Experimental Determination of Lift and Lift Distributions for Wings In Formation Flight

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(ABSTRACT)

Experimental methods for the investigation of trailing vortex strengths, total lift, and lift distributions for three-dimensional wings in close proximity flight were developed. With these experiments we model compound aircraft flight either docked tip-to-tip, or flying in formation. There is a distinct lack of experimental formation flight data using three-dimensional wing models for tests. The absence of fixed walls on either end of the wing permits the development of the asymmetric shedding of vortices, and the determination of the asymmetric circulation distribution induced by the proximity of the leading wing. The pair consisted of a swept NACA-0012 non-cambered wing simulating one half of a leading aircraft and a rectangular cambered NACA 63-420 wing simulating the trailing aircraft. Important aspects of the work included theoretical development, experimental setup, data acquisition and processing, and results validation. Experimentally determining the lift for formation flight, in addition to the local flow behavior for a pair of wings, can provide valuable insight for the proposition of flying actual aircraft in formation to increase mission efficiency. To eliminate the need for bulky mounting stings and direct load measurement devices that can potentially interfere with the local flowfield, a minimally invasive velocity probe method is developed. A series of experiments were performed to assist with the development of the method. Velocity and vorticity distributions obtained along a near-field plane were processed to calculate wingtip vortex strengths. Additionally, vortex position instabilities and the shedding of vorticity inboard of the wingtips were observed. To determine the circulation distributions for the trailing wing, the initial method is modified. By processing velocity information acquired in a near-field plane, both the lift and induced drag were calculated for the trailing airfoil. Comparisons are made to directly measured loads and to results reported earlier. Directly measured lift and drag coefficients were found to agree with existing literature.

Dedication

Creation seems to come from imperfection. Creation of a significant entity when beginning with something imperfect is the ability to achieve. To every person who cannot, will not, and then does not give up nor give in when trying to accomplish what seems to be the impossible. You will prevail.

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Nomenclature

$\{X,Y,Z\}$	Global Coordinate
$\{x,y,z\}$	Local Coordinate
α	Angle of Attack
L	Lift
D	Drag
D_i	Induced drag
ρ	Flow Density
Γ	Circulation
U_o	Free Stream Velocity
u, v, w	Velocity Components
c	Chord of trailing wing
C	Chord of leading wing
B	Span of leading wing
b	Span of trailing wing
P	Pressure
AR	Aspect Ratio
C_L	Lift Coefficient
C_D	Drag Coefficient
C_m	Rolling Moment Coefficient
η	Spanwise Separation
ξ	Streamwise Separation

Chapter 1

1 BACKGROUND AND INTRODUCTION

1.1 Motivation

It has been known for decades that birds flying in formation take advantage of aerodynamic interference that reduces the drag for the formation. Similar benefits are available to aircraft that fly in formation. This drag reduction can also be realized as an increase in lift, either for each individual aircraft, or for the formation as a whole. Usually it is the trailing aircraft that receives the greatest individual benefit. In either case, the drag reduction or lift increase is beneficial to mission efficiency and can result in substantial cost savings. Studies of these benefits have been a topic of aerospace interest for many years.

A majority of the previous and current research related to the benefits of formation flight is either computational or theoretical work. There has been a small amount of experimental research performed related directly to the measurement of formation lift and drag and the flow effects for three-dimensional multiple aircraft systems, however, research studies are few and far between due to cost, required equipment, and experimental testing method constraints. Experimental work is generally more expensive (due to material, model fabrication, and equipment costs) and often more time consuming than other methods. Of these experimental studies, virtually none of them have used near wake velocity measurements to calculate L/D benefits or lift distributions. When testing three-dimensional models (versus 2D models) in a wind tunnel, determining and implementing practical mounting and data acquisition systems can be challenging, especially for configurations requiring multiple wing or aircraft models. In addition, the performance of trailing models is highly dependent on the quality and condition of the oncoming flow. Large mounting apparatuses of upstream models can cause disruptions of the oncoming natural flow structure, and lead to inaccurate results. The problem is compounded as the number of models for a given experimental test increases. In spite of the few disadvantages, there is one considerable advantage of experimental work: Results obtained from well-planned and executed experimental research are generally more accurate than basic computational models. In some cases exact theoretical calculations can be performed yielding excellent results, but often the theory is not well enough developed for use with complex setups and will suffer the same accuracy loss as a generic computer model. To develop a better understanding of the flow structures and lift distributions for formation flight, a minimally invasive method utilizing a flow measurement probe and fully three-dimensional wing models was designed and implemented.

Recently, a topic of interest has been how to predict and determine the necessary flight inputs, or control laws, for automatic formation flight, or AFF. Automatic formation flight entails the development of either an active or passive system that can monitor dynamically changing flow conditions and react by adjusting various aircraft control mechanisms (i.e. flaps, rudders, engines) to keep the aircraft aloft and in control. As stated, the success of these control laws depends on the ability for the system to react to dynamically changing flow conditions. Before designing and implementing an automatic flight control system however, it is beneficial to understand how the flow over a given lifting surface behaves, and how it contributes to loads experienced by the aircraft. Although a rudimentary AFF system can be designed with only general knowledge of the anticipated flowfield and the resultant loading, a more thorough knowledge of both these parameters, including three-dimensional effects and their eccentricities, can lead to an even more robust design from the outset. The application of an instrumentation system as used for this research could prove to be a useful tool for such inquiries.

1.2 Introduction

The first studies of formation flight were basic theoretical postulations around the turn of the century. Several attempts as early as 1898 and into the mid 1900's, spawned by the observation of the formation flight of birds, were made at understanding the power reduction that occurs for formation flight. Because the physical background of the aerodynamic benefits had not been properly understood up to this point, explanations were often incorrect or unclear at best. One of the first investigators to accurately put down in writing, the theory of multiple lifting surfaces was Max M. Munk in 1921. Of a purely theoretical nature, his work took the known theory for a single wing and applied it to the case of multiple lifting elements by use of mathematical expressions. At first, his work was not meant to be a study of aeronautical interest, but rather a mathematical exercise.

Between the time of World War I and II, when very little attention was being paid publicly to military aviation, Italy began to develop its air forces with hopes to gain bargaining power with Germany in case of a future war. The Fascist dictator Mussolini hoped that developing an air force would give Italy valuable bargaining power in dealing with Germany. In 1933 Italo Balbo, Italy's Minister of Air, organized a flight from Europe to the Chicago World's Fair in the United States which involving twenty-five specially equipped twin-hulled Savoia-Marchetti SM-55X flying boats. This very impressive long-range formation flight was known as the Italian formation flight of 1933, and was performed as a display of air superiority. This came two years after general Balbo led a similar flight formation across the south Atlantic from Portuguese Guinea to Brazil on January 6, 1931. During this flight, twelve Savoia-Marchetti SM-55Xs were used. Wherever the group flew, they were always eagerly welcomed and treated like heroes⁴⁵. These

formation flights were not meant to be studies in aerodynamics, or intended to further the field of aeronautics, but rather only impressive demonstrations of power.

Several aircraft configurations involving more than one aircraft were trailed during the mid-1900's. Meyers illustrates several concepts for flying together³⁹. One of the first studies was completed by the Navy and consisted of the indirect docking of pursuit aircraft to special attach points on the dirigibles AKRON and MACON. This combination of aircraft was primarily intended for reconnaissance purposes. The pursuit aircraft would hang from the larger craft providing a means for their quick dispatch. Another venture with similar objectives, but utilizing a different type of transport, was the FICON (fighter conveyor) project. In this case a large B-36 bomber with the ability to fly long distances was used to transport fighter aircraft within its cargo hold³⁹. The fighter was an XF-85, and was an egg-shaped, jet-propelled, folding-wing, landing-gearless experimental aircraft specially designed for the mission. The long-term application of this project was unfortunately terminated due to unforeseen docking problems. One of the first true docked flight configurations was the well-known TOM-TOM project. This project was a follow up to the FICON project. Instead of storing the fighter craft inside a large cargo plane, the F-84 fighter was physically docked while in flight to the wingtip of the B-36 bomber. The purpose of wingtip-coupling several aircraft was to increase the range of the formation as a whole by effectively creating a single, larger span, lifting surface. Physically docking the smaller craft to the bombers' wingtip brought up the issues of flight stability and flight control systems. These issues are discussed by Anderson³. Other docked formation flight attempts included effective combinations of a C-47A and a Q-14B and later, the coupling of a B-29 with two F-84 aircraft. Numerous flights were accomplished, but always with aircraft under the manual control of the pilot. The next step in the program was the implementation of an automatic flight control system to maintain the proper flight attitude. Unfortunately, the control system was unable to properly adjust for the aerodynamic effects of coupled flight and both aircraft were destroyed.

After these initial inquiries, demonstrations, and trials, more definitive research began to better develop a further understanding of formation flight flow dynamics. In the following section several important topics related to wings and wings in formation flight are discussed. Section 1.3 provides general technical background on several topics of importance to formation flight analyses. Lift and drag, vorticity, the Biot-Savart law, and several ways of representing wings for theoretical and computational purposes are described. Readers with previous knowledge of these basic physical phenomena may choose not to pursue section 1.3. Following section 1.3, a review of existing formation flight research is explained. Sections 1.3.1 through 1.3.3 describe past theoretical, computational, and experimental research initiatives. An attempt is made to discuss the benefits and drawbacks of each different type of investigative method (theoretical, computational, or experimental) and to provide a basis for the author's choice to pursue an experimental

approach. Because the current research initiative is experimental, special attention is paid to previous experimental methods.

1.3 Technical Background and Methodology

Lift and Drag

Several definitions related to formation flight must first be established. Perhaps the most important parameters related to aircraft flight are lift and drag. The reduction of drag is one of the main objectives of many studies. Any body moving through a fluid medium experiences drag, therefore drag is usually the main parameter to be minimized. Often a lift-to-drag ratio C_L/C_D is calculated to compare various scenarios. Although it is the total drag that is ultimately important for aircraft performance, the total drag on the body, D , can be categorized into three subtypes of drag. These subtypes have been defined in such a way that each type corresponds to drag caused by different physical phenomenon. This method of characterization is used to simplify the evaluation of drag effects for particular research. Figure 1 shows the commonly accepted separation of the total drag into different subcategories.

Figure 1: Classification of drag.

The first subtype is the induced drag, denoted by D_i . This drag is a penalty of creating lift and is present for any wing generating lift. Often this type of drag is called wave drag. Induced drag is the drag created by the vortices at the tip of an aircraft's wing. Induced drag is the drag due to lift. The high pressure underneath the wing causes the airflow at the tips of the wings to curl around from bottom to top in a circular motion. This results in a trailing vortex. Induced drag increases in direct proportion to increases in the angle of attack. The circular motion creates a change in the angle of attack near the wing tip, which causes an increase in drag. The greater the angle of attack up to the critical angle (where a stall takes place), the greater the amount of lift developed and the greater the induced drag. For a lifting wing, vorticity is constantly being generated on its surface and being shed from the trailing edge creating a wake. The term vortex drag is commonly used when referring to the induced (or wave) drag caused solely by trailing vortical structures.

The second type of drag is called form drag since it is due to the shape, or form, of the aircraft. Form drag is related to the amount of force it takes to move a body through a fluid medium due to the area of the body exposed to the flow. The separation of flow creates turbulence and results in pockets of low and high pressure that leave a wake behind the wing (thus the name pressure drag). For any solid body having a measurable thickness, form drag will always be present. Form drag, when combined with the induced drag, is often referred to as pressure drag, and is a component of the total drag.

The last type of drag is friction (or viscous) drag. Skin friction drag is caused by the actual contact of the air particles against the surface of the aircraft. This is analogous to the friction caused because of the motion between any two objects or substances. Because skin friction drag is an interaction between a solid (the wing surface) and a fluid (the air), the magnitude of skin friction drag depends on the properties of both the solid and of the gas. Along any solid surface immersed in a fluid flow, a boundary layer of low energy flow is generated. The magnitude of the skin friction depends on the state of this flow. Typical aerodynamic conditions will yield turbulent boundary layers, although for low velocities or very small aircraft models, laminar boundary layers may develop. For turbulent boundary layers skin friction drag can be reduced, and airspeed can be increased, by keeping the wings' surface highly polished and clean. The contribution to the drag due to the fluid depends on the viscosity of the fluid. A higher viscosity results in an increase of the magnitude of the skin friction drag.

Similarly parasite drag is simply the sum of the form drag and the skin friction drag. The total drag is the sum of the induced drag and the parasite drag. Alternatively, the total drag can also be considered as the sum of the pressure drag and the skin friction drag¹⁴. Regardless of how the various types of drag are categorized, all of them must be accounted for when determining performance characteristics for subsonic flight. As was stated, one way to group the total drag is as a combination of the induced drag and the parasite drag. This is perhaps one of the most convenient categorizations, as most research attempts to reduce solely the induced drag, D_i , since it is the most readily reduced type of drag using a formation flight approach. The induced drag is directly related to the effects of the upwash and downwash created by all the aircraft in a formation. Although the total drag must eventually be used to evaluate the performance savings (such as range, or fuel usage) of an individual aircraft or a system as a whole, parametric studies of induced drag alone are useful when determining the effect due to various formation flight configurations. Most research performed in the area of formation flight assumes an inviscid and incompressible flow that simplifies the formulation even further.

Vorticity, Vortex Lines, Filaments, Vortices, and Circulation

To understand how lift and drag forces are created for a wing when placed in a fluid flow, and how these forces are affected by changes within the flow, several important concepts must be understood. These concepts will be used extensively in the development and explanation of the currently proposed methods. Vorticity, vortex lines, vortex tubes/filaments, vortices, and circulation are briefly defined. *Vorticity* is one of the most important flow effects to understand. Vorticity can be defined as a measure of the rotation of a fluid. Given an arbitrary velocity field, vorticity can be mathematically expressed as the curl of the velocity.

$$\vec{\omega} = \nabla \times \vec{V} \quad (1)$$

or

$$\vec{\omega} = \left(\frac{\partial w}{\partial y} - \frac{\partial v}{\partial z}\right)\hat{i} + \left(\frac{\partial u}{\partial z} - \frac{\partial w}{\partial x}\right)\hat{j} + \left(\frac{\partial v}{\partial x} - \frac{\partial u}{\partial y}\right)\hat{k} \quad (2)$$

Vortex lines and vortex tubes/filaments are two types of theoretical vortical structures created to facilitate the study of flow mechanics. Vortex lines are theoretical lines that are everywhere parallel to the vorticity vector. Vortex lines move with the fluid and cannot begin or end in the interior of the fluid. Although in theory, a vortex line must form a closed loop, for our practical case, they are assumed to extend to infinity. Although the concept of a vortex line may seem somewhat abstract, they prove to be invaluable when describing the fluid flow around a lifting surface. Vortex tubes are collections of vortex lines. A vortex tube moves with the fluid and always consists of the same fluid elements. A vortex tube with an infinitely small cross-sectional area is called a vortex filament.

The term *vortex* is a term loosely defined, and is often used to describe any flow structure containing vorticity. In three-dimensional space, a vortex tube could be considered a vortex. Because a vortex tube has constant vorticity along its length, the vortex would have constant vorticity along its length. This is an idealization though, because in real fluid flows viscous effects tend to diminish the strength of any free vortical structure. Acknowledging this fact, taking slices through what would be considered a vortex tube for inviscid flow along its length will yield different amounts of vorticity. Depending on the situation, two-dimensional slices of a flowfield are often being considered. In this case, a vortex could be considered any flow structure within the 2D plane that has an in-plane vorticity component.

Although many representations of vortical structures exist, there is one common way to measure their strengths. The strength of any vortical structure can be measured in terms of its *circulation*. Circulation is closely related to vorticity. By integrating the velocity tangential to a closed contour that surrounds the

origin of a vortex we obtain the circulation for the vortex. Equation 3 shows the general expression for the calculation of circulation by velocity integration.

$$\Gamma = \oint \vec{V} \bullet d\vec{l} \quad (3)$$

Circulation can also be defined as the vorticity perpendicular to a surface integrated over an area. Using Stoke's theorem Equation 3 can be written in an alternate form shown in Equations 4 and 5 below.

$$\Gamma = \int (\nabla \times \vec{V}) \bullet d\vec{A} \quad (4)$$

or

$$\Gamma = \int \vec{\omega} \bullet d\vec{A} \quad (5)$$

Real Versus Ideal Vortices

There are two commonly applied assumptions used when representing a vortical flowfield. The most simplistic representation of a flowfield relies on principles of ideal fluid mechanics. The ideal flow assumption neglects the effects of viscosity. In actuality, without viscosity there could exist no vorticity...but let us not regress. Ideal flow representations are still of great importance and can accurately represent certain flow conditions when assumptions are met. Very frequently, ideal representations of velocity fields are used to simplify calculations. Usually an ideal representation is sufficient enough to obtain the desired results. Occasionally a more accurate representation of the flowfield is required for mathematical or computational purposes. On these occasions, a real (viscous) approximation that includes the effects of viscosity is applied. One commonly applied representation for a real vortex is that of an Oseen vortex. An Oseen vortex attempts to better approximate a real vortex by simulating the reduction in tangential velocity that occurs within the core of real vortices due to viscous effects. For an ideal vortex the tangential velocity approaches infinity as the origin is approached because of the inviscid fluid assumption. For an Oseen vortex, the velocity initially increases as the distance from the origin decreases, but after a certain critical radius, begins to decrease and eventually becomes zero at the origin. As stated, this effect better represents the viscous core present within the center portion of a real vortex. The tangential velocity profile is plotted versus radial distance from the vortex center for both a real and an ideal vortex in Figure 2. Figure 3 shows the same vortex behavior but viewed within an intersecting plane. The velocity vector at the origin of the

ideal vortex is not plotted, as it approaches infinity. Real versus ideal vortex differences become important when developing an experimental method to measure their strengths.

Figure 2: Effect of viscosity on tangential velocity component near the vortex core: Ideal vortex (left) and Oseen vortex (right).

Figure 3: Velocity fields for Ideal and Oseen vortices.

Induced Velocity Relations

As shown above, the tangential velocity profile for a vortex varies as a function of the distance from the origin. Expressions have been developed to determine the velocity magnitude at an arbitrary distance from the vortex origin. These expressions are most important for the development of theoretical or

numerical schemes, some of which are described in Chapters 3.1.1 and 3.1.2. For ideal fluid mechanics, the Biot-Savart law for incompressible flow, shown in Equation 6, is one such relation. The velocity calculated according the Biot-Savart is the velocity perpendicular to a line extending from the vortex origin to the point in question. According to this law, the induced velocity caused by an infinitely long vortex filament is a function of the strength of the vortex, Γ , and the distance from the filament to the location in question.

$$\vec{V} = \frac{\Gamma}{4\pi} \int \frac{\hat{t} \times (\vec{r} - \vec{r}_1)}{|\vec{r} - \vec{r}_1|^3} ds \quad (6)$$

In the case of an Oseen vortex the tangential velocity can be expressed in polar form as shown in Equation 7. The parameter B is a constant used to define the strength of the viscous effect present.

$$U_\theta = \frac{\Gamma}{2\pi r} [1 - e^{-r^2/4B^2}] \quad (7)$$

Airfoil Representation

Understanding the general anticipated flowfield surrounding an airfoil is important when organizing formation flight scenarios. A lifting surface placed in a uniform flow produces an upwash ahead of the wing while leaving a wake behind. The wake is characterized by the downwash behind the wing, and by an accompanying upwash in the area on either side of the downwash region. Figure 4 shows regions of upwash and downwash for a single wing. Here it is possible to see the transition from a downwash region to an upwash region at the tip of the wing. In formation flight configurations it is this upwash region, outboard of the leading aircrafts' wingtip, that trailing aircraft must fly within to gain a lift advantage. The increase in lift occurs because the upwash field effectively rotates the lift vector forward, reducing the induced drag. Applying the Biot-Savart law everywhere in the surrounding domain, it is easy to see how the vortex lines cause upwash ahead of the wing, downwash behind the wing, and upwash behind and to the outboard region of the wing.

Figure 4: Upwash / Downwash regions for a lifting wing.

Many previous investigations have implemented theoretical techniques to approximate the benefits of formation flight. The utilization of fluid mechanics theory for the development of a formation flight model is not only a cost effective strategy, but comprehensive models can yield surprisingly accurate results even for complex and viscous flows. The key to developing an accurate theoretical model lies in the experimenter's ability to correctly integrate as many physical laws as possible into a manageable model. This can be, at times, a considerable task due to the complexity of the combined phenomena. Most formation flight studies are parametric studies however, and are concerned only with the variation in certain parameters for different formation positions. Often investigations try to model drag reductions, either for individual aircraft, or for the whole system of aircraft. Leaving out "higher order" effects for all cases is not substantially significant. Other investigations may concentrate on fuel efficiency, stability analysis, design of automatic formation flight controllers, or economic impact, and may not be concerned with the intricate details of the flow. They are not concerned with how the flow behaves very near each aircraft, but how the flow behaves in a larger, more global sense. Because of these reasons, a simplified model of an aircraft or wing can be used. Following are discussions of several common types of theoretical representations for wings.

The first type of representation concentrates on formation flight systems that use *horseshoe vortices* to represent the individual aircraft or wings. The author assumes that the reader possesses some previous

knowledge of the formation and dynamics of vortex lines, vortex sheets, and vortices. The horseshoe vortex wing model is the simplest of the three types of representations, and yields fairly accurate results. Actually, the upwash distribution calculated based on a horseshoe vortex representation is a very good approximation for the values produced by a plane vortex sheet with an elliptic spanwise distribution of circulation. When representing a lifting surface by a horseshoe vortex, a single U-shaped vortex line is used. Figure 5 shows how a horseshoe vortex is applied to represent an aircraft. The three-sided horseshoe vortex has constant circulation strength along each of its segments and the trailing segments of the vortex extend to infinity. By applying the Biot-Savart law, Equation 6, one can calculate the induced velocity field at any point in the surrounding domain. Knowing the induced velocity field will enable the calculation of the resultant effects seen by neighboring aircraft.

Figure 5: Lifting surface represented by a single horseshoe vortex of constant strength Γ .

Another way of representing individual aircraft or wings is by means of a *lifting line*. This method is credited to Prandtl in 1918, and is often called Prandtl's Lifting-Line Theory. In general, we can replace the wing with a sheet of vortex lines running in the spanwise direction as in Figure 6³⁰. Because the lift usually varies along the span, the total amount of circulation around a given cross-section of the wing also varies,

and goes to zero at the wingtips. This forces more and more vortex lines to bend and exit the trailing edge of the wing as the wingtip is approached. When we assume that the wing has a large aspect ratio, the bound vortex lines that make up the sheet may be approximated by a single bound line. It is this line that is called the lifting line and is seen in Figure 7³⁰. An assumption of lifting line theory is that the trailing wake is infinitely parallel to the direction of the free stream, and that the wake retains its original width. In actuality, there is a narrowing and of the wake and a drop in its elevation due to self-induced velocities. This assumption is usually acceptable for measurements and calculations performed in the near field, but for far field or Trefftz plane measurements, it can pose a problem.

Figure 6: Lifting surface represented by a bound vortex sheet.

Figure 7: Bound vortex sheet simplified as a single line for a large aspect ratio wing.

The third method used to represent a wing is using an explicit *lifting surface*. For this method, some type of vortex lattice is formed to represent the lifting surface. Often times these methods are simply called panel methods because the wing is represented by a surface, or arrangement of panels. In the past, the lattice

has been formed using an array of horseshoe vortices. This is a method similar to the horseshoe vortex representations except that multiple horseshoe vortices of varying strength are placed side by side along the wingspan. By distributing multiple discrete vortices, each can have a different strength, and a better representation of various wing geometries can be achieved. A more recent, and more computationally demanding approach, is called the Vortex Lattice Method (VLM) where discrete quadrilateral vortex filaments are used to represent the surface. In general, the surface of the wing is divided up into finite areas onto which each is placed one of these quadrilateral vortices. An example of this arrangement is seen in Figure 8. A boundary condition is imposed where the fluid velocity normal the surface is equal to zero. This normal velocity is a combination of a component of the free stream velocity and contributions from all the trailing vortices and bound quadrilateral vortices in the lattice.

Figure 8: Quadrilateral vortex elements for use with the VLM.

Doing this results in a set of simultaneous equations to be solved for the unknown quadrilateral strengths. After finding the vortex strengths, forces and moments can be calculated. The use of the vortex lattice method yields more realistic results than is possible with the single horseshoe vortex model.

1.3.1 Theoretical and Computational Research

One of the most fundamental types of research is that based on theory. Based off existing theoretical principles, and subject to various assumptions, new derivations applicable to different physical conditions can be established. Although the monetary cost for such research is generally less than for other types, and the solutions of an exact form, developing theoretical expressions for highly complex flow

geometries can be challenging without the verification of results from other (i.e. experimental) means. For this reason, most theoretical inquiries tend to either be very simplistic, or rely on strict limiting assumptions. To get around the issues of simplistic models and time consuming hand calculations, a computational approach is often applied. Applying a computational approach involves the implementation of a computer to iteratively perform calculations of a higher degree of complexity in a short period of time. Although most computational schemes do not yield an exact closed form solution, as do many theoretical works, the governing equations are identical. By dividing each lifting surface and the trailing wake up into smaller sections, panels, or elements, physical equilibrium equations can be applied to each section. The results for the wing as a whole are approximations. Accurate results are obtained by choosing the correct physical parameters, element sizes, initial conditions, and boundary conditions. Common topics of theoretical and computational formation flight research include total and/or total drag reduction or aircraft power reduction. Different wing and aircraft representations, including lifting lines, horseshoe vortices, unrolled vortex sheets, rolled vortex sheets are often used.

Schlichting and Truckenbrodt⁴⁶ performed calculations on symmetric v-shaped formations on the basis of lifting surface theory. Later, Lissaman and Schollenