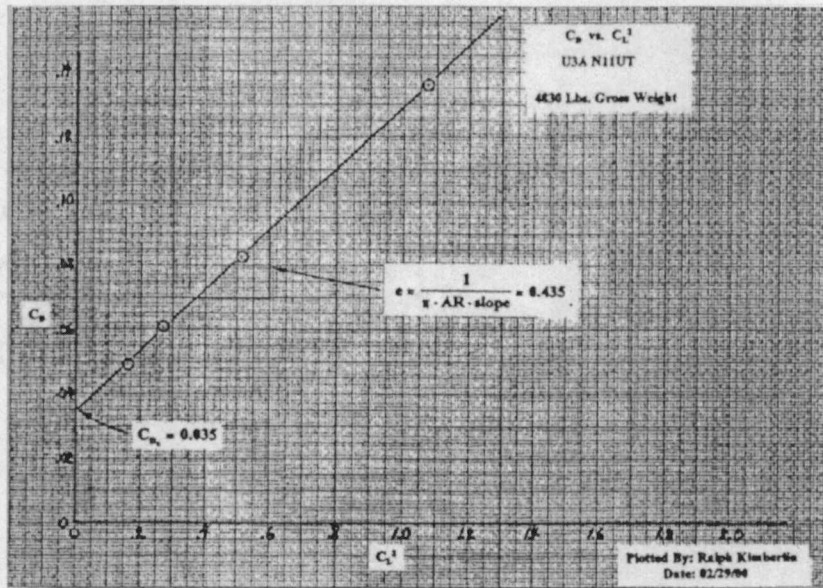


Figure 5.5. C_D versus C_L^2 for a Cessna U3A

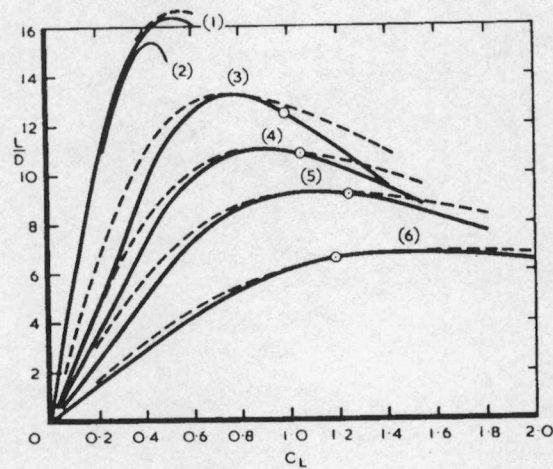


Figure 5.6. Lift to Drag Ratios of Commercial Aircraft in Different Flight Configurations

Each of these plots are produced using the data attained in flight with the strakes on and off. The plots are compared and analyzed to see what kind of effect the strakes had on each of these parameters.

The changes observed in these graphs are also compared with the changes observed in the graphs obtained during Wallis' study.

The changes in lift to drag are compared with Wallis' study to analyze the delta strakes' performance in actual flight as compared to what was obtained in Wallis' wind tunnel study. They are also compared to analyze the 18% overlap, 2% gap configuration used on the Piper Navajo in actual flight test with respect to this same configuration and the 12% overlap, 5% gap configuration used on the model in Wallis' wind tunnel study.

Analysis of the Tuft Pictures

The pictures taken of the tufts at the six different speeds of 130, 120, 110, 100, 90 and 80 knots are analyzed. Things to observe are when the tufts stop flowing straight and

neat When the tufts stop flowing straight and neat, it is a sign that the flow is starting to separate. It is important to observe the airspeeds in which this starts to happen. The pictures obtained with the strakes on and off are compared to observe and analyze the effect the strakes have on when and where along the wing the flow starts to separate.

CHAPTER VI

DATA ANALYSIS

Baseline Configuration

1. Power Required Curve

The PIW-VIW plot for the baseline configuration of the Piper Navajo is shown in Figure 6.1. This plot shows a maximum range speed of approximately 124 knots and a maximum endurance speed of approximately 89 knots. The power required for this maximum range speed is 282.2 horsepower. The minimum power required for this baseline configuration corresponding to the maximum endurance speed is 238.57 horsepower. The PIW times VIW versus VIW^4 for the Piper Navajo baseline configuration is shown in Figure 6.2.

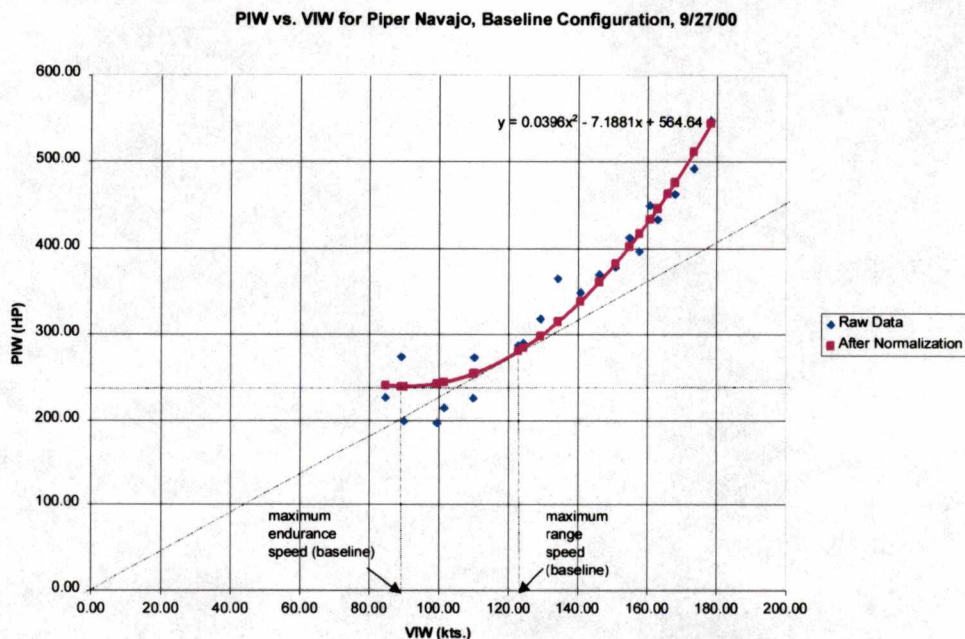


Figure 6.1. Power Required Curve for the Piper Navajo, Baseline Configuration

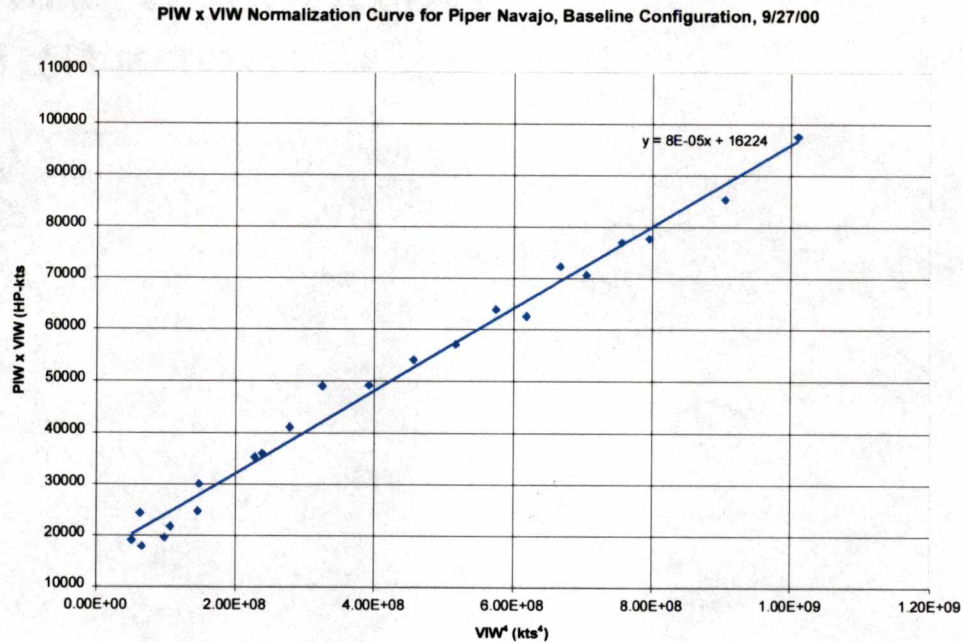


Figure 6.2. Normalization Plot to Obtain Power Required Curve for Piper Navajo, Baseline Configuration

2. Drag Polar and Drag Coefficient vs. C_L^2

The drag polar for the Piper Navajo baseline configuration is shown in Figure 6.3. As with other typical drag curves, this curve turns out to be a polar plot. This plot shows that the Piper Navajo in a clean configuration has a zero lift drag coefficient of 0.0278. At a lift coefficient of 1.17, the airplane has a drag coefficient of approximately 0.139.

The drag coefficient versus C_L^2 plot is shown in Figure 6.4. It is in this chart where it is determined that the drag coefficient with zero lift is calculated to be 0.0278. The Piper Navajo has an aspect ratio (b^2/S) of 7.22. The slope of this curve is 0.0806, giving an Oswald's efficiency factor value of 0.547.

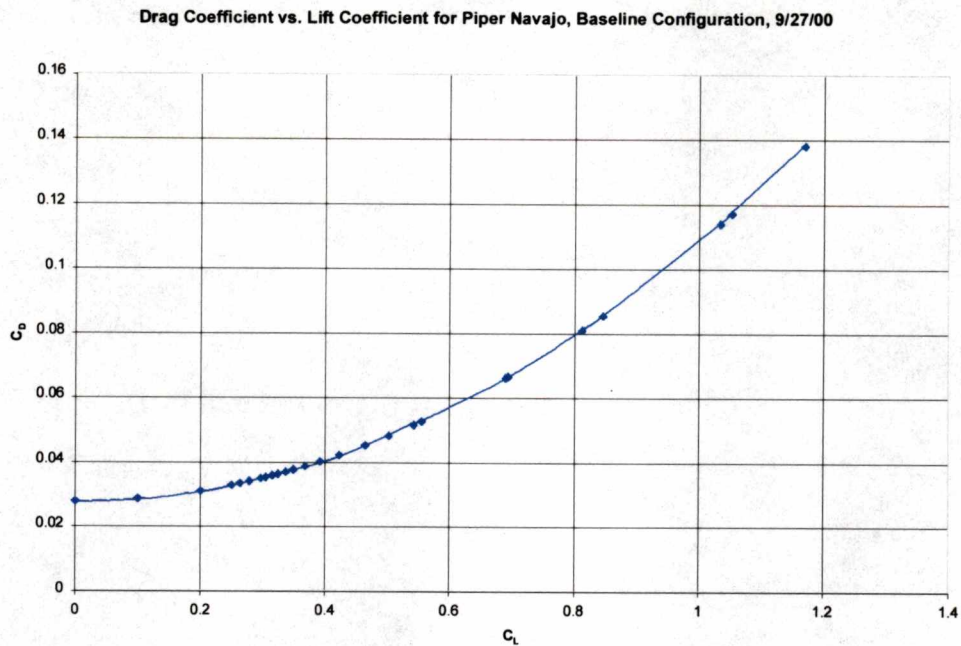


Figure 6.3. Drag Polar for the Piper Navajo, Baseline Configuration

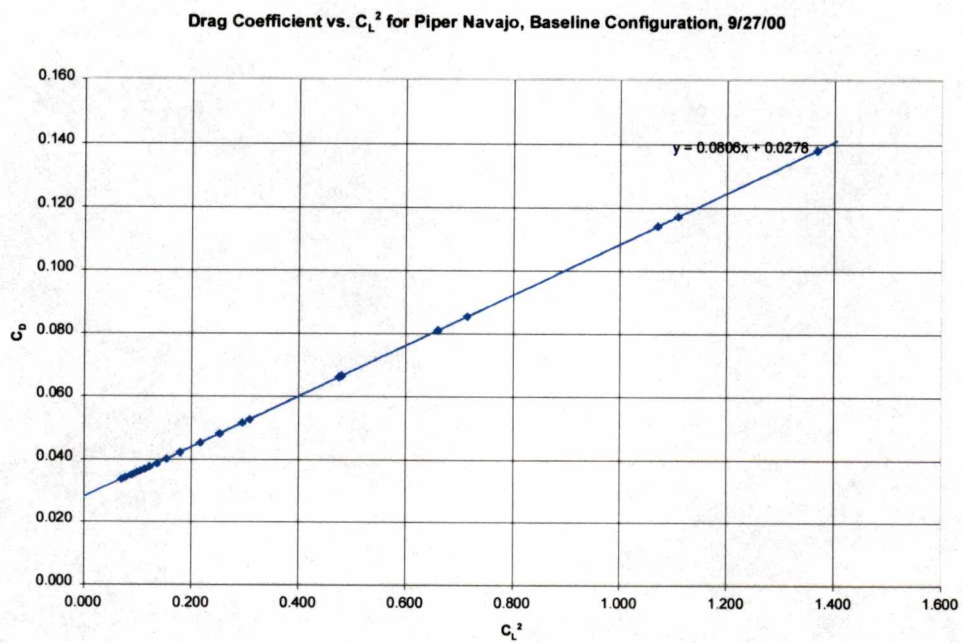


Figure 6.4. C_D versus C_L^2 for the Piper Navajo, Baseline Configuration

3. Lift to Drag Ratio

A plot of the lift to drag ratio for the Piper Navajo baseline configuration is shown in Figure 6.5. This plot shows that the Piper Navajo attains a maximum lift to drag ratio of 10.53 at a lift coefficient of 0.556. Taking the equation for lift coefficient and rearranging it to solve for airspeed, using the standard takeoff gross weight of 6500 pounds gives an airspeed of 122.7 knots. This value agrees very well with the maximum range airspeed of 124 knots determined in the PIW-VIW graph.

Strake Configuration

1. Power Required Curve

The PIW-VIW plot for the Piper Navajo with the strakes installed is shown in Figure 6.6. This plot shows a maximum range speed of approximately 117.5 knots and

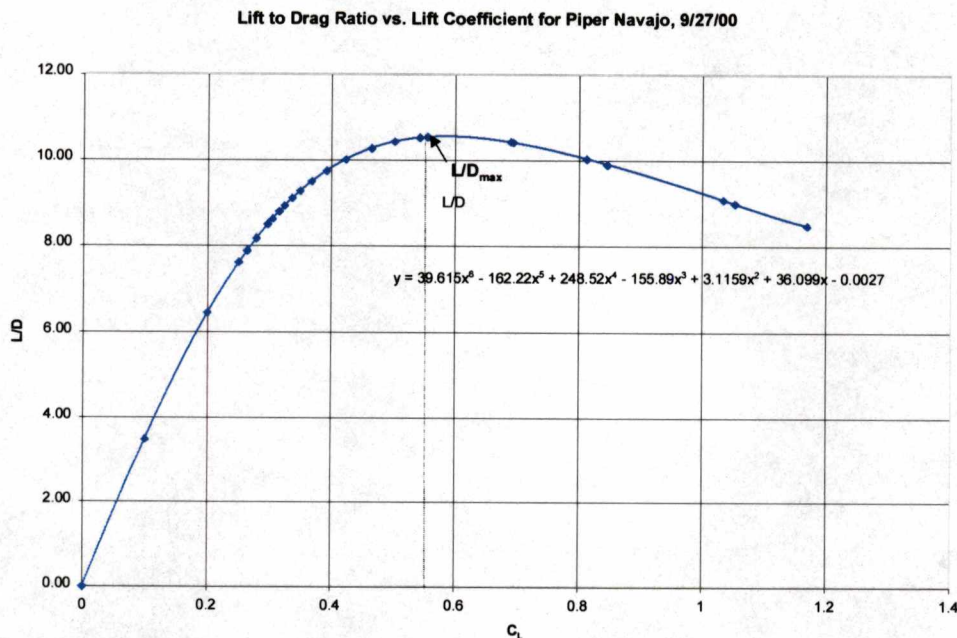


Figure 6.5. Lift to Drag Ratio of Piper Navajo, Baseline Configuration

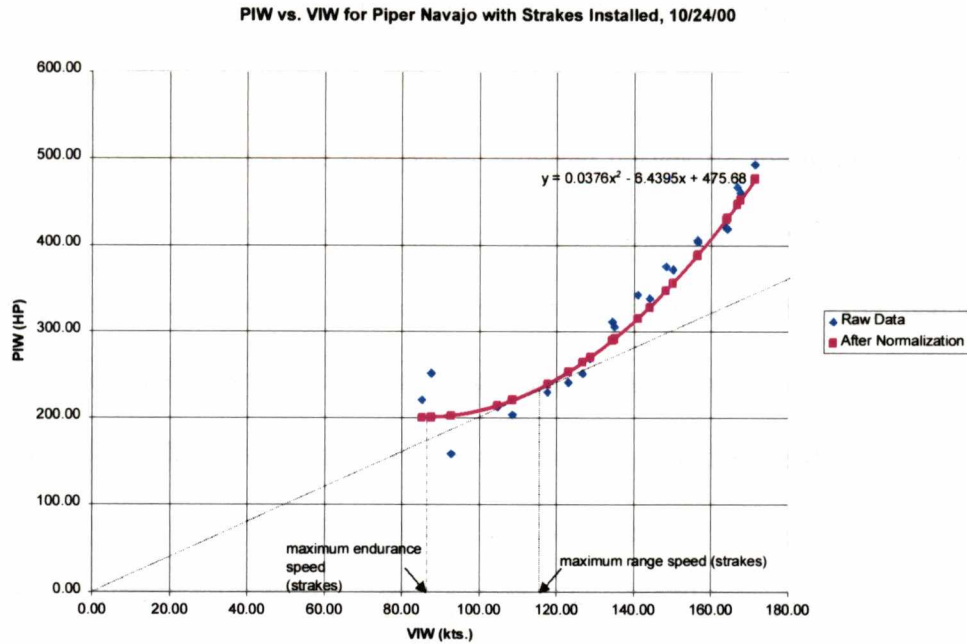


Figure 6.6. Power Required Curve for the Piper Navajo with Strakes Installed

a maximum endurance speed of approximately 87 knots. The power required for this maximum range speed is 238.15 horsepower. The minimum power required for this configuration corresponding to the maximum endurance speed is 200.04 horsepower.

The maximum range speed decreased by 6.5 knots from its baseline speed of 124. The power required for maximum range decreased by 44.05 horsepower from its baseline value of 282.2 horsepower, a decrease of 15.6%. It can be said, therefore, that the maximum range of the Piper Navajo was increased by 15.6%.

Even though the maximum endurance speed only decreased by 2 knots from its baseline value of 89 knots, the minimum power required for maximum endurance decreased by 38.53 horsepower from its baseline value of 238.57 horsepower, a decrease of 16.15%.

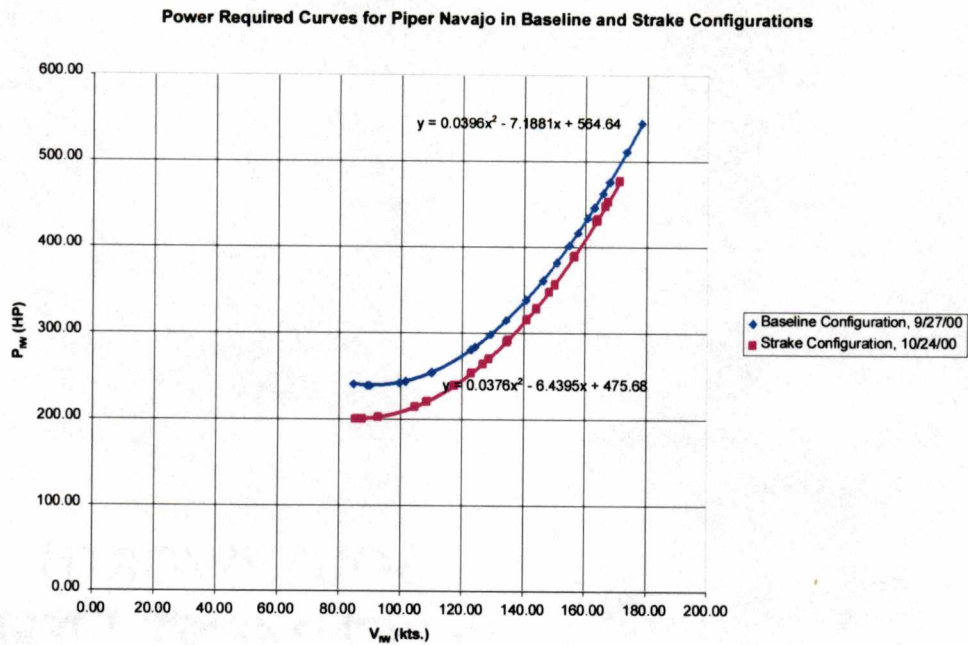


Figure 6.8. Power Required Curves for Piper Navajo in Baseline and Strake Configurations

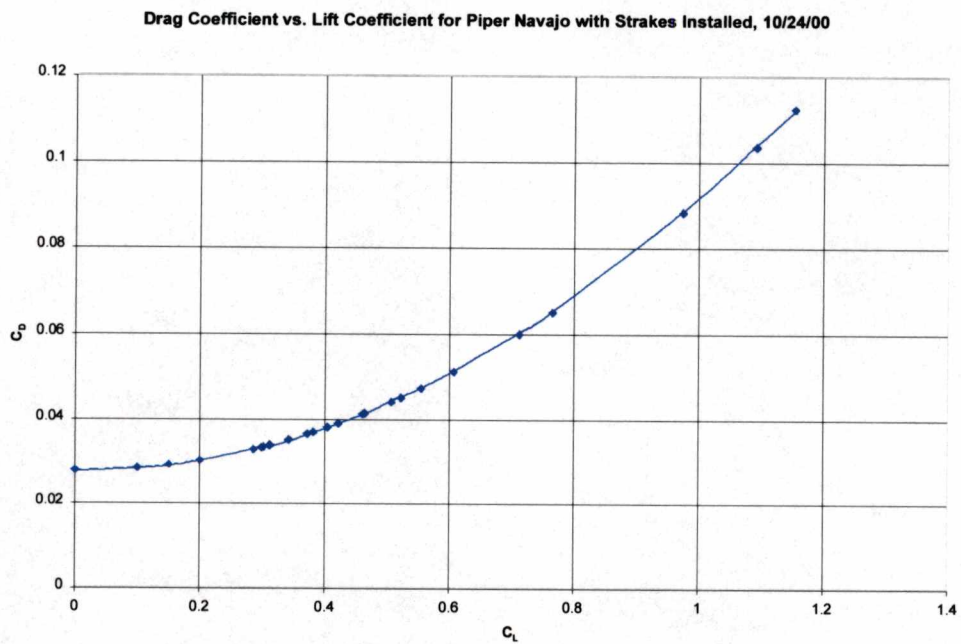


Figure 6.9. Drag Polar for the Piper Navajo with Strakes Installed

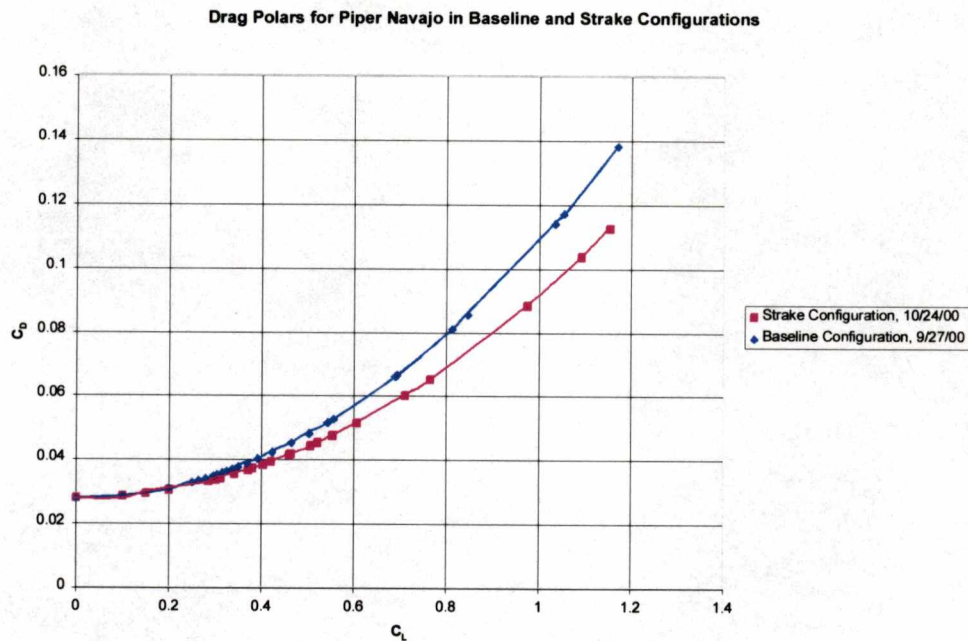


Figure 6.10. Drag Polars for the Piper Navajo in Baseline and Strake Configurations

reduced by the addition of the strakes. At a lift coefficient of about 0.27, the drag curves for the baseline and strake configurations start to diverge apart. At a lift coefficient of about 1.15, the baseline drag curve has a drag coefficient of about 0.138 while the drag curve for the Piper Navajo with strakes installed has only a drag coefficient of about 0.112. This is a difference in drag coefficient of 0.026 and the drag is reduced 18.84% at this lift coefficient.

The drag coefficient versus C_L^2 plot is shown in Figure 6.11. The drag coefficient with zero lift is calculated to be 0.0278, just as the baseline configuration. This chart shows that adding the strakes increases the efficiency of the Piper Navajo. The slope of this curve is 0.0636, giving an Oswald's efficiency factor value of 0.693. This is an improvement from its baseline efficiency factor value of 0.547, thus making

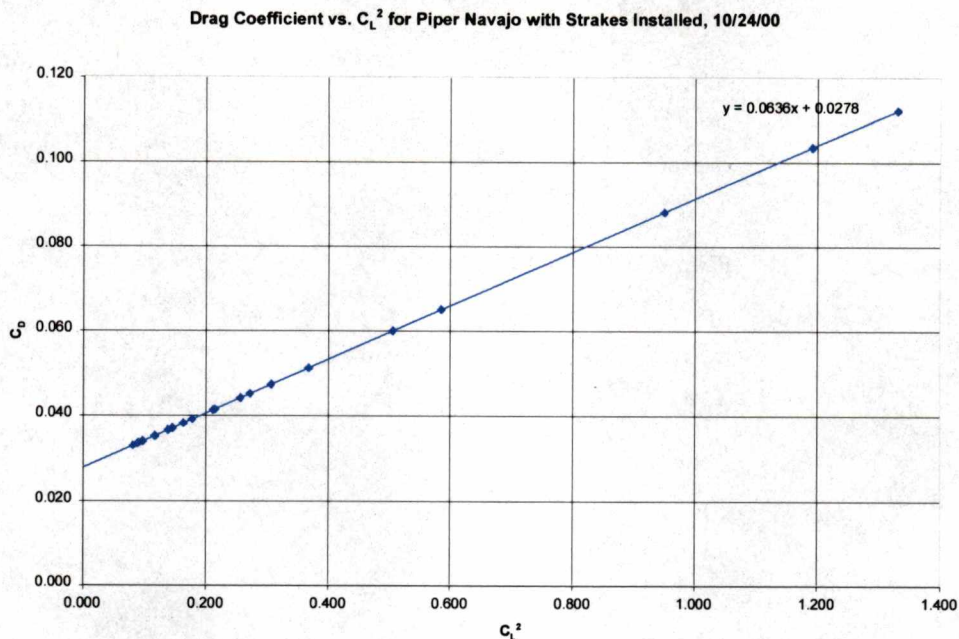


Figure 6.11. C_D versus C_L^2 for the Piper Navajo with Strakes Installed

the plane more efficient. The efficiency of the Piper Navajo was therefore increased by 26.69%.

3. Lift to Drag Ratio

A plot of the lift to drag ratio for the Piper Navajo with the strakes installed is shown in Figure 6.12. This plot shows that the Piper Navajo attains a maximum lift to drag ratio of 11.6 at a lift coefficient of 0.63. Taking the equation for lift coefficient and rearranging it to solve for airspeed, using the standard takeoff gross weight of 6500 pounds gives an airspeed of 115.28 knots. This value agrees very well with the maximum range airspeed of 117.5 knots determined in the PIW-VIW graph.

Figure 6.13 compares the lift to drag ratios of the Piper Navajo in the baseline and strake configurations. This plot also shows that drag was reduced. The addition of

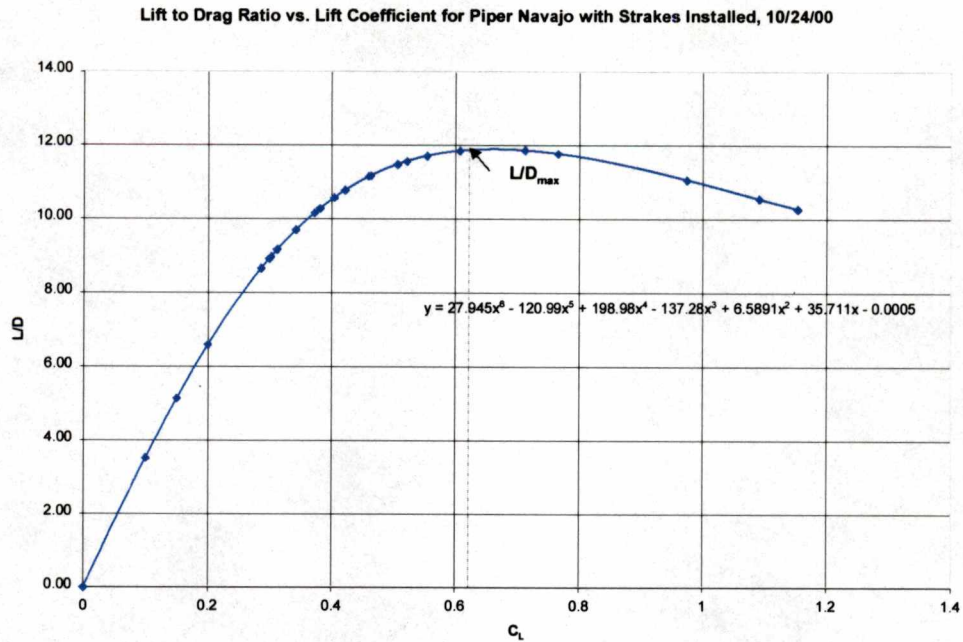


Figure 6.12. Lift to Drag Ratio of Piper Navajo with Strakes Installed

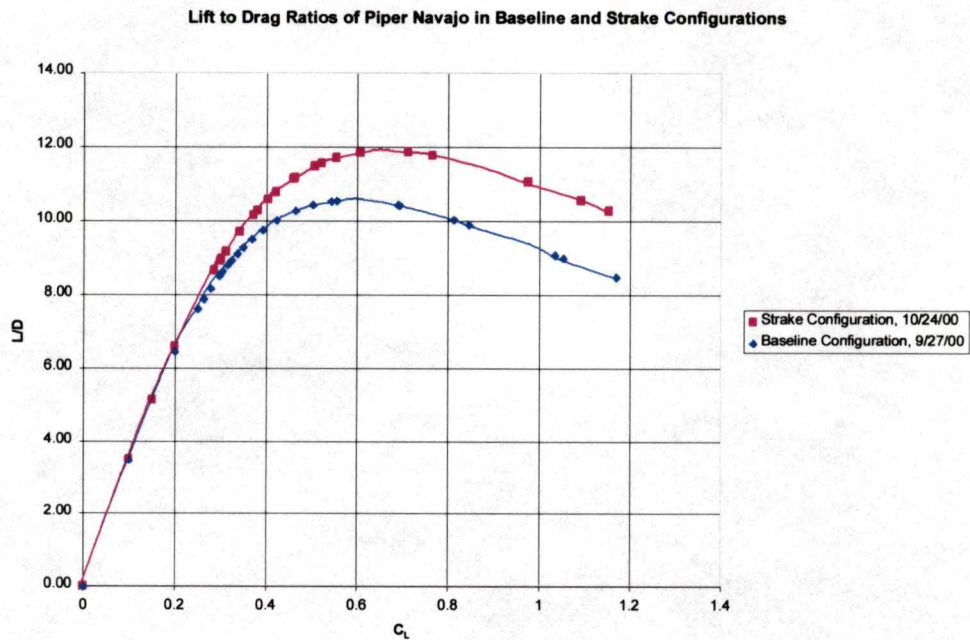


Figure 6.13. Lift to Drag Ratios of Piper Navajo in Baseline and Strake Configurations

the strakes increased the maximum lift to drag ratio from 10.53 to 11.6, thereby increasing the lift to drag ratio by 10.16%. The lift coefficient at which the maximum lift to drag ratio is attained was also increased from 0.556 to 0.63. Figure 6.13 shows that the strakes significantly increase the lift to drag ratio of the airplane. The airplane with strakes was able to reach higher lift to drag ratios within the same range of lift coefficients than the airplane in the baseline configuration.

Flow Visualization Analysis

Flow visualizations of the left and right wings of the Piper Navajo, between the left and right nacelles and the fuselage, in the baseline configuration are compared with those in the strake configuration. These pictures show the flow at observed airspeeds of 130, 120, 110, 100, 90 and 80 knots.

At 130 knots, the flow is still very well attached, as shown in Figures 6.14a and b. The flow is also attached at the trailing edge of the right wing for the baseline and strakes configurations, as shown in Figures 6.14 c and d, but it is not as smooth. The flow is very well attached on the left wing. This is shown in Figures 6.14e-h. The flow on the left trailing edge is much smoother than on the right trailing edge, indicating that the flow is better attached.

Figures 6.15a-h show that the flow becomes a little less attached at 120 knots. No significant difference can be seen between the flow visualization at the right wing trailing edge between the baseline and strake configurations. The flow on the left wing is still very well attached at 120 knots.

The flow at 110 knots is shown in Figures 6.16a-h. At 110 knots, the right wing

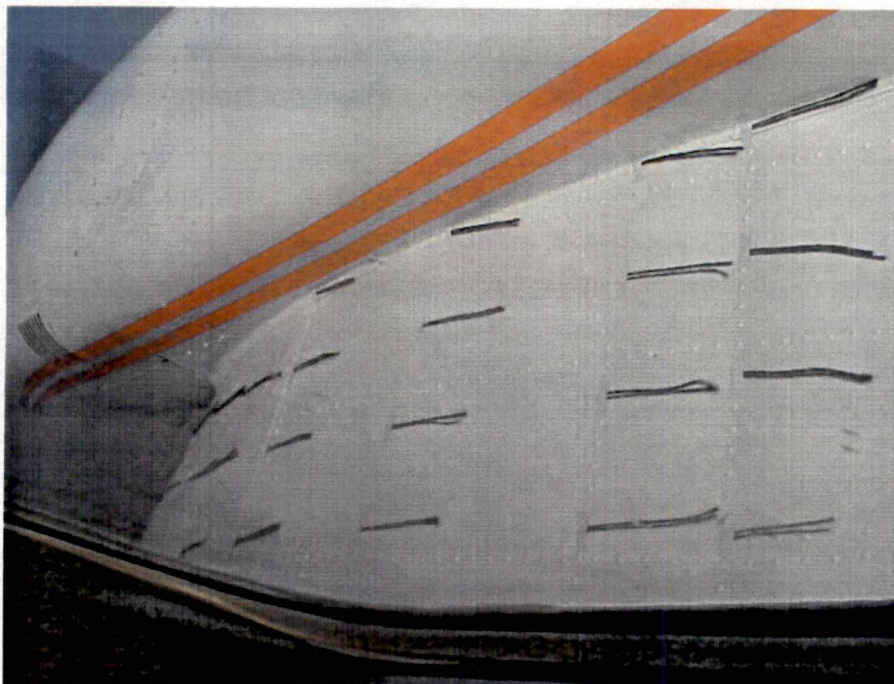


Figure 6.14a. Flow Visualization between Fuselage and Right Nacelle towards Leading Edge at 130 Knots - Baseline Configuration



Figure 6.14b. Flow Visualization between Fuselage and Right Nacelle towards Leading Edge at 130 Knots - Strake Configuration

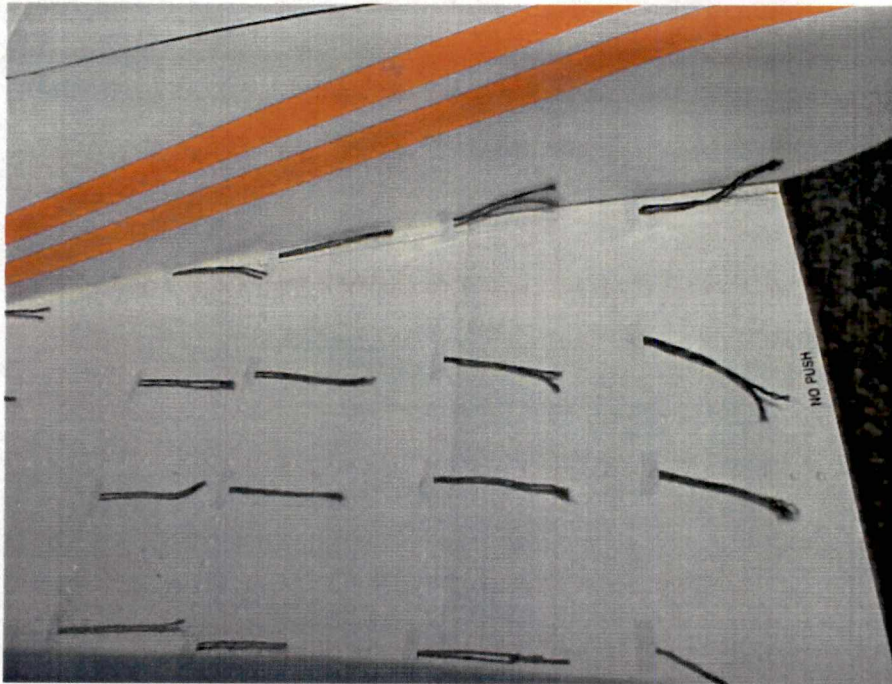


Figure 6.14c. Flow Visualization at Trailing Edge of Right Wing at 130 Knots - Baseline Configuration

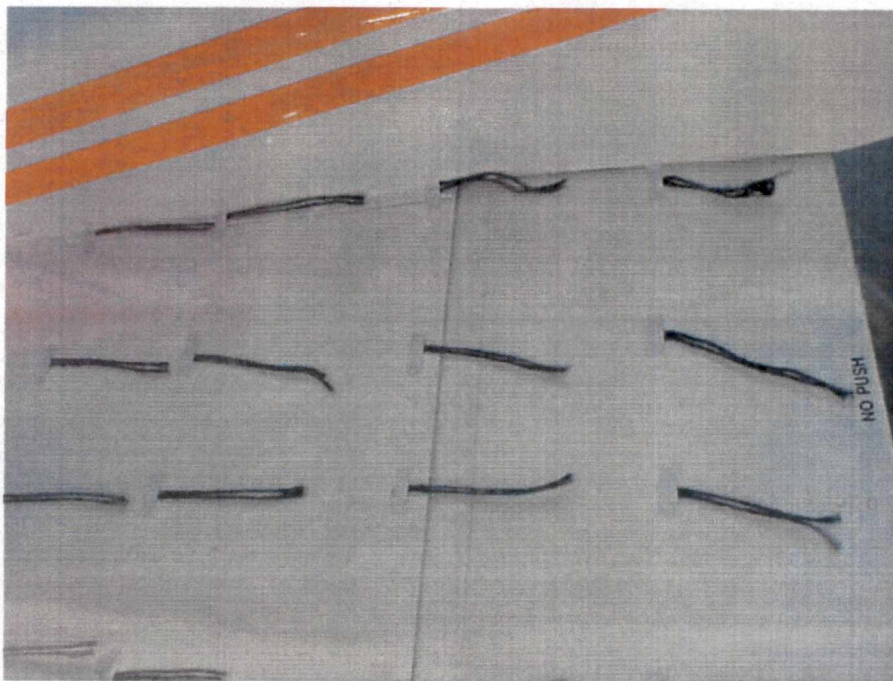


Figure 6.14d. Flow Visualization at Trailing Edge of Right Wing at 130 Knots - Strake Configuration

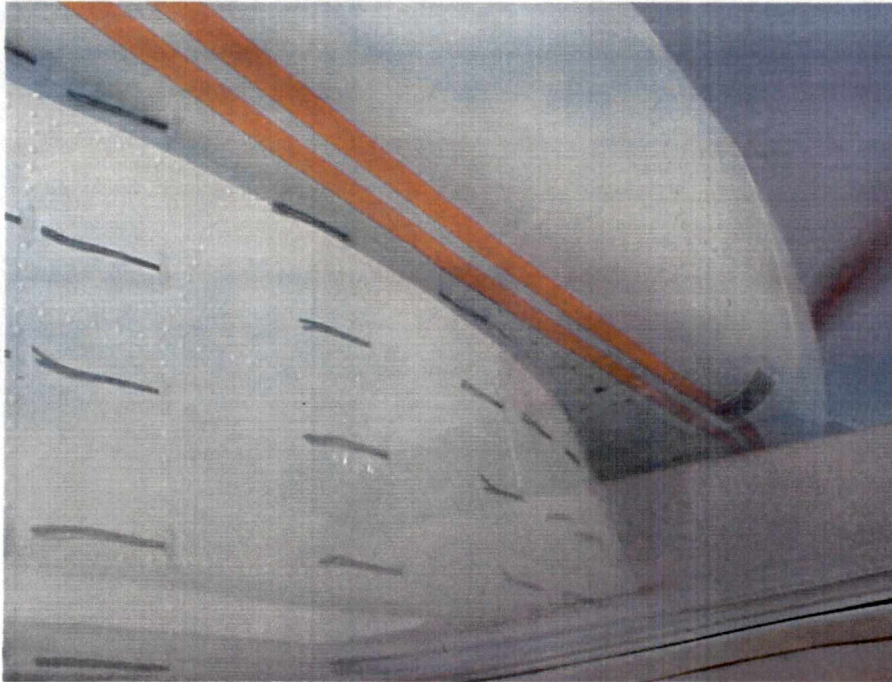


Figure 6.14e. Flow Visualization between Fuselage and Left Nacelle towards Leading Edge at 130 Knots - Baseline Configuration

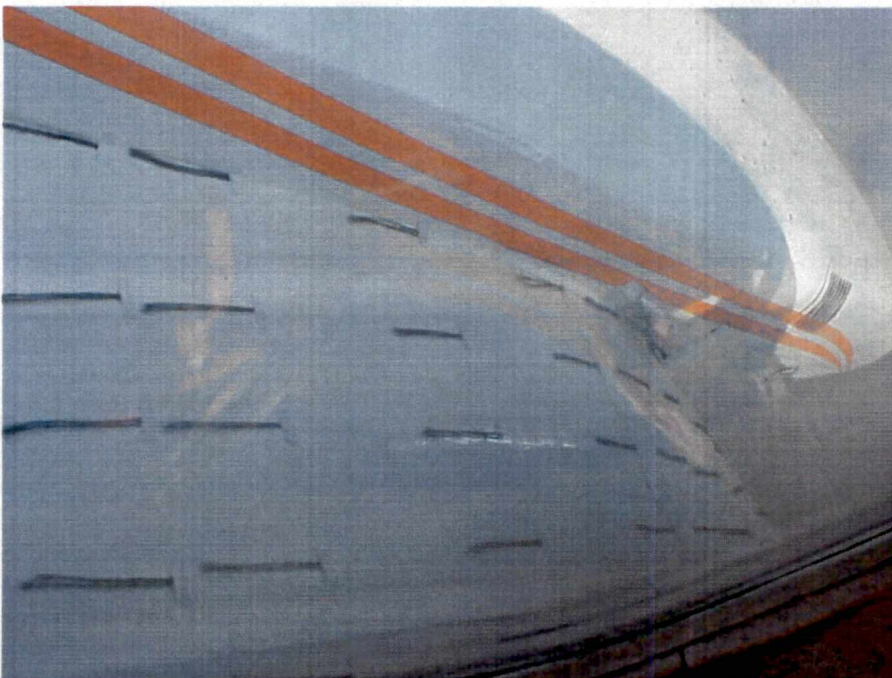


Figure 6.14f. Flow Visualization between Fuselage and Left Nacelle towards Leading Edge at 130 Knots - Strake Configuration

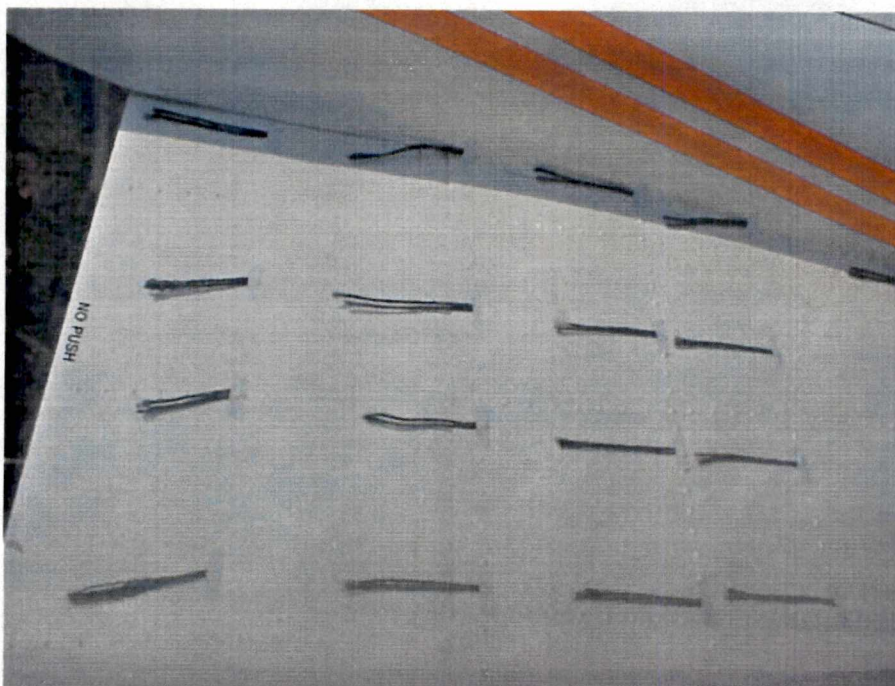


Figure 6.14g. Flow Visualization at Trailing Edge of Left Wing at 130 Knots - Baseline Configuration

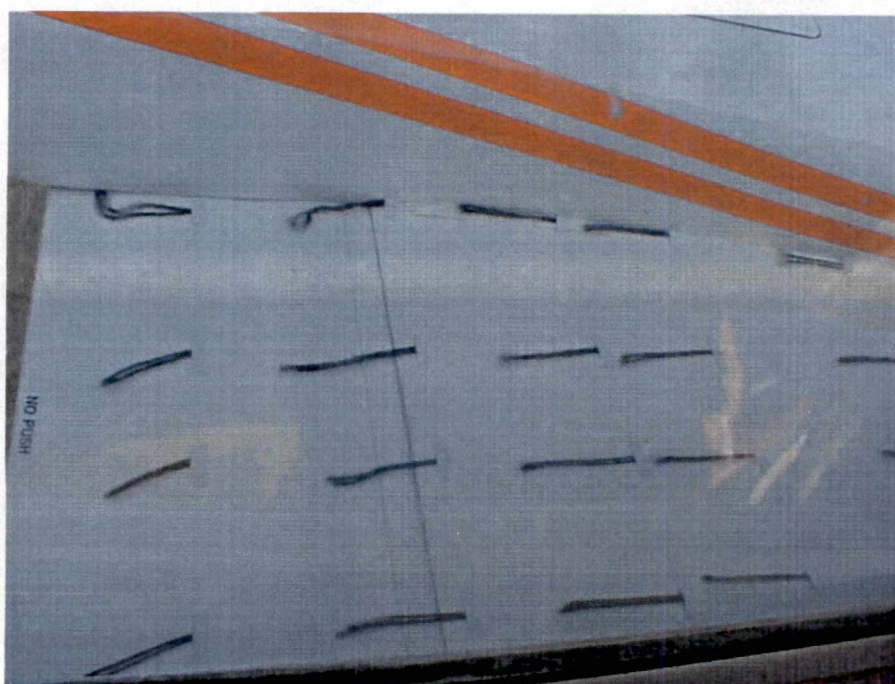


Figure 6.14h. Flow Visualization at Trailing Edge of Left Wing at 130 Knots - Strake Configuration

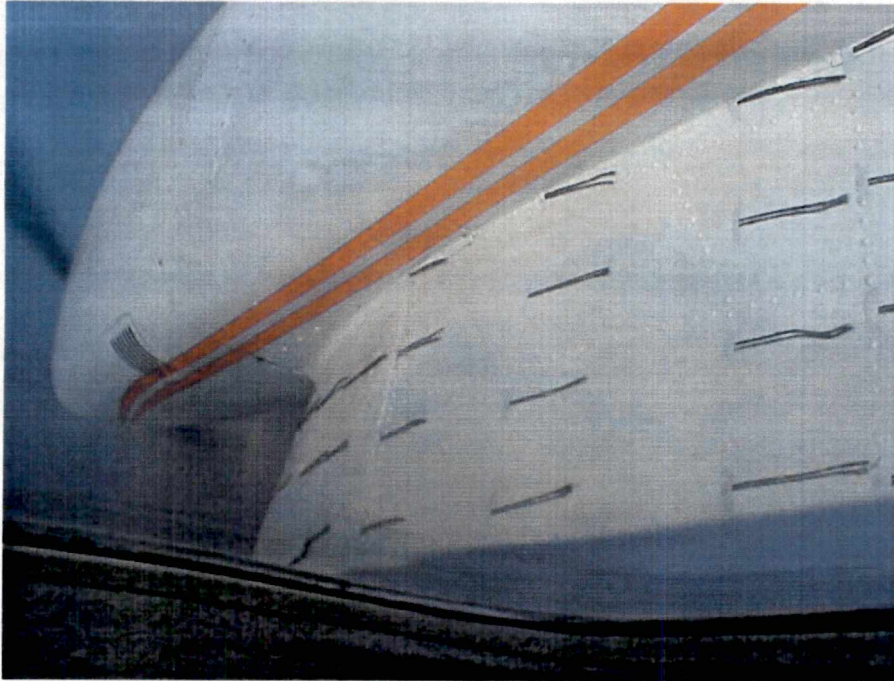


Figure 6.15a. Flow Visualization between Fuselage and Right Nacelle towards Leading Edge at 120 Knots - Baseline Configuration

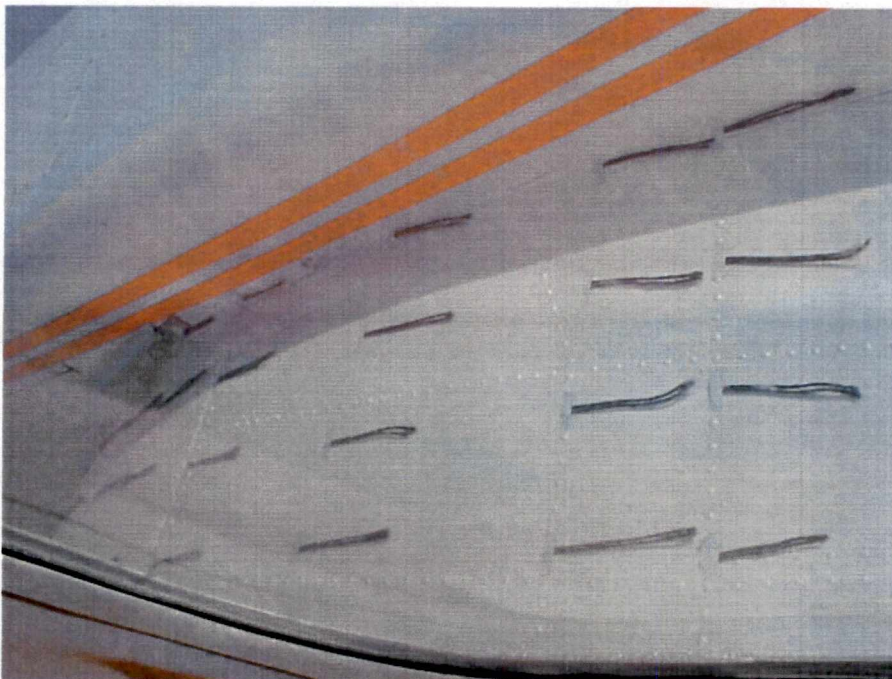


Figure 6.15b. Flow Visualization between Fuselage and Right Nacelle towards Leading Edge at 120 Knots - Strake Configuration

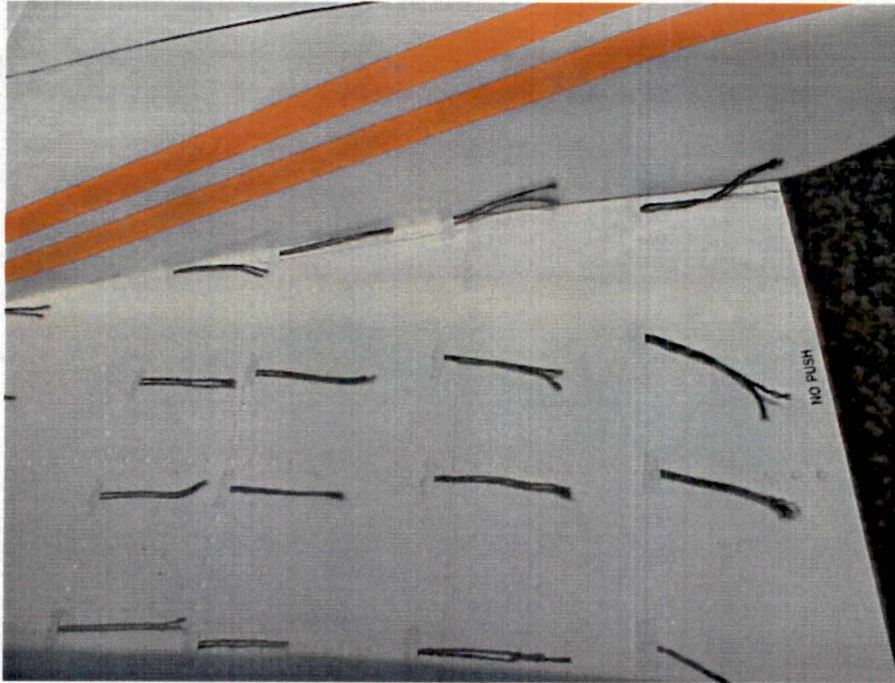


Figure 6.15c. Flow Visualization at Trailing Edge of Right Wing at 120 Knots - Baseline Configuration



Figure 6.15d. Flow Visualization at Trailing Edge of Right Wing at 120 Knots - Strake Configuration

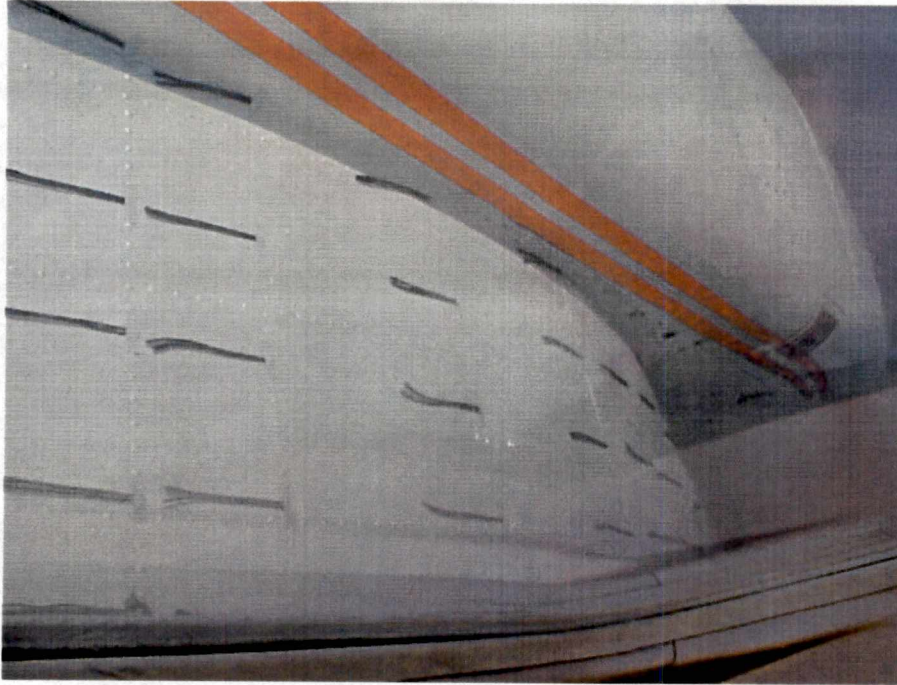


Figure 6.15e. Flow Visualization between Fuselage and Left Nacelle towards Leading Edge at 120 Knots - Baseline Configuration

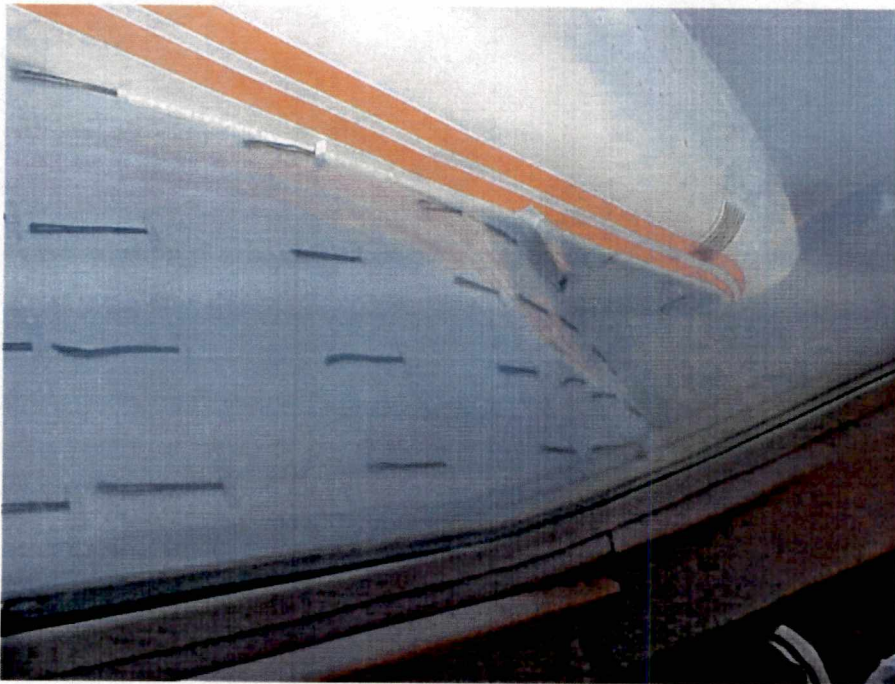


Figure 6.15f. Flow Visualization between Fuselage and Left Nacelle towards Leading Edge at 120 Knots - Strake Configuration

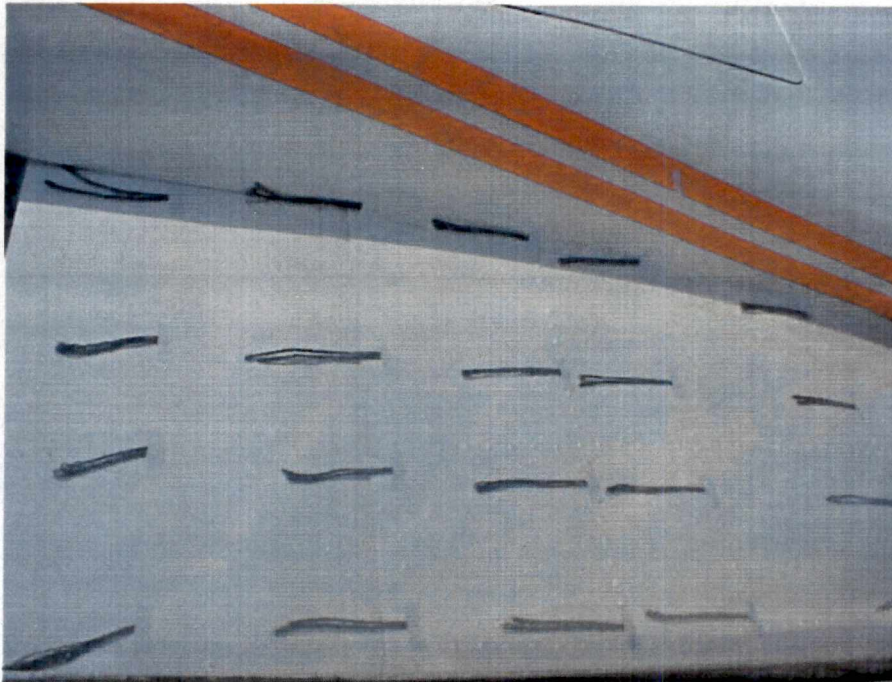


Figure 6.15g. Flow Visualization at Trailing Edge of Left Wing at 120 Knots - Baseline Configuration

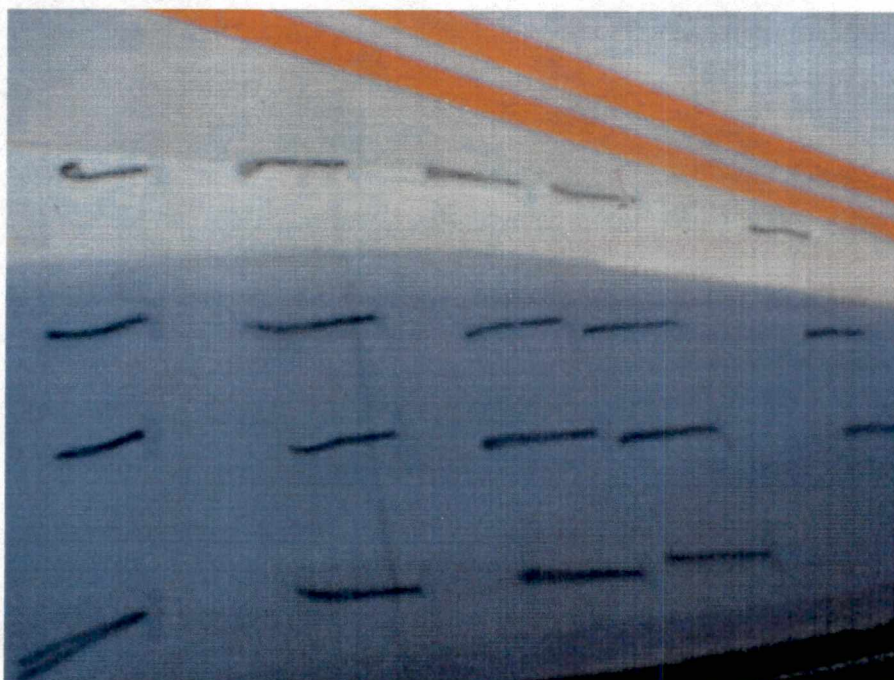


Figure 6.15h. Flow Visualization at Trailing Edge of Left Wing at 120 Knots - Strake Configuration

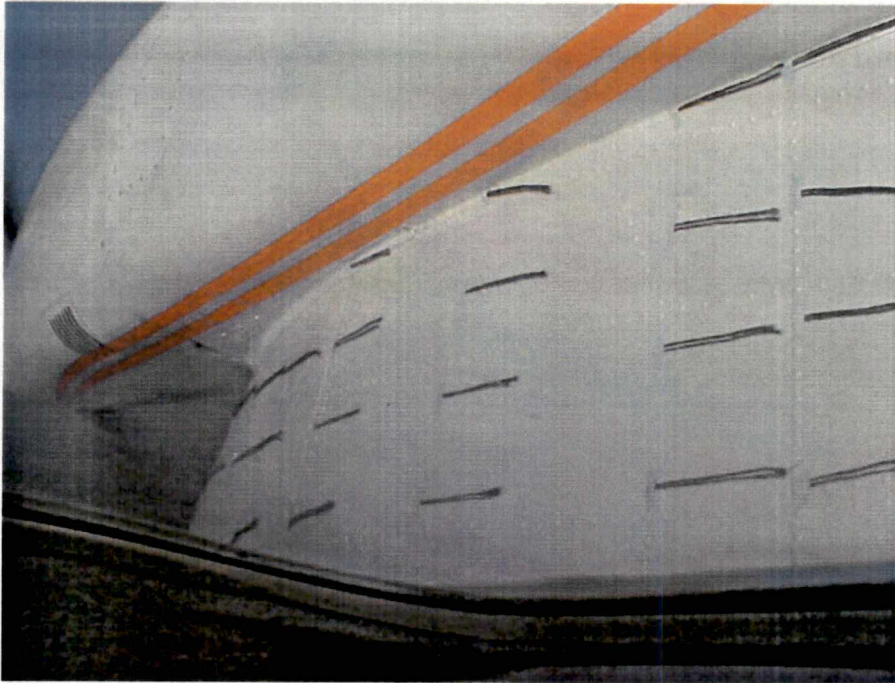


Figure 6.16a. Flow Visualization between Fuselage and Right Nacelle towards Leading Edge at 110 Knots - Baseline Configuration



Figure 6.16b. Flow Visualization between Fuselage and Right Nacelle towards Leading Edge at 110 Knots - Strake Configuration

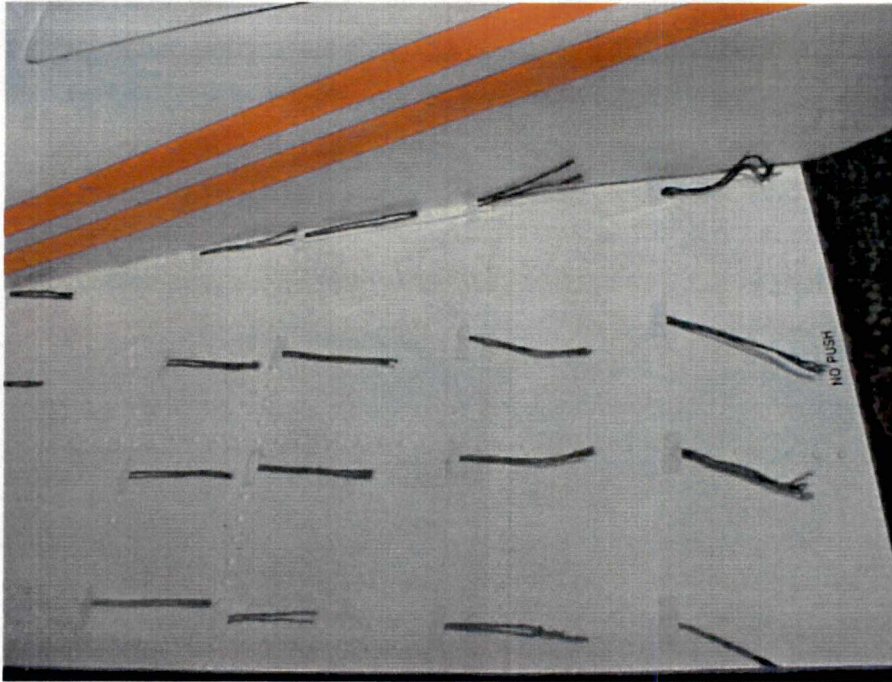


Figure 6.16c. Flow Visualization at Trailing Edge of Right Wing at 110 Knots - Baseline Configuration



Figure 6.16d. Flow Visualization at Trailing Edge of Right Wing at 110 Knots - Strake Configuration



Figure 6.16e. Flow Visualization between Fuselage and Left Nacelle towards Leading Edge at 110 Knots - Baseline Configuration



Figure 6.16f. Flow Visualization between Fuselage and Left Nacelle towards Leading Edge at 110 Knots - Strake Configuration

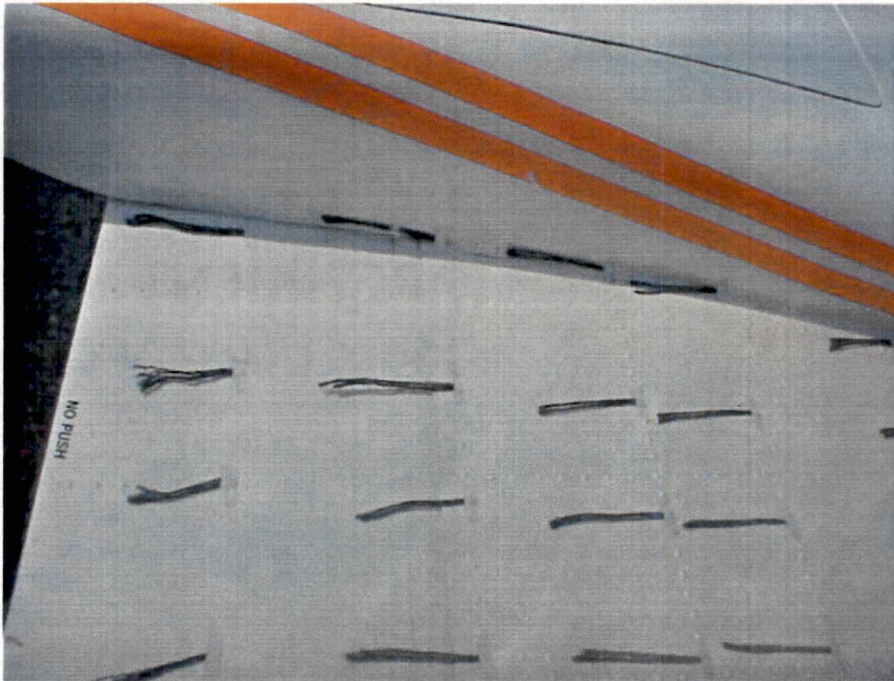


Figure 6.16g. Flow Visualization at Trailing Edge of Left Wing at 110 Knots - Baseline Configuration

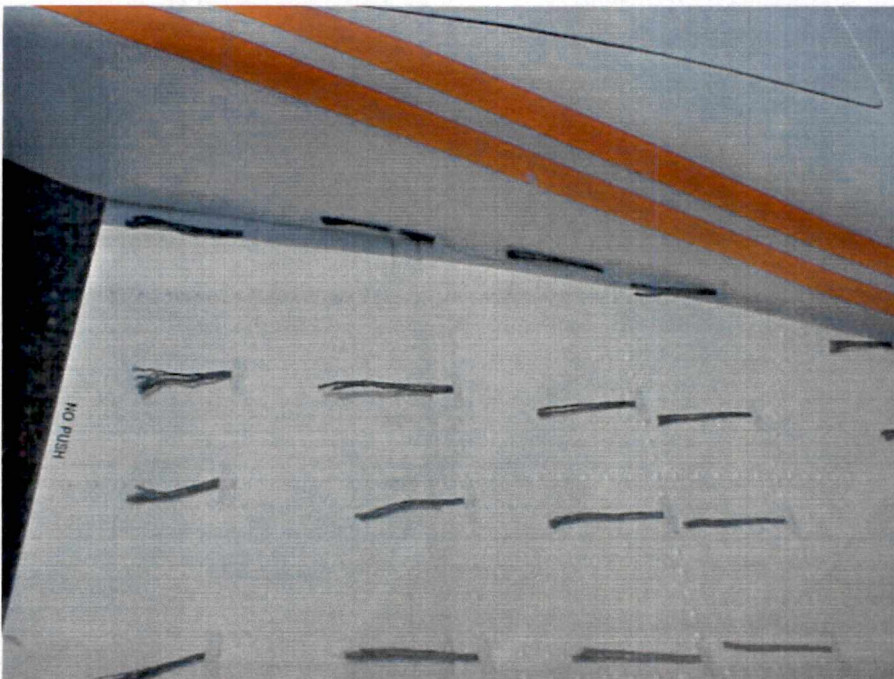


Figure 6.16h. Flow Visualization at Trailing Edge of Left Wing at 110 Knots - Strake Configuration

trailing edge flow became a little less smooth, as shown in Figure 6.16c. Figure 6.16d shows that the second row forward of the wing trailing edge has a smoother flow in the strake configuration due to the fact that the tufts remain straighter.

Figures 6.17a-h show the flow at 100 knots. The separation started to move forward on the right wing at 100 knots, as shown in Figures 6.17c and d. No significant difference can be seen between the flow visualization at the right wing trailing edge between the baseline and strake configurations.

Figures 6.18a-h show the flow at 90 knots. The flow towards the leading edge of the right wing became a little less smooth at this speed. Looking at Figures 6.18a and b, the flow is still attached but the tufts in the fourth and fifth rows are not flowing as straight. There is still no significant difference seen between the baseline and strake configurations in these two pictures. Figures 6.18c and c show that the flow separation has progressed forward from the right wing trailing edge. The flow on the left wing trailing edge started to become less smooth at 90 knots, as shown in Figures 6.18g and h. The tufts fastened closer to the leading edge start to flow less straight at 80 knots as shown in Figures 6.19a and b. This shows that the flow is commencing to separate in this region. Figures 6.19c and d definitely show that flow separation has occurred at the right wing trailing edge and has progressed forward. The tufts in the fourth row from the right wing trailing edge flow straighter in the strake configuration. Flow separation of the left wing trailing edge started to commence as seen in Figures 6.19g and h.

Observing the flow visualization reveals that the right wing tends to separate before the left wing. This indicates that there may be less interference drag on the left

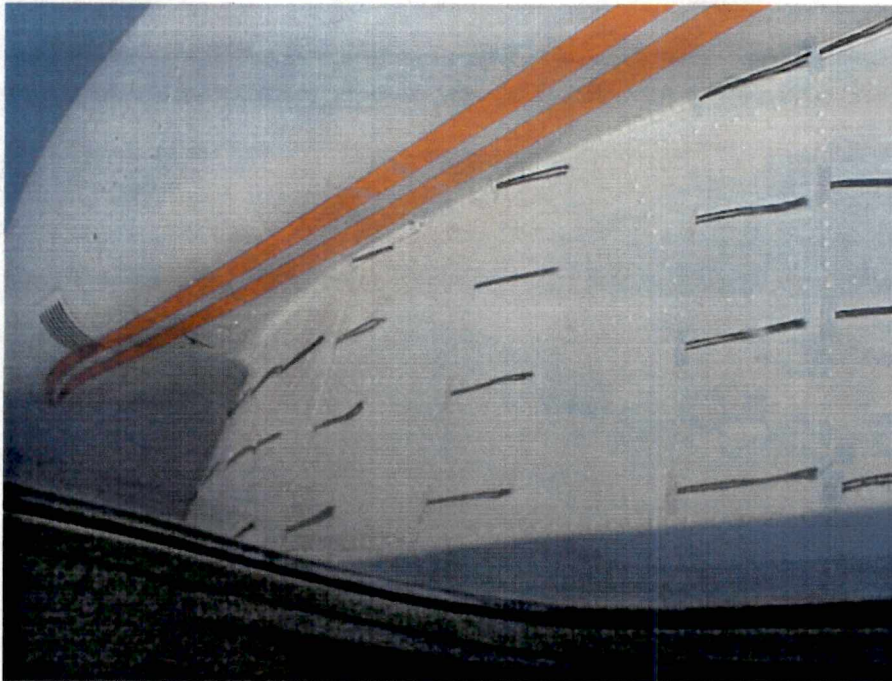


Figure 6.17a. Flow Visualization between Fuselage and Right Nacelle towards Leading Edge at 100 Knots - Baseline Configuration

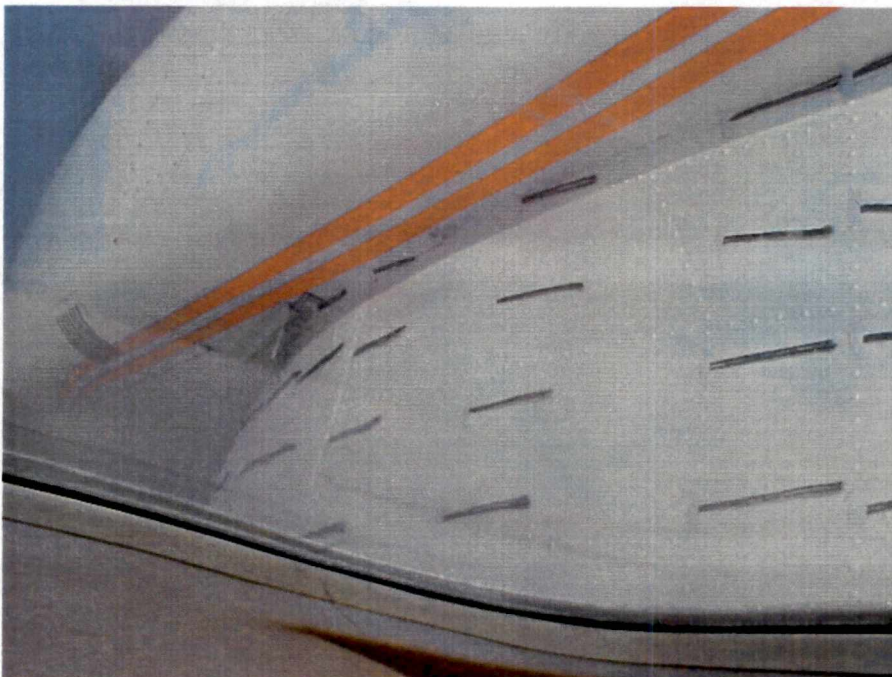


Figure 6.17b. Flow Visualization between Fuselage and Right Nacelle towards Leading Edge at 100 Knots - Strake Configuration

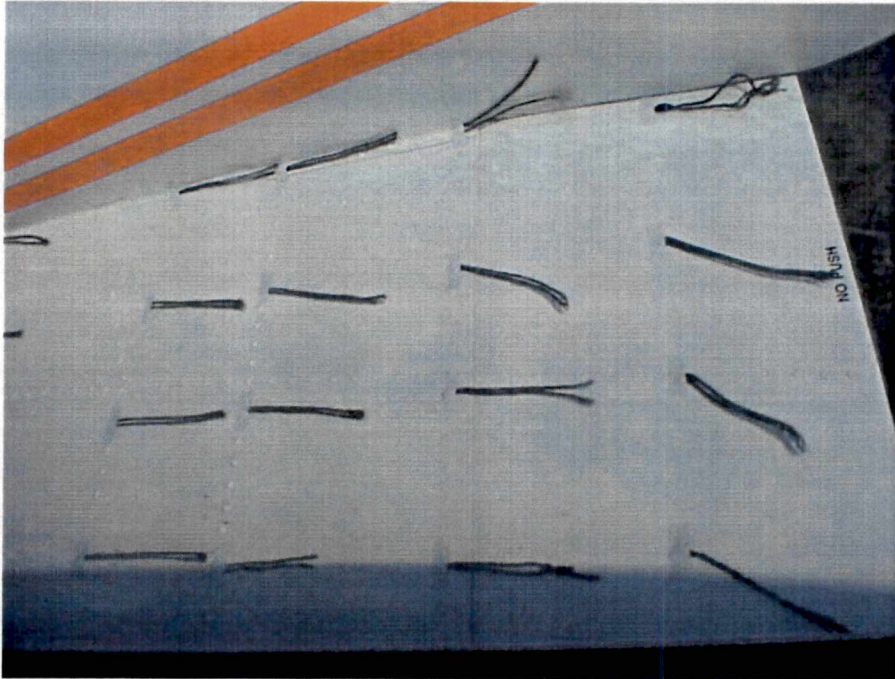


Figure 6.17c. Flow Visualization at Trailing Edge of Right Wing at 100 Knots - Baseline Configuration

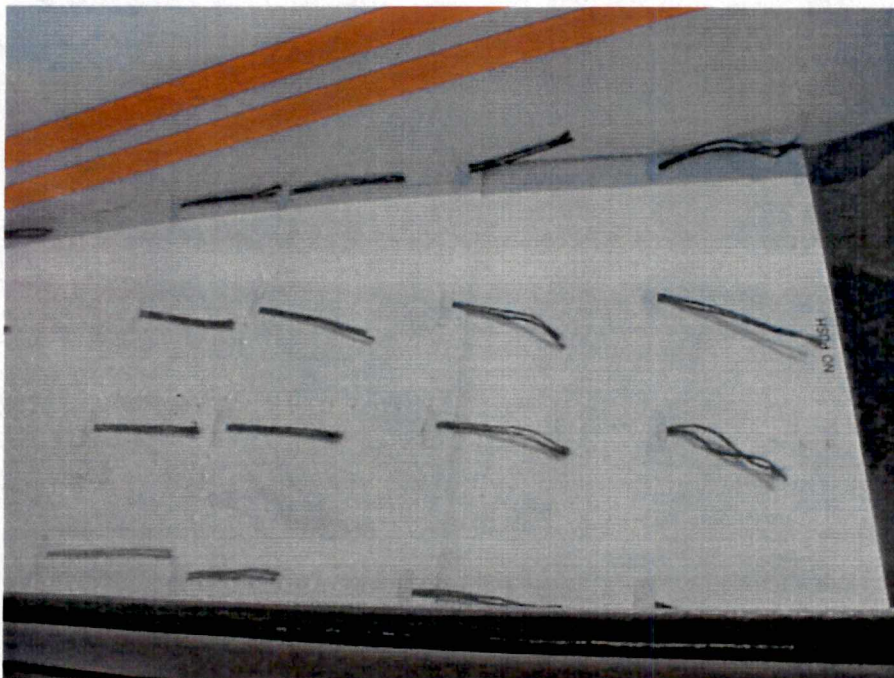


Figure 6.17d. Flow Visualization at Trailing Edge of Right Wing at 100 Knots - Strake Configuration

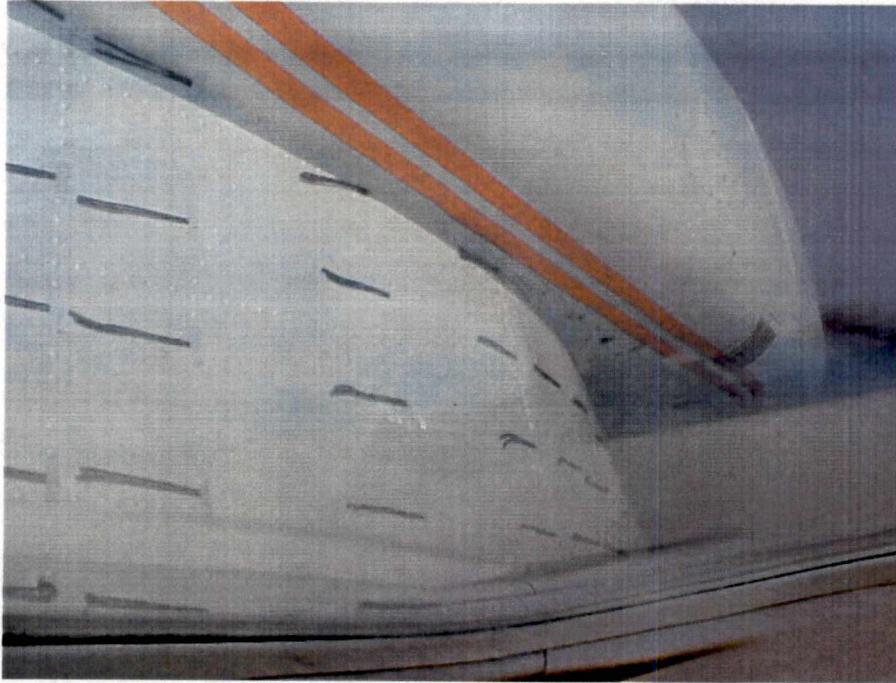


Figure 6.17e. Flow Visualization between Fuselage and Left Nacelle towards Leading Edge at 100 Knots - Baseline Configuration

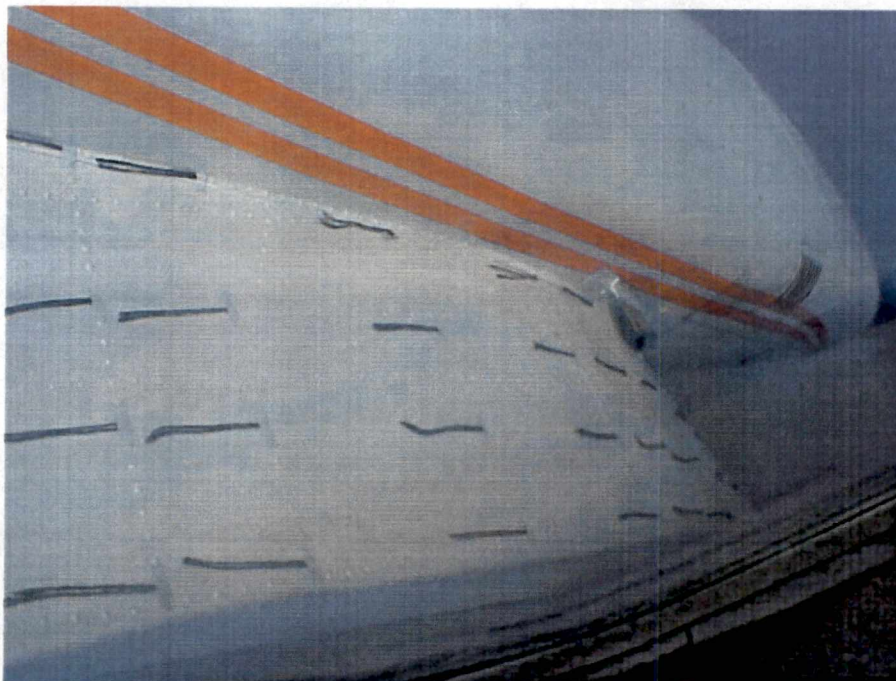


Figure 6.17f. Flow Visualization between Fuselage and Left Nacelle towards Leading Edge at 100 Knots - Strake Configuration

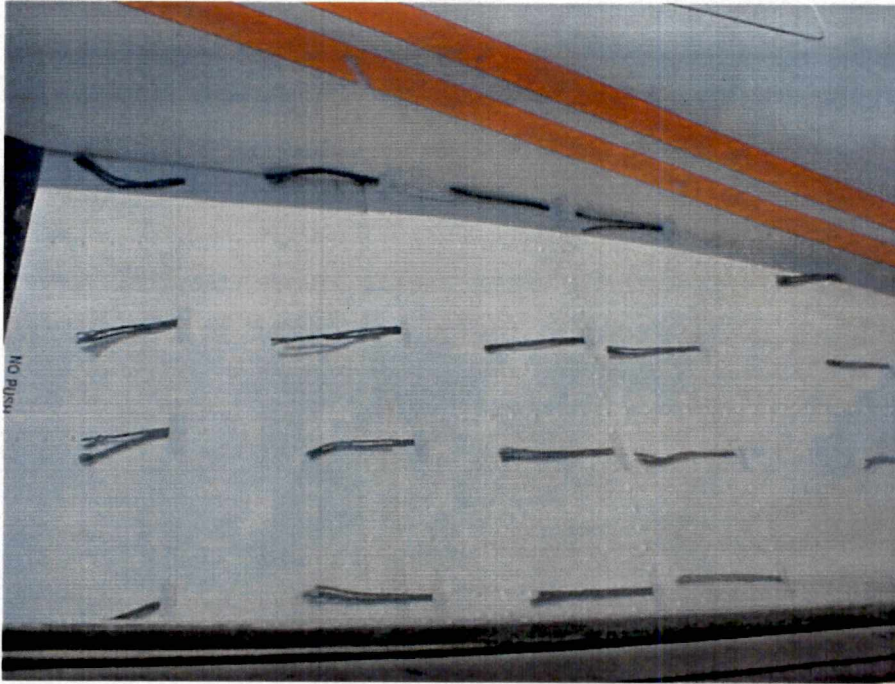


Figure 6.17g. Flow Visualization at Trailing Edge of Left Wing at 100 Knots - Baseline Configuration

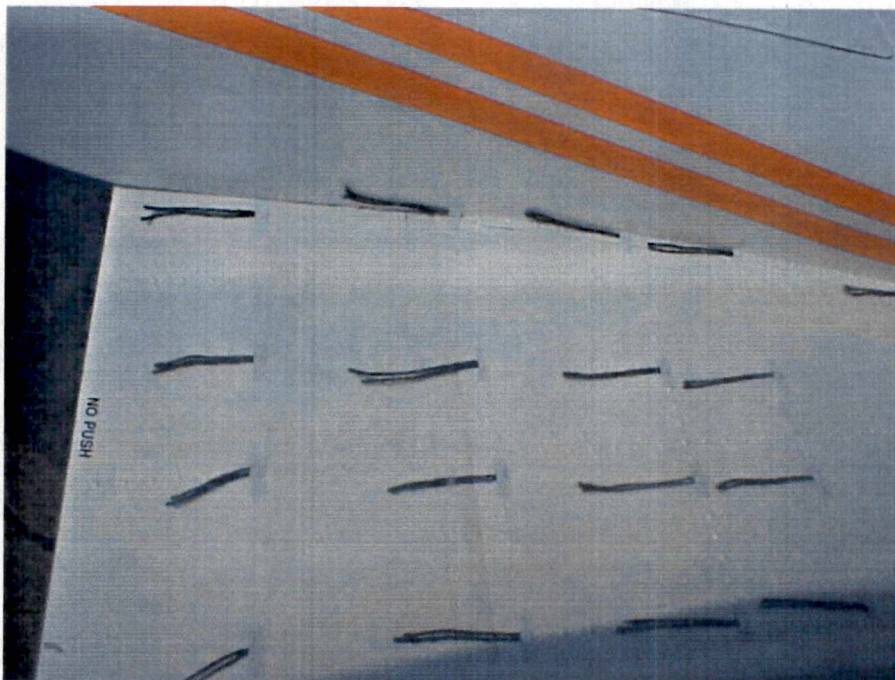


Figure 6.17h. Flow Visualization at Trailing Edge of Left Wing at 100 Knots - Strake Configuration

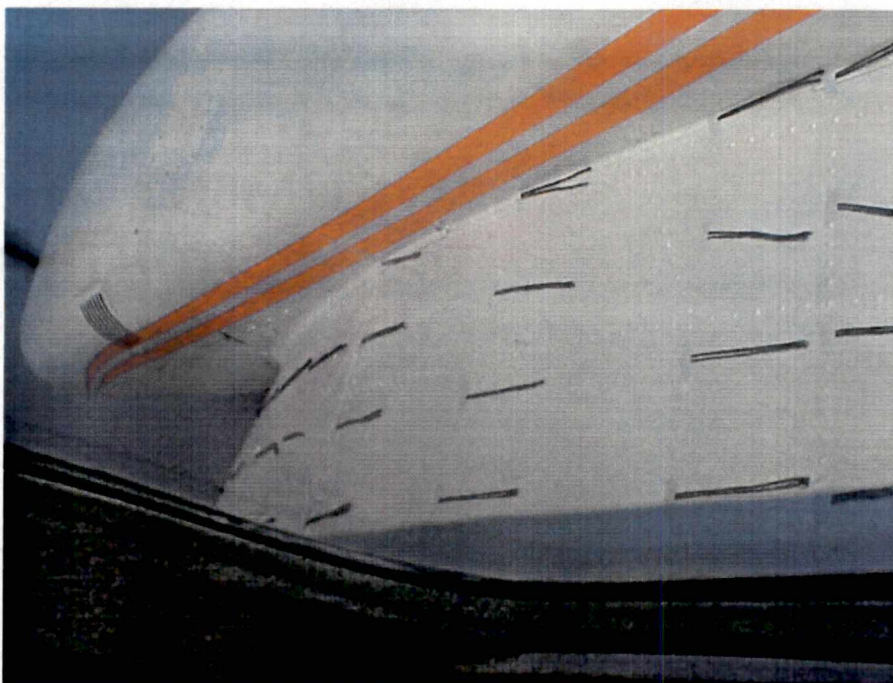


Figure 6.18a. Flow Visualization between Fuselage and Right Nacelle towards Leading Edge at 90 Knots - Baseline Configuration

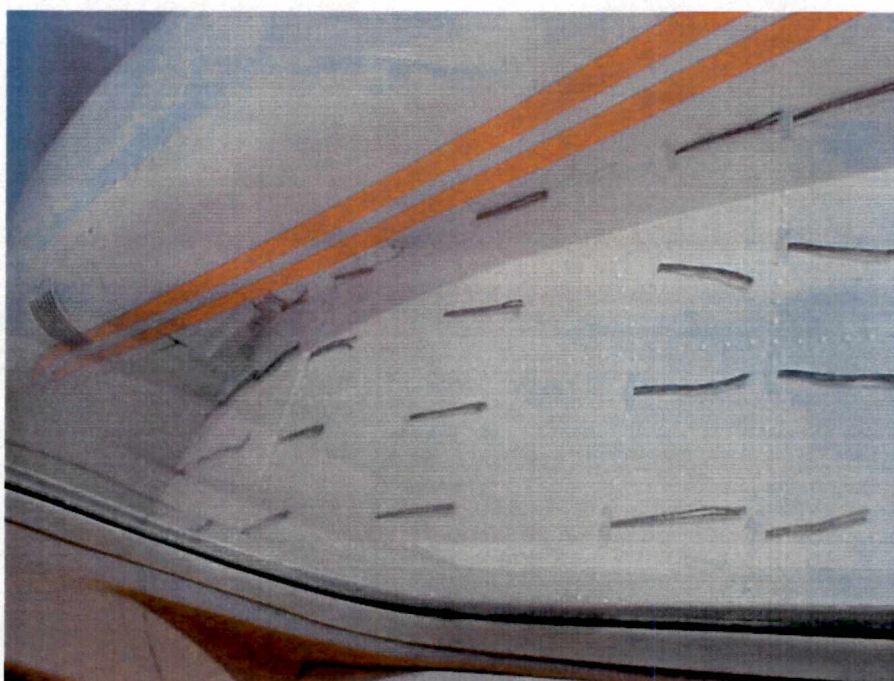


Figure 6.18b. Flow Visualization between Fuselage and Right Nacelle towards Leading Edge at 90 Knots - Strake Configuration

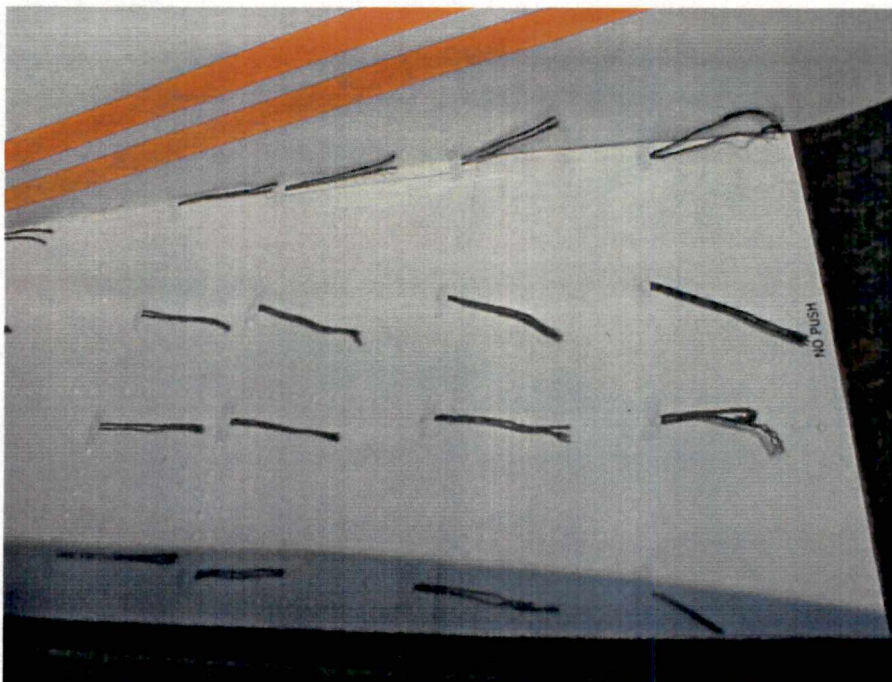


Figure 6.18c. Flow Visualization at Trailing Edge of Right Wing at 90 Knots - Baseline Configuration



Figure 6.18d. Flow Visualization at Trailing Edge of Right Wing at 90 Knots - Strake Configuration

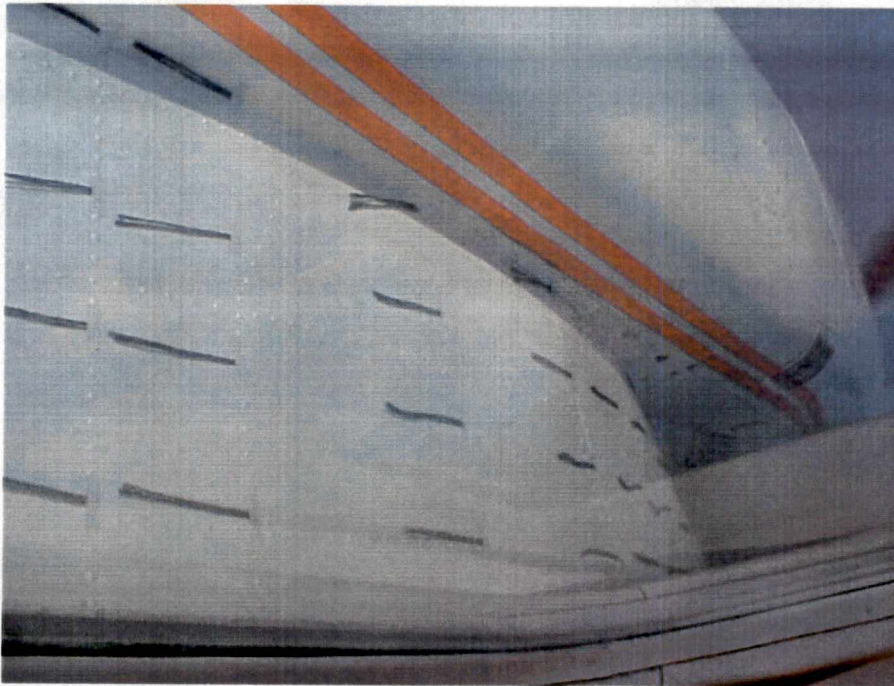


Figure 6.18e. Flow Visualization between Fuselage and Left Nacelle towards Leading Edge at 90 Knots - Baseline Configuration

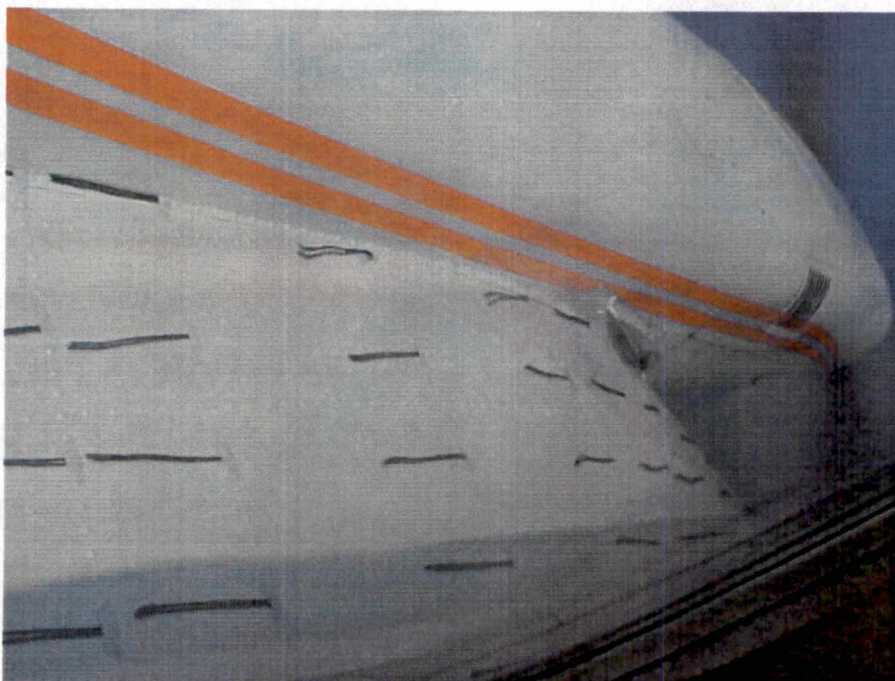


Figure 6.18f. Flow Visualization between Fuselage and Left Nacelle towards Leading Edge at 90 Knots - Strake Configuration

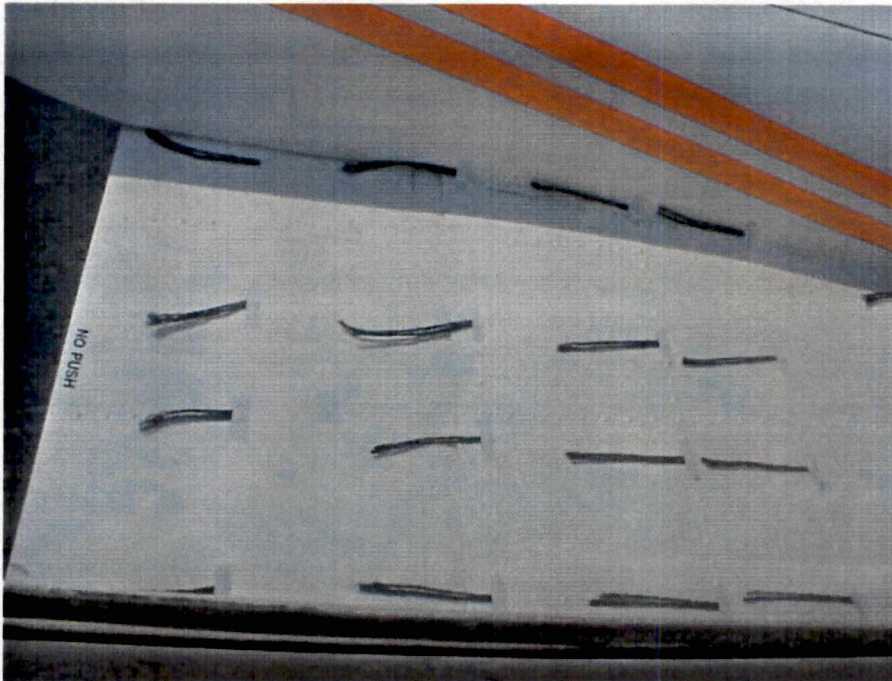


Figure 6.18g. Flow Visualization at Trailing Edge of Left Wing at 90 Knots - Baseline Configuration

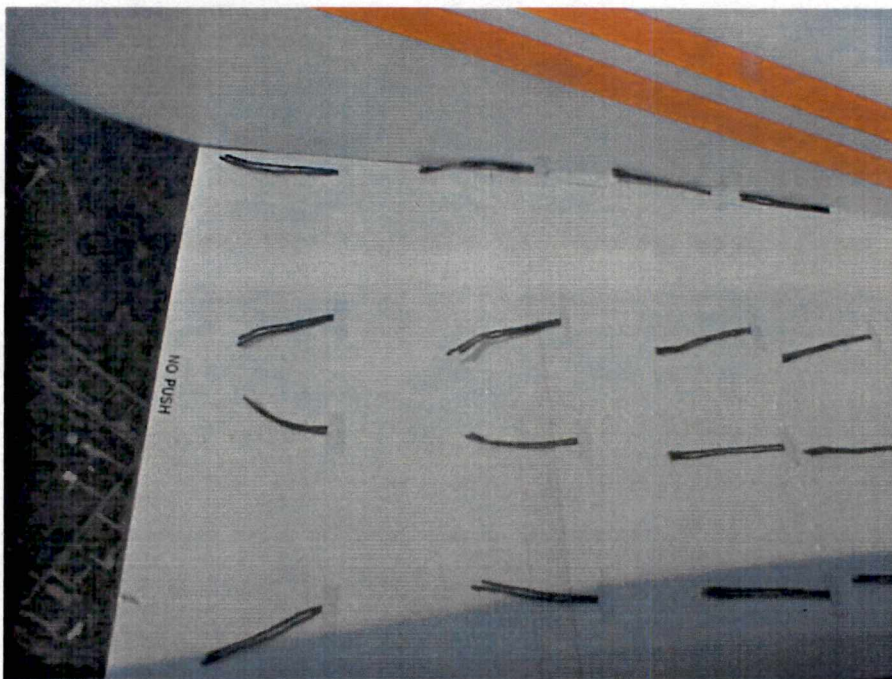


Figure 6.18h. Flow Visualization at Trailing Edge of Left Wing at 90 Knots - Strake Configuration

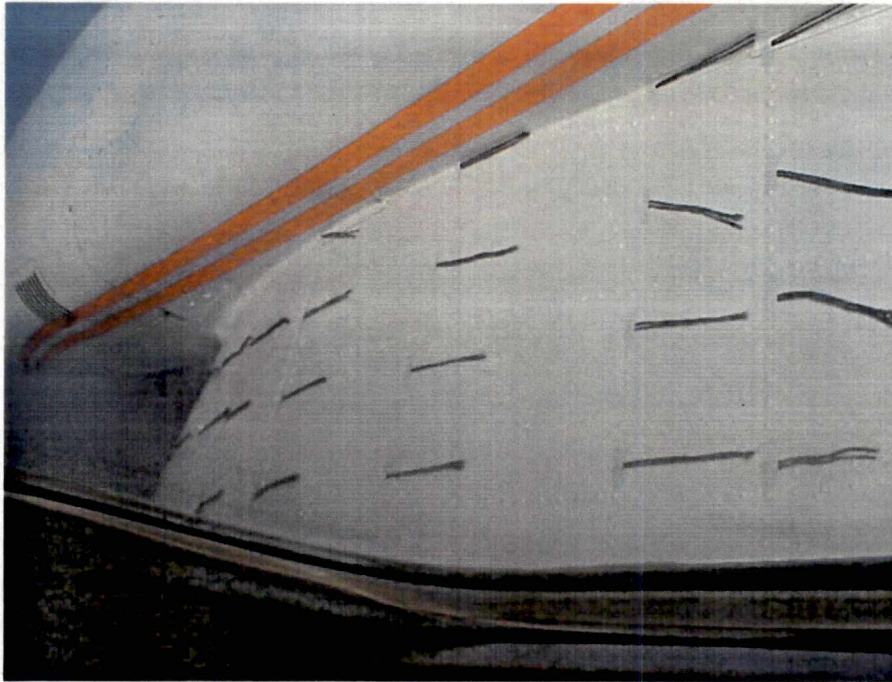


Figure 6.19a - Flow Visualization between Fuselage and Right Nacelle towards Leading Edge at 80 Knots - Baseline Configuration

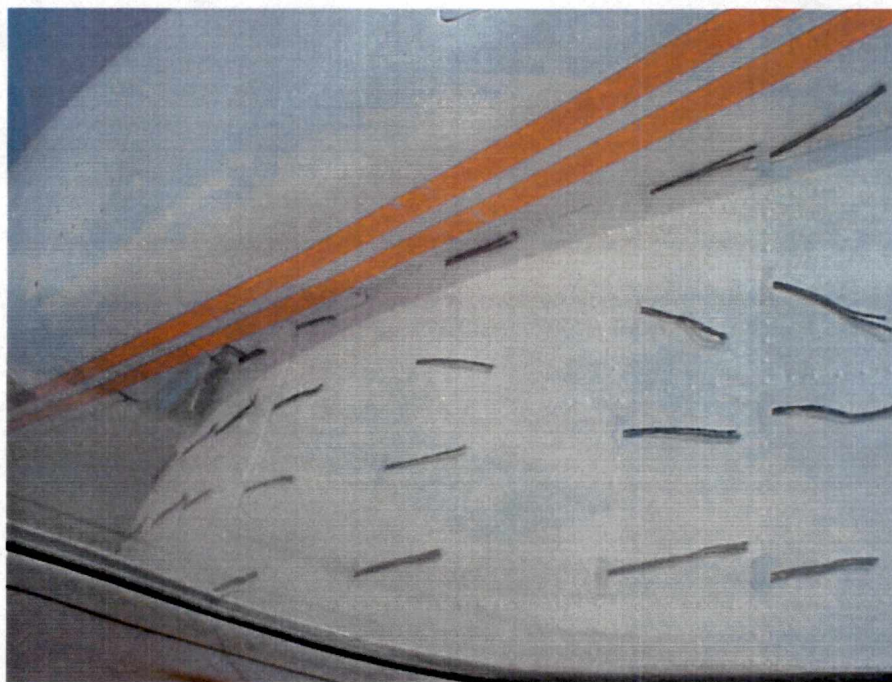


Figure 6.19b - Flow Visualization between Fuselage and Right Nacelle towards Leading Edge at 80 Knots - Strake Configuration

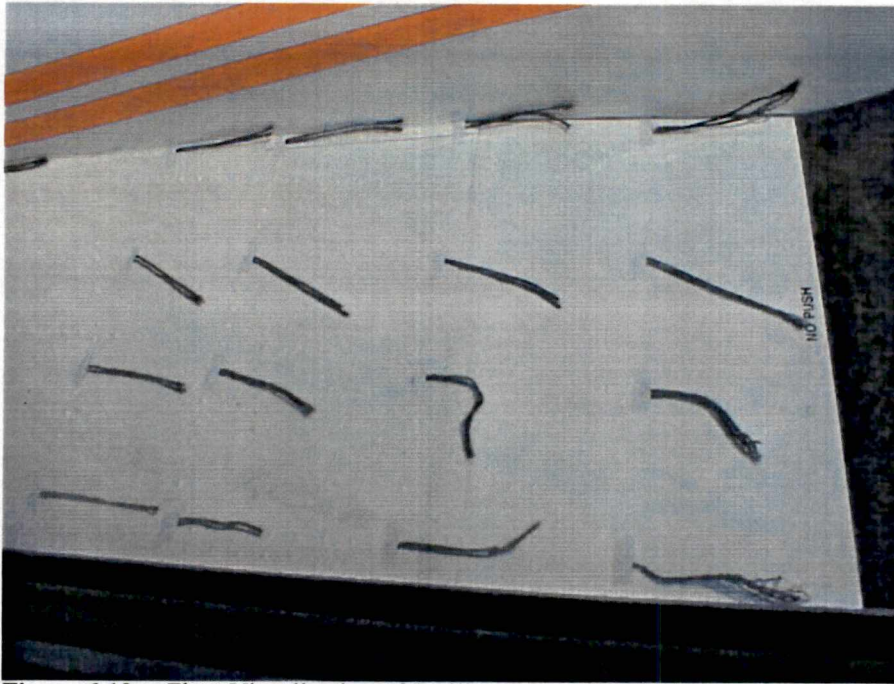


Figure 6.19c. Flow Visualization of Trailing Edge of Right Wing at 80 Knots - Baseline Configuration



Figure 6.19d. Flow Visualization of Trailing Edge of Right Wing at 80 Knots - Strake Configuration



Figure 6.19e. Flow Visualization between Fuselage and Left Nacelle towards Leading Edge at 80 Knots -Baseline Configuration

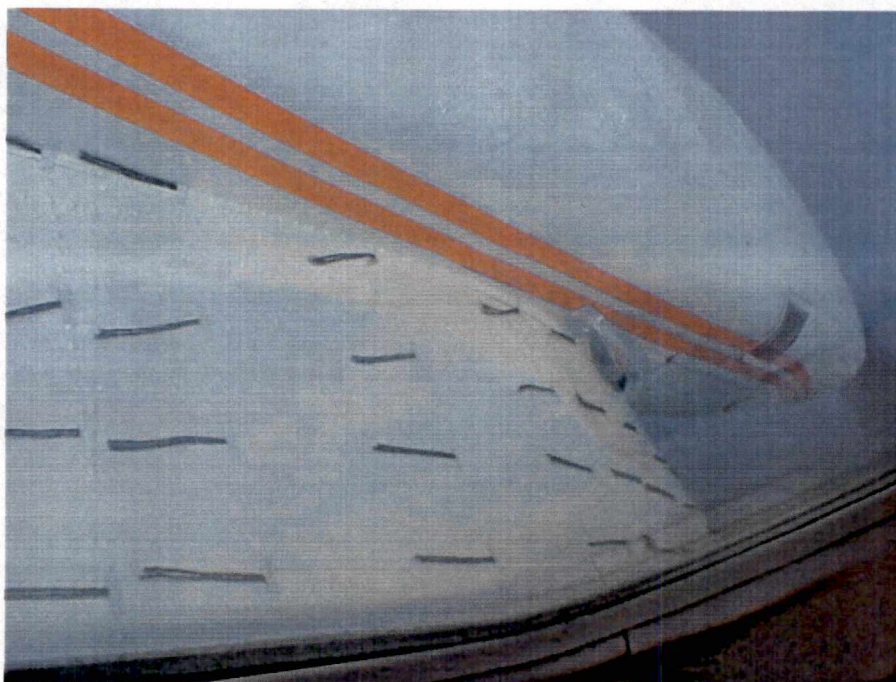


Figure 6.19f. Flow Visualization between Fuselage and Left Nacelle towards Leading Edge at 80 Knots - Strake Configuration

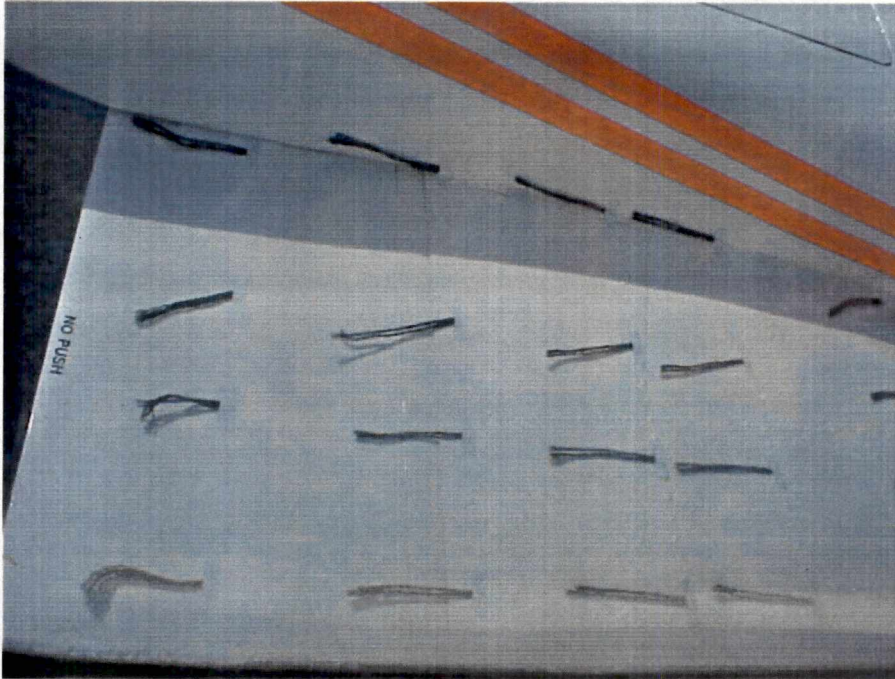


Figure 6.19g. Flow Visualization at Trailing Edge of Left Wing at 80 Knots - Baseline Configuration

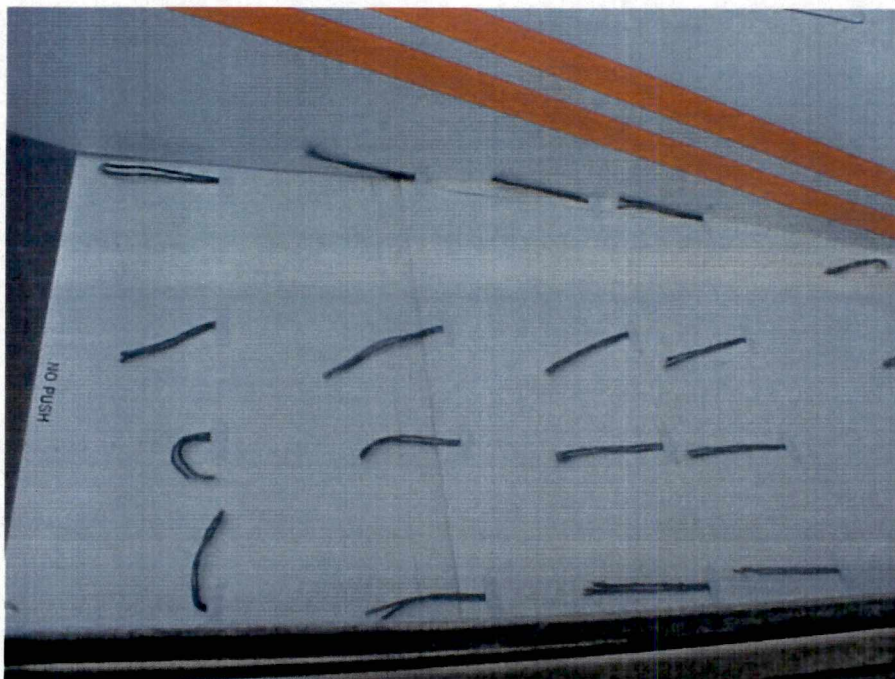


Figure 6.19h. Flow Visualization at Trailing Edge of Left Wing at 80 Knots - Strake Configuration

wing. It is difficult to determine any significant difference between the baseline and strake configurations by observing the tuft pictures. From this it can be inferred that the strakes' effects are not visually seen until the airplane is operating at very slow speeds near the stall.

CHAPTER VII

CONCLUSIONS AND RECOMMENDATIONS

Conclusions

The data gathered from the four flight tests performed in this study have shown that the delta wing strakes, configured at 18% overlap, 2% gap, do indeed enhance the performance of the Piper Navajo. The improvements in the flight tests are shown to be better than the improvements in the wind tunnel study.

The strakes significantly reduce the amount of power required to fly the Piper Navajo. The maximum range and endurance of the airplane are significantly improved. This was shown by the downward shifting of the power-required curve with the addition of the strakes.

Drag was reduced significantly by the addition of the strakes, as was shown by the reduction in the drag polar with the strakes added. This is relevant because it shows that less lift-induced drag is produced with increasing lift. Comparing Figures 2.13 and 6.11 show that the 18% overlap, 2% gap performed better in reducing the drag on the Piper Navajo than on the wind tunnel model. A reduction in drag on the Piper Navajo was seen at low lift coefficients whereas no significant reduction in drag was seen on the wind tunnel model until higher lift coefficients were reached. The airplane was also made more efficient as a result to the addition of the strakes.

The lift to drag ratio was greatly enhanced by the strakes. Speeds for maximum lift to drag and maximum endurance were reduced. Figures 2.14 and 6.14 also show that the strakes performed better on the Piper Navajo than on the wind tunnel model.

The strake configuration caused the maximum lift to drag ratio to decrease in the wind tunnel study. This 18% overlap, 2% gap configuration, however significantly improved the maximum lift to drag ratio of the Piper Navajo

Even though it was difficult determining the effect of the strakes on the point of separation, the flow visualization experiment hinted that the flow became smoother in the strake configurations at low speeds.

Despite the fact that the designers chose the 12% overlap, 5% gap configuration for the Cessna T303 Crusader, This thesis shows that the 18% overlap, 2% gap will also work well on aircraft. These improvements in performance parameters of the Piper Navajo, as well as the delta wing strake's success in previous experiments, indicate that it is a very useful device for reducing drag and improving overall performance on aircraft.

Recommendations

A study on the strakes' effects on climb performance is highly recommended. The reduction in the power required curve in this thesis suggests that the strakes would be able to significantly improve an aircraft's rate of climb.

Flight-tests to determine of the effects of the delta wing strake's effect on the stalling characteristics of general aviation aircraft is also suggested. The strake's ability to enable better-attached flow at low speeds hints that it would have an effect on the stall speed. Repeating the flow visualization experiment at speeds lower than 80 knots, closer to the stall speed is recommended in order to better visually see the effects of the strakes.

It is also recommended that further study be done to investigate the effects of span and sweep of parallel mounted strakes on general aviation planes. This would prove the relevance of Luckring's study (Luckring, 1979), which showed that strake sweep and span had an effect on lift and drag characteristics of fighter strake-wing integrated models.

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BIBLIOGRAPHY

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APPENDICES

APPENDIX A

AIRCRAFT AND MODEL SPECIFICATIONS

PA31-310 Piper Navajo

Wing Area	229 Sq. Ft
Wing Span	40.67 Ft.
Mean Aerodynamic Chord	5.63 Ft.
Root Chord	9.42 Ft.
Tip Chord	7.17 Ft.
Aspect Ratio	7.22
Maximum Gross Weight	6500 Lbs
Maximum Horsepower	620 HP. (310 HP./Eng)

Wallis' Wind Tunnel Model

Wing Area	7.4 Sq. Ft
Wing Span	9 Ft.
Mean Aerodynamic Chord	10.16 In
Aspect Ratio	10.95
Root Chord	12.83 In
Tip Chord	6.89 In
Nacelle Frontal Area	21.66 Sq. In.

APPENDIX B

DETERMINING THE BRAKE HORSEPOWER

The power being produced by the engine must be determined in order to be able to measure the drag that the airplane is producing in flight. Since in level, unaccelerated flight, thrust is equal to drag, by determining the brake horsepower being produced one can then reduce the data to determine the thrust produced and thus the drag.

In this thesis, the method used for determining the brake horsepower is the engine chart method. In this method, an engine chart is used to solve for brake horsepower when manifold pressure, RPM and pressure altitude are known. The conventional engine chart consists of two plots, a sea level chart and altitude chart. The sea level chart is plotted sea level brake horsepower versus manifold pressure and consists of a family of constant RPM lines. The altitude chart is plotted sea level brake horsepower versus pressure altitude and consists of the family of constant RPM lines and a family of vertical constant manifold pressure lines. A conventional engine power chart for the Lycoming IO-540 engine is shown in Figure B 1.

To use the conventional power chart to find the brake horsepower, first a straight vertical line is drawn from the known manifold pressure on the x-axis to the known RPM in the family of constant RPM lines on the sea level chart, as shown at point B in Figure B 1. A straight horizontal line is then drawn from that point to the corresponding brake horsepower value on the y-axis, as shown at point C in Figure B.1. On the altitude chart, a line is then drawn from the point where the given manifold pressure and the given RPM

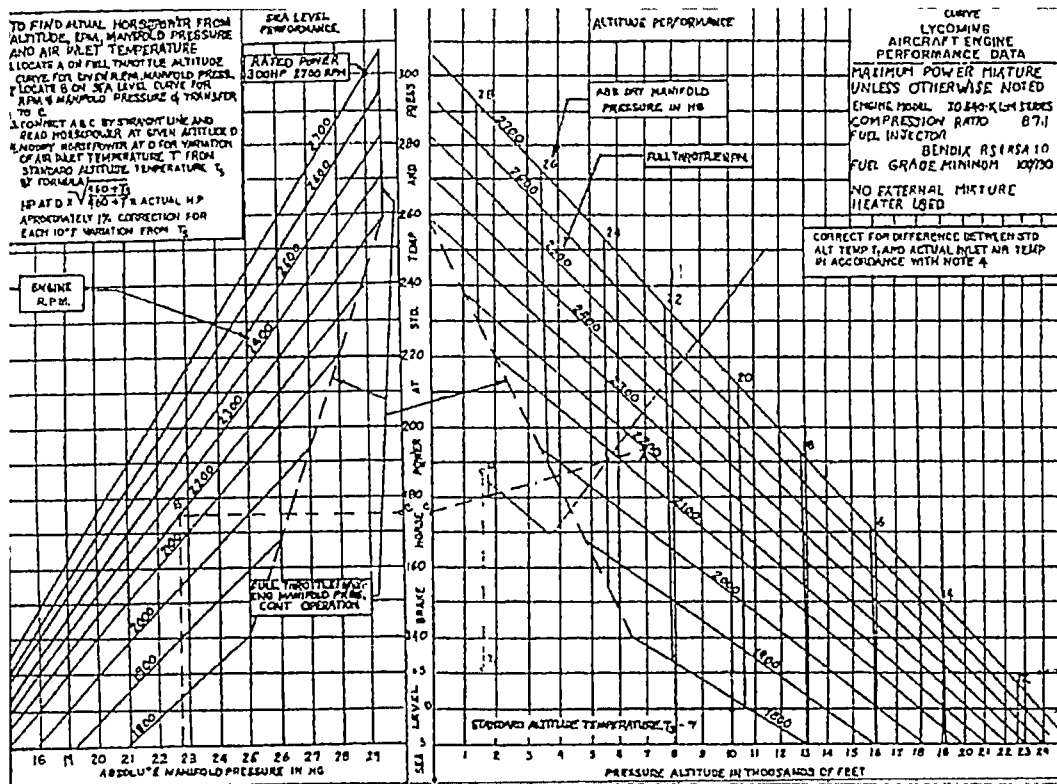


Figure B.1 Engine Power Chart for the Lycoming IO-540 Engine

value intersect (point A in Figure B.1) to the point C. This line is used to determine the corresponding sea level brake horsepower from the known pressure altitude

The type of engine power chart used for the Piper Navajo is different from the conventional engine power chart. The difference in this type of engine power chart from the conventional one is that it only has one constant RPM line instead of a family of them. The altitude chart on this type of power chart has a family of constant manifold pressure curves for the given RPM. The corresponding brake horsepower is determined from the given manifold pressure curve. An engine power chart for the Piper Navajo at 2400 RPM is shown in Figure B 2

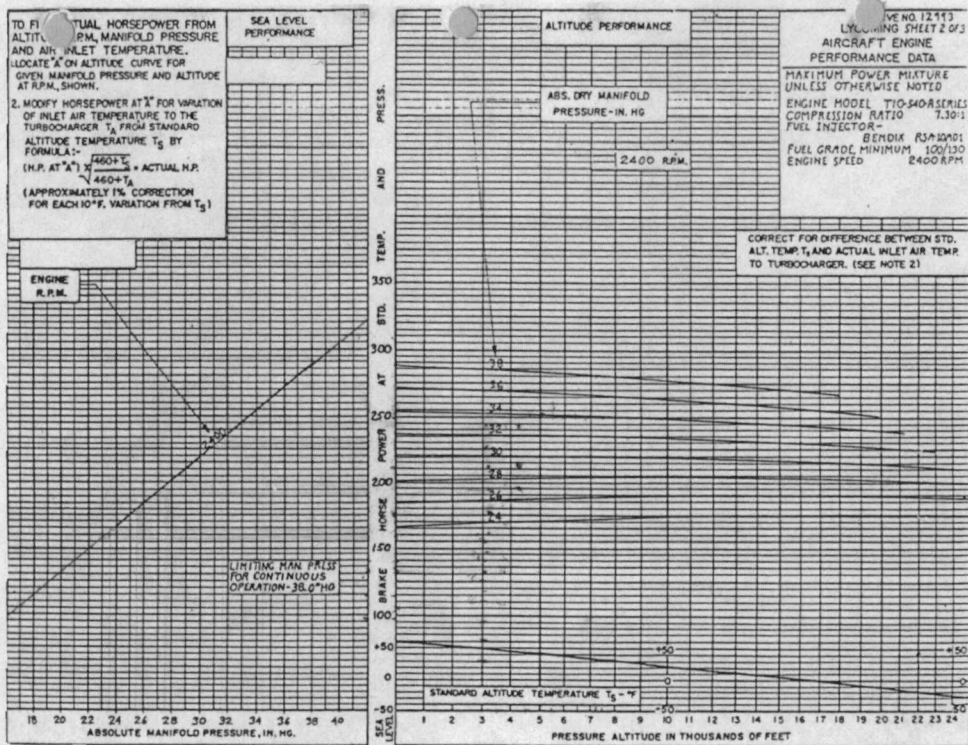


Figure B.2. Engine Power Chart for the Lycoming Engines in the Piper Navajo at 2400 RPM

APPENDIX C

EQUATIONS USED IN DATA REDUCTION

1.) Level Flight Performance Equations

The equations used to produce the power required curve (PIW, VIW) are

$$VIW = \frac{V_c}{\sqrt{\frac{W_T}{W_S}}}$$

Where V_C is the calibrated airspeed;

W_T is the aircraft test weight at that particular instant,

W_S is the standard maximum takeoff gross weight

$$PIW = \frac{BHP_T \sqrt{\sigma_T}}{\left(\frac{W_T}{W_S}\right)^{\frac{3}{2}}}$$

Where BHP_T is the brake horsepower at that instant,

σ_T is the density ratio at that instant;

W_S is the standard takeoff gross weight, which is set at the beginning of the flight. W_T is computed by observing the amount of gallons burned on the fuel meter; multiplying the number by six and subtracting that number from the standard takeoff gross weight, W_S . The density ratio is found by using the following equation.

$$\sigma = \frac{\delta}{\theta}$$

Where δ = pressure ratio (P_a/P_{SL}) and θ = temperature ratio (T_a/T_{SL})

The pressure ratio is determined using the following equation:

$$\delta = \frac{P_a}{P_{SL}} = \left(1 - 6.87535 \times 10^{-6} H_p\right)^{5.2561}$$

Where H_p is the pressure altitude at that instant;

The temperature ratio, θ , is found by taking the outside air temperature in Celsius and using the following equation.

$$\theta = \frac{(T_a + 273.15)}{288.15}$$

Where 288.15 is the absolute temperature at standard sea level conditions in degrees Kelvin

The brake horsepower is determined using the method discussed previously.

2.) Equations used in computing Lift and Drag Coefficients

The lift coefficient is obtained using the following equation:

$$C_L = \frac{2L}{\rho_o V_{IW}^2 S}$$

Where S is the wing area and ρ_o is the standard sea level density, 0.002377 slugs per cubic feet. The V_{IW} value obtained in a PIW - V_{IW} plot is used to make the C_L value good for all air densities and weights. In level and unaccelerated flight, however, lift is equal to weight, therefore the equation becomes

$$C_L = \frac{2W_s}{\rho_o V_{IW}^2 S}$$

Where W_s is the standard maximum takeoff gross weight.

The drag coefficient is found by using the PIW and V_{IW} values obtained in the PIW - V_{IW} plot. This PIW value is used in the equation:

$$F_R = \frac{550 PIW \eta_p}{V_{IW}}$$

Where F_R is the thrust required in pounds and η_p is the propeller efficiency. When using the PIW-VIW method, one assumption that is made is that the propeller efficiency is constant. In this case a constant value of 0.83 for propeller efficiency is used. In level and unaccelerated flight, thrust required is equal to drag, just as lift is equal to weight.

The thrust required computed from this equation is then used in the following equation to obtain the drag coefficient.

$$C_D = \frac{2D}{\rho_o VIW^2 S} = \frac{2F_R}{\rho_o VIW^2 S}$$

Where D is the drag.

3.) Oswald's Efficiency Factor

Oswald's efficiency factor is determined by the equation.

$$e = \frac{1}{\pi AR \frac{dC_D}{dC_L^2}}$$

Where AR is the aspect ratio and dC_D/dC_L^2 is the slope of the C_D versus C_L^2 line.

4.) Deriving Airspeed from Lift Coefficient

It is necessary to derive the corresponding airspeed from the lift coefficient equation in order to find out such things as what the airspeed for the maximum lift to drag ratio on lift to drag ratio versus lift coefficient plots. This airspeed value is then compared to the value corresponding to the point where a line starting from the origin on a PIW-VIW plot is put tangent to the PIW-VIW curve. The airspeed, in knots, knowing lift coefficient is derived by

$$V = \frac{\sqrt{\frac{2L}{\rho_o S C_L}}}{1.689} = \frac{\sqrt{\frac{2W_s}{\rho_o S C_L}}}{1.689}$$

Where 1 689 is the conversion factor to obtain the value in knots

APPENDIX D

INSTRUMENT CORRECTION CHARTS

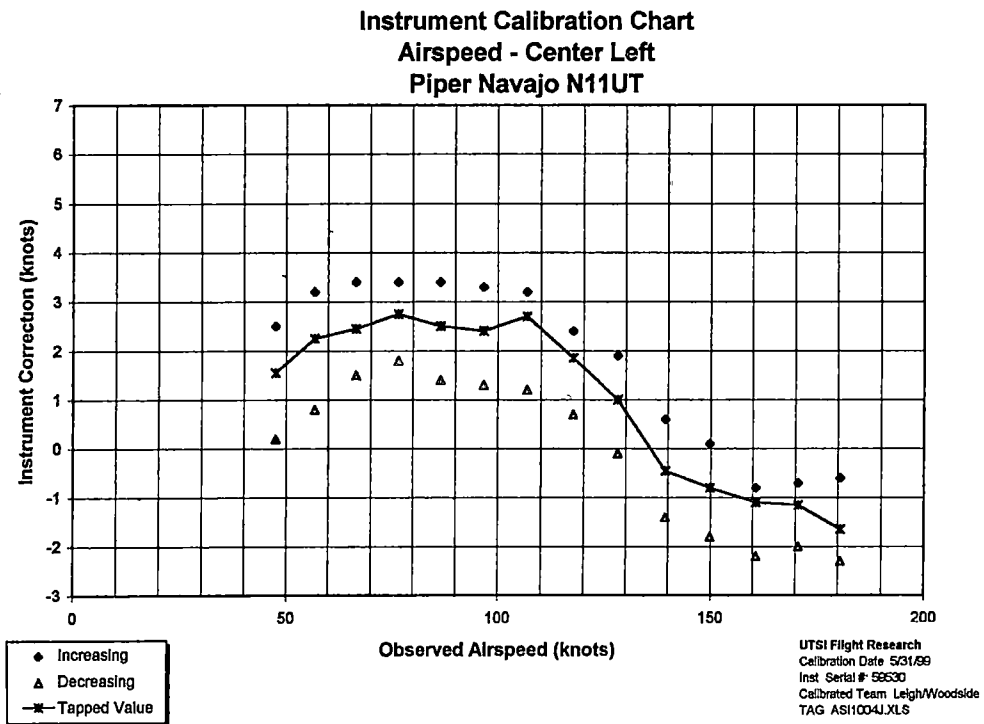


Figure D.1. Instrument Correction Chart to Obtain Indicated Airspeed from Observed Airspeed for Center Left Student Console in Piper Navajo

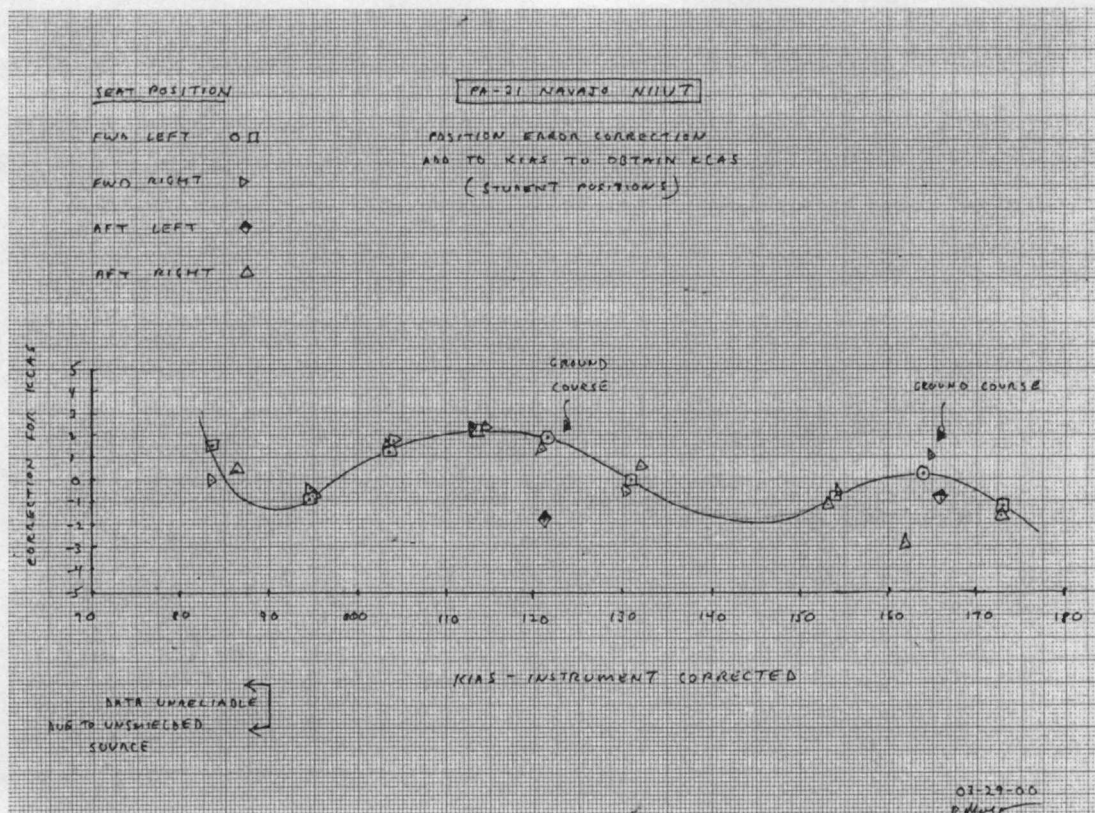


Figure D.2. Position Correction Chart to Obtain Calibrated Airspeed from Indicated Airspeed for all Student Consoles in Piper Navajo

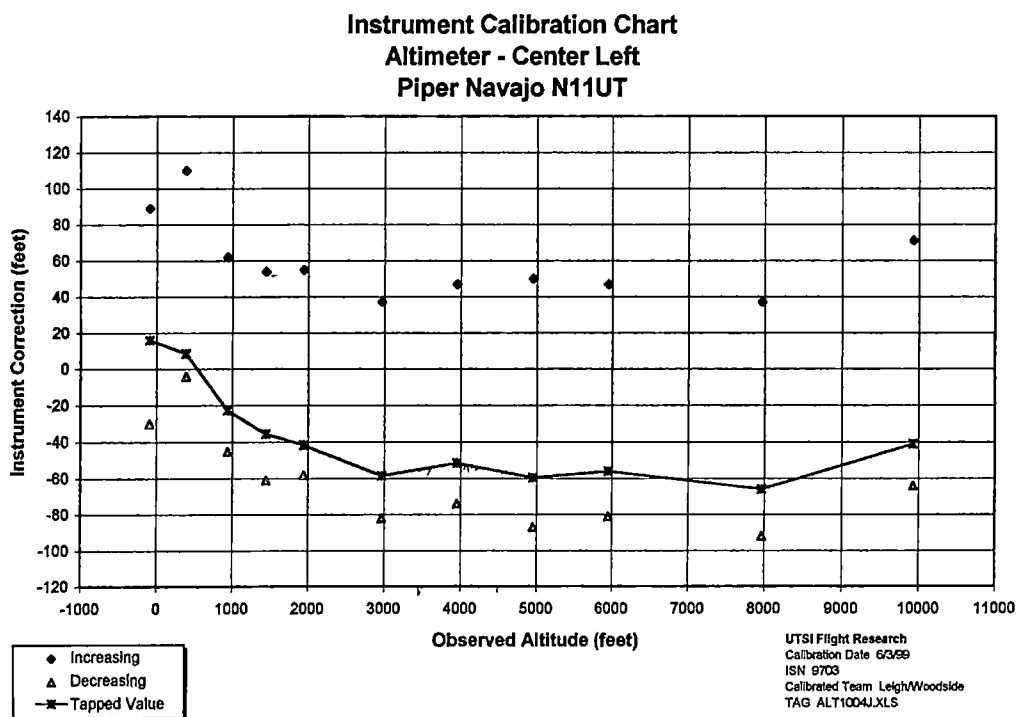


Figure D.3. Altimeter Correction Chart for Center Left Console in Piper Navajo

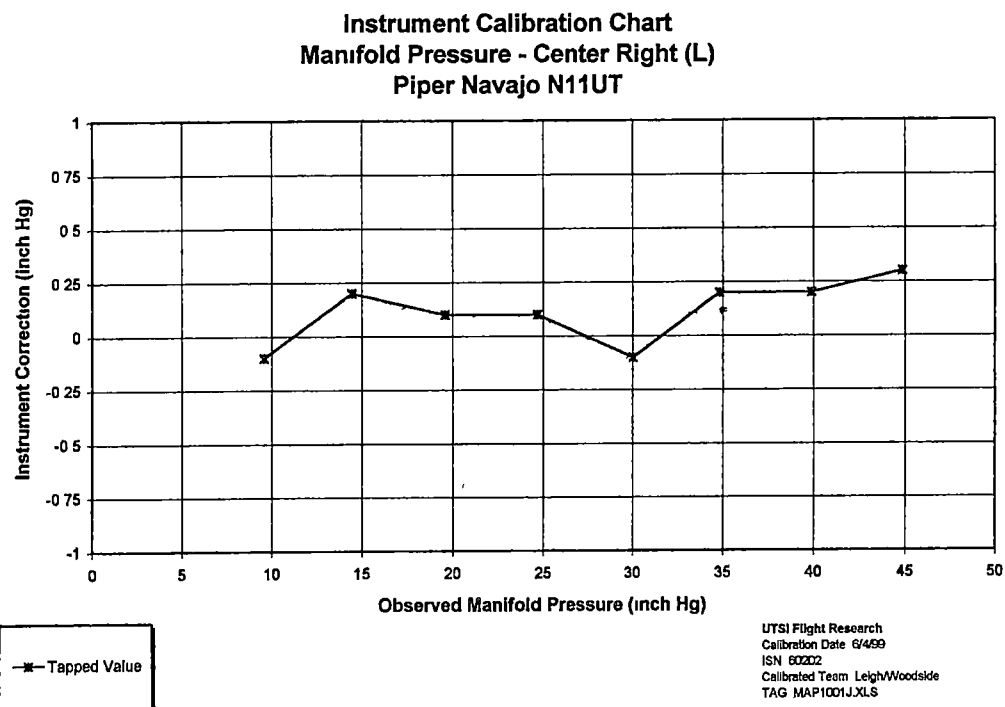


Figure D.4. Manifold Pressure Correction Chart for Center Left Console in Piper Navajo

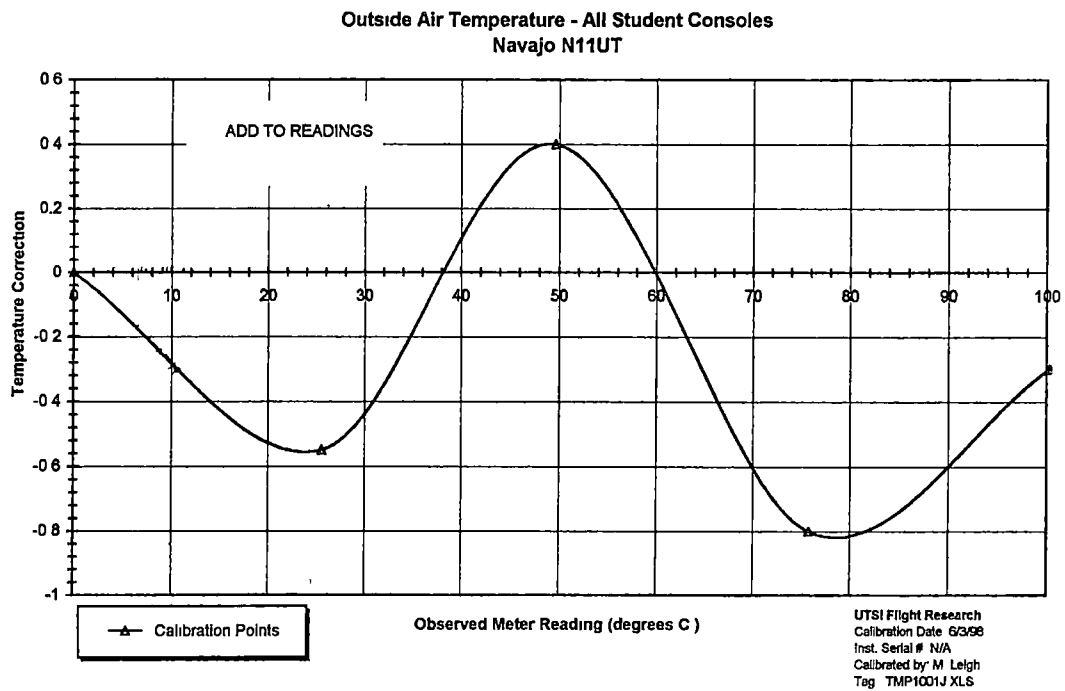


Figure D.5. Ambient Air Temperature Correction Chart for Consoles in Piper Navajo

APPENDIX E

DATA REDUCTION TABLES

Level Flight Performance Data, Baseline Flight Test

9/27/00

Certified Gross Aircraft Weight = 6500 lb

Beginning Gross Aircraft Weight = 6491 lb.

Table E.1. Instrument Corrected Parameters and Computed Temperature, Pressure and Density Ratios for Baseline Configuration

Run #	V _C (kts)	H _P (ft)	T _a (K)	RPM	MP (in Hg)	W _T (lb)	θ	δ	σ
1	165	4198	288 24	2400	33 63	6437	1 000	0 85733	0 857
2	162	4206	288 04	2400	31 50	6425	1 000	0 85707	0 857
3	156 7	4189	287 64	2400	29 50	6419	0 998	0 85761	0 859
4	149 8	4189	287 445	2400	27 49	6413	0 998	0 85761	0 860
5	139 65	4149	287 64	2400	25 56	6401	0 998	0 85889	0 860
6	128	4089	287 74	2400	23 71	6389	0 999	0 86081	0 862
7	121 7	4049	287 64	2400	21 61	6383	0 998	0 86209	0 864
8	108 9	4069	286 96	2400	19 12	6383	0 996	0 86145	0 865
9	98 6	4029	286 76	2400	16 64	6383	0 995	0 86273	0 867
10	89 1	3989	286 57	2400	17 13	6377	0 994	0 86401	0 869
11	83 8	3989	286 57	2400	19 12	6377	0 994	0 86401	0 869
12	174 25	3496	284 72	2400	36 20	6210 4	0 988	0 87996	0 891
13	169 45	3496	284 44	2400	33 06	6204 40	0 987	0 87996	0 891
14	164	3506	284 24	2400	31 55	6198 40	0 986	0 87963	0 892
15	156 8	3526	284 12	2400	30 09	6186 40	0 986	0 87898	0 891
16	150 9	3506	283 94	2400	27 50	6174 40	0 985	0 87963	0 893
17	142 4	3516	283 66	2400	25 58	6168 40	0 984	0 87931	0 893
18	130 7	3466	283 37	2400	23 61	6162 40	0 983	0 88094	0 896
19	120 9	3446	282 98	2400	21 11	6156 40	0 982	0 88159	0 898
20	107 2	3405	282 98	2400	20 11	6150 40	0 982	0 88293	0 899
21	98 7	3415	282 98	2400	17 13	6144 4	0 982	0 8826	0 899
22	86 65	3425	282 88	2400	20 11	6138 40	0 982	0 88227	0 899

Table E.2 Calculated PIW, VIW, C_L , F_R , C_D and L/D Values for Baseline Configuration

Run #	$\sqrt{\sigma}$	BHP _T	BHP _W	V _W (kts)	PIW x VIW	VIW ⁴ (kts ⁴)	PIW (HP)	C_L	C_L^2	F_R (lb)	C_D	L/D
								0	0.000		0.0278	0.00
								0.1	0.010		0.02861	3.50
								0.2	0.040		0.03102	6.45
								0.25	0.063		0.03284	7.61
1	0.926	494	464.07	165.81	76945.489	7.56E+08	462.51	0.305	0.093	753.93	0.035	8.62
2	0.926	460	433.43	162.94	70623.941	7.05E+08	445.66	0.315	0.099	739.24	0.036	8.79
3	0.927	420	396.69	157.69	62552.983	6.18E+08	416.55	0.337	0.113	713.99	0.037	9.10
4	0.927	400	378.46	150.81	57076.992	5.17E+08	381.99	0.368	0.135	684.58	0.039	9.49
5	0.928	368	349.31	140.73	49156.477	3.92E+08	338.24	0.423	0.179	649.62	0.042	10.01
6	0.928	334	318.23	129.11	41085.386	2.78E+08	297.83	0.502	0.252	623.48	0.048	10.43
7	0.929	301	287.45	122.81	35302.259	2.27E+08	280.29	0.555	0.308	616.85	0.053	10.54
8	0.930	236	225.56	109.89	24787.791	1.46E+08	253.80	0.693	0.481	624.22	0.067	10.41
9	0.931	206	197.10	99.50	19611.75	9.80E+07	241.86	0.846	0.715	656.99	0.085	9.89
10	0.932	208	199.51	89.96	17947.204	6.55E+07	238.59	1.035	1.070	716.86	0.114	9.07
11	0.932	236	226.37	84.60	19151.897	5.12E+07	240.21	1.170	1.368	767.38	0.138	8.47
12	0.944	542	547.68	178.27	97633.472	1.01E+09	544.22	0.263	0.069	825.12	0.033	7.88
13	0.944	486	492.05	173.44	85340.999	9.05E+08	510.93	0.278	0.077	796.20	0.034	8.16
14	0.944	456	462.42	167.94	77660.697	7.96E+08	475.55	0.297	0.088	765.32	0.035	8.49
15	0.944	442	449.46	160.73	72239.542	6.67E+08	433.10	0.324	0.105	728.31	0.036	8.92
16	0.945	404	412.30	154.83	63835.488	5.75E+08	401.70	0.349	0.122	701.24	0.038	9.27
17	0.945	362	370.09	146.18	54098.899	4.57E+08	360.87	0.392	0.154	667.24	0.040	9.74
18	0.946	356	365.01	134.23	48996.478	3.25E+08	314.36	0.465	0.216	632.96	0.045	10.27
19	0.947	282	289.87	124.23	36009.793	2.38E+08	283.97	0.542	0.294	617.83	0.052	10.52
20	0.948	265	273.00	110.20	30085.851	1.48E+08	254.29	0.689	0.475	623.66	0.066	10.42
21	0.948	208	214.55	101.52	21780.585	1.06E+08	243.51	0.812	0.660	648.33	0.081	10.03
22	0.948	265	273.75	89.17	24408.915	6.32E+07	238.67	1.053	1.109	723.44	0.117	8.98

Table E.3. Parameters Taken during Flight Test for Baseline Configuration

Run #	V _O (kts)	V _I (kts)	H _{PO} (ft)	T _O °C	T _C °C	RPM _O	MP _O (in Hg)
1	166	164.8	4250	15.5	15.08	2400	33.5
2	163	161.8	4260	15.3	14.88	2400	31.5
3	158	156.9	4240	14.9	14.48	2400	29.6
4	152	151.2	4240	14.7	14.285	2400	27.5
5	142	141.45	4200	14.9	14.48	2400	25.5
6	126	127.2	4140	15	14.58	2400	23.6
7	118	119.7	4100	14.9	14.48	2400	21.5
8	104.5	107.1	4120	14.2	13.8	2400	19
9	96	98.4	4080	14	13.6	2400	16.5
10	88	90.5	4040	13.8	13.41	2400	17
11	80	82.7	4040	13.8	13.41	2400	19
12	177	175.35	3550	11.9	11.56	2400	36
13	171	169.85	3550	11.6	11.28	2400	33
14	165	163.8	3560	11.4	11.08	2400	31.5
15	158	157	3580	11.3	10.96	2400	30
16	153	152.1	3560	11.1	10.78	2400	27.5
17	145	144.4	3570	10.8	10.5	2400	25.5
18	130	130.7	3520	10.5	10.21	2400	23.5
19	117	118.8	3500	10.1	9.82	2400	21
20	103	105.6	3460	10.1	9.82	2400	20
21	96	98.4	3470	10.1	9.82	2400	17
22	85	87.65	3480	10	9.72	2400	20

Level Flight Performance, Flight Test with Strakes Installed

10/24/00

Certified Gross Aircraft Weight = 6500 lb.

Beginning Gross Aircraft Weight = 6509 lb.

Table E.4. Instrument Corrected Parameters and Computed Temperature, Pressure and Density Ratios for Strake Configuration

Run #	V _C (kts)	H _P (ft)	T _a (K)	RPM	MP (in Hg)	W _T (lb)	θ	δ	σ
1	170.7	2470	293.42	2400	34.65	6461	1.018	0.91391	0.898
2	166.85	2550	293.23	2400	32.56	6449	1.018	0.91122	0.895
3	163.1	2520	293.23	2400	29.90	6443	1.018	0.91223	0.896
4	155.6	2510	293.23	2400	28.91	6431	1.018	0.91256	0.897
5	147.4	2550	293.04	2400	27.00	6425	1.017	0.91122	0.896
6	140.1	2530	292.84	2400	25.10	6419	1.016	0.91189	0.897
7	133.5	2510	292.64	2400	23.10	6413	1.016	0.91256	0.899
8	125.8	2490	292.05	2400	20.10	6407	1.013	0.91324	0.901
9	116.6	2450	291.65	2400	18.61	6401	1.012	0.91458	0.904
10	103.8	2350	291.85	2400	17.62	6401	1.013	0.91795	0.906
11	86.9	2310	291.55	2400	20.10	6401	1.012	0.9193	0.909
12	164.6	7058	284.44	2400	34.40	6341	0.987	0.76993	0.780
13	162	7078	284.24	2400	31.45	6335	0.986	0.76935	0.780
14	154.4	7158	283.56	2400	29.90	6323	0.984	0.76701	0.779
15	147.95	7198	283.26	2400	27.50	6317	0.983	0.76585	0.779
16	141.85	7058	283.66	2400	25.57	6305	0.984	0.76993	0.782
17	132.8	7088	283.56	2400	23.60	6299	0.984	0.76905	0.782
18	126.6	7078	283.46	2400	21.60	6299	0.984	0.76935	0.782
19	121	6958	283.46	2400	20.10	6293	0.984	0.77286	0.786
20	106.7	6918	283.26	2400	17.62	6287	0.983	0.77403	0.787
21	91.15	6928	283.36	2400	16.15	6287	0.983	0.77374	0.787
22	83.75	6898	283.26	2400	18.61	6281	0.983	0.77462	0.788

Table E.5. Calculated PIW, VIW, C_L , F_R , C_D and L/D Values for Strake Configuration

								C _L	C _L ²	F _R (lb)	C _D	L/D
								0	0 000		0 0278	0 00
								0 1	0 010		0 02844	3 52
								0 15	0 023		0 02923	5 13
Run #	√G	BHP _T	BHP _W	V _W (kts)	PIW x VIW	VIW ⁴ (kts ⁴)	PIW (HP)	0 2	0 040		0 03034	6 59
1	0 947	516	493 28	171 21	84456 579	8 59E+08	476 25	0 286	0 082	751 81	0 033	8 65
2	0 946	481	460 58	167 51	77150 392	7 87E+08	452 39	0 298	0 089	729 95	0 034	8 90
3	0 947	439	421 18	163 82	68997 509	7 20E+08	429 82	0 312	0 097	709 14	0 034	9 17
4	0 947	422	406 08	156 43	63523 742	5 99E+08	388 04	0 342	0 117	670 44	0 035	9 70
5	0 947	390	375 66	148 26	55693 97	4 83E+08	347 00	0 381	0 145	632 60	0 037	10 28
6	0 947	355	342 67	140 98	48309 448	3 95E+08	314 92	0 421	0 177	603 75	0 039	10 77
7	0 948	322	311 47	134 40	41862 297	3 26E+08	289 43	0 463	0 215	582 03	0 041	11 17
8	0 949	259	251 23	126 71	31832 979	2 58E+08	263 73	0 521	0 272	562 55	0 045	11 55
9	0 951	236	229 57	117 50	26973 553	1 91E+08	238 67	0 606	0 368	549 00	0 051	11 84
10	0 952	218	212 37	104 60	22214 189	1 20E+08	213 88	0 765	0 585	552 65	0 065	11 76
11	0 953	258	251 66	87 57	22037 302	5 88E+07	199 83	1 092	1 192	616 78	0 104	10 54
12	0 883	510	467 47	166 65	77903 775	7 71E+08	447 04	0 301	0 091	725 02	0 034	8 97
13	0 883	457	419 47	164 10	68833 485	7 25E+08	431 47	0 311	0 097	710 66	0 034	9 15
14	0 883	439	403 97	156 55	63239 331	6 01E+08	388 65	0 342	0 117	671 00	0 035	9 69
15	0 883	404	372 20	150 08	55859 375	5 07E+08	355 68	0 372	0 138	640 54	0 037	10 15
16	0 884	366	338 82	144 03	48798 956	4 30E+08	327 85	0 404	0 163	615 24	0 038	10 57
17	0 884	330	305 81	134 90	41254 203	3 31E+08	291 25	0 460	0 212	583 52	0 041	11 14
18	0 884	290	268 84	128 60	34573 789	2 74E+08	269 65	0 506	0 256	566 70	0 044	11 47
19	0 886	259	240 99	122 97	29635 861	2 29E+08	252 82	0 554	0 306	555 66	0 047	11 70
20	0 887	218	203 36	108 49	22063 01	1 39E+08	220 10	0 711	0 506	548 31	0 060	11 85
21	0 887	170	158 53	92 68	14692 321	7 38E+07	201 74	0 975	0 950	588 33	0 088	11 05
22	0 888	236	220 55	85 20	18790 345	5 27E+07	199 65	1 153	1 330	633 38	0 112	10 26

Table E.6. Parameters Taken during Flight Test for Strake Configuration

Run #	V _O (kts)	V _I (kts)	H _{PO} (ft)	T _O °C	T _C °C	RPM _O	MP _O (in Hg)
1	173	171.7	2520	20.8	20.26	2400	34.5
2	168	166.85	2600	20.6	20.07	2400	32.5
3	164	162.9	2570	20.6	20.07	2400	30
4	157	156	2560	20.6	20.07	2400	29
5	150	149.2	2600	20.4	19.88	2400	27
6	142.5	141.9	2580	20.2	19.68	2400	25
7	134	134.3	2560	20	19.48	2400	23
8	123	124.4	2540	19.4	18.89	2400	20
9	112	114.4	2500	19	18.49	2400	18.5
10	100	102.6	2400	19.2	18.69	2400	17.5
11	85.5	88	2360	18.9	18.39	2400	20
12	165.5	164.4	7120	11.6	11.28	2400	33.5
13	163	161.8	7140	11.4	11.08	2400	31.5
14	156	155	7220	10.7	10.4	2400	30
15	150.5	149.65	7260	10.4	10.1	2400	27.5
16	144.5	143.85	7120	10.8	10.5	2400	25.5
17	133	133.4	7150	10.7	10.4	2400	23.5
18	124	125.4	7140	10.6	10.3	2400	21.5
19	117	118.8	7020	10.6	10.3	2400	20
20	102.5	105.1	6980	10.4	10.1	2400	17.5
21	90	92.45	6990	10.5	10.2	2400	16
22	78	80.65	6960	10.4	10.1	2400	18.5

APPENDIX F

WEIGHT & BALANCE

Table F.1. Weight and Balance Sheet for Baseline Level Flight Performance Test

ITEM	WT. (lb)	ARM (in.)	MOMENT (lb-ft.)
Airplane	4,444.00	127.80	567,943.20
Pilot's Seat	180 00	119.00	21,420.00
Copilot's Seat	170.00	119 00	20,230.00
Right Center Seat	150 00	159.00	23,850 00
Left Center Seat	150.00	159 00	23,850.00
Right Rear Seat	160.00	198 00	31,680.00
Left Rear Seat	157 00	198.00	31,086.00
Fwd. Baggage		43 00	0.00
Rear Baggage		255.00	0 00
Rt. Nac. Baggage		145.00	0.00
(Fwd Half)			0 00
Rt. Nac. Baggage		192 00	0.00
(Rear Half)			0 00
Lft. Nac. Baggage		145 00	0 00
(Fwd. Half)			0.00
Lft. Nac. Baggage		192.00	0.00
(Rear Half)			0.00
Inboard Fuel	636 00	126.80	80,644.80
Outboard Fuel	444.00	148 00	65,712.00
Other			0 00
Total Weight	6,491 00	Total Moment	866,416.00

Takeoff C G = 133 48 inches aft of datum

Table F.2. Weight and Balance Sheet for Baseline Flight with Tufts

ITEM	WT (lb.)	ARM (in.)	MOMENT (lb-ft)
Airplane	4,444.00	127 80	567,943 20
Pilot's Seat	180.00	119 00	21,420 00
Copilot's Seat	120.00	119 00	14,280.00
Right Center Seat	170.00	159 00	27,030.00
Left Center Seat	170.00	159.00	27,030.00
Right Rear Seat	230 00	198.00	45,540.00
Left Rear Seat	115 00	198.00	22,770.00
Fwd. Baggage		43 00	0 00
Rear Baggage		255.00	0.00
Rt Nac. Baggage		145 00	0.00
(Fwd. Half)			0.00
Rt. Nac Baggage		192 00	0 00
(Rear Half)			0 00
Lft. Nac. Baggage		145.00	0.00
(Fwd. Half)			0.00
Lft. Nac. Baggage		192.00	0.00
(Rear Half)			0.00
Inboard Fuel	636 00	126 80	80,644.80
Outboard Fuel	444.00	148 00	65,712.00
Other			0.00
Total Weight	6,509.00	Total Moment	872,370 00

Takeoff C G. = 134 03 inches aft of datum

Table F.3. Weight and Balance Sheet for Level Flight Performance Test with Strakes Mounted

ITEM	WT (lb.)	ARM (in)	MOMENT (lb-ft)
Airplane	4,444.00	127.80	567,943.20
Pilot's Seat	180.00	119.00	21,420.00
Copilot's Seat	120.00	119.00	14,280.00
Right Center Seat	170.00	159.00	27,030.00
Left Center Seat	170.00	159.00	27,030.00
Right Rear Seat	172.50	198.00	34,155.00
Left Rear Seat	172.50	198.00	34,155.00
Fwd Baggage		43.00	0.00
Rear Baggage		255.00	0.00
Rt. Nac. Baggage		145.00	0.00
(Fwd. Half)			0.00
Rt. Nac Baggage		192.00	0.00
(Rear Half)			0.00
Lft. Nac. Baggage		145.00	0.00
(Fwd. Half)			0.00
Lft. Nac. Baggage		192.00	0.00
(Rear Half)			0.00
Inboard Fuel	636.00	126.80	80,644.80
Outboard Fuel	444.00	148.00	65,712.00
Other			0.00
Total Weight	6,509.00	Total Moment	872,370.00

Takeoff C G = 134.03 inches aft of datum

Table F.4. Weight and Balance Sheet for Flight with Tufts and Strakes

ITEM	WT. (lb.)	ARM (in.)	MOMENT (lb-ft.)
Airplane	4,444.00	127.80	567,943.20
Pilot's Seat	180.00	119.00	21,420.00
Copilot's Seat	120.00	119.00	14,280.00
Right Center Seat	170.00	159.00	27,030.00
Left Center Seat	170.00	159.00	27,030.00
Right Rear Seat	230.00	198.00	45,540.00
Left Rear Seat	115.00	198.00	22,770.00
Fwd. Baggage		43.00	0.00
Rear Baggage		255.00	0.00
Rt. Nac. Baggage		145.00	0.00
(Fwd. Half)			0.00
Rt. Nac. Baggage		192.00	0.00
(Rear Half)			0.00
Lft. Nac. Baggage		145.00	0.00
(Fwd. Half)			0.00
Lft. Nac. Baggage		192.00	0.00
(Rear Half)			0.00
Inboard Fuel	636.00	126.80	80,644.80
Outboard Fuel	444.00	148.00	65,712.00
Other			0.00
Total Weight	6,509.00	Total Moment	872,370.00

Takeoff C.G. = 134.03 inches aft of datum

VITA

Courthney Carlos Cumberbatch was born the oldest of three children on December 24, 1974 in Washington, D.C. He is the son of Ival and Carlos Cumberbatch. When he was two years old, Courthney's parents moved to Silver Spring, Maryland where he has lived the majority of his life. He attended Paint Branch High school in the Montgomery County, Maryland public school system. While there, he participated on the track and field, soccer and cross-country teams. He graduated from Paint Branch in the spring of 1993.

Courthney received a full honors scholarship to Morgan State University in Baltimore, Maryland. While at Morgan State he participated on the track and field team and in the National Society of Black Engineers (NSBE). He graduated from Morgan State University with a Bachelor of Science in electrical engineering in December of 1997.

After completing his undergraduate, he went on to receive a graduate research assistantship at the University of Tennessee Space Institute in Tullahoma, Tennessee. His assistantship work dealt with calibrating a shear sensitive liquid crystal coating wind tunnel for the use of shear stress measurement. He continued to be active in NSBE and was the secretary and telecommunications chair for the local chapter. He graduated from the University of Tennessee with a Master of Science in aviation systems in December of 2000. Courthney plans to pursue a career in aviation as either a flight test engineer or an aircraft designer.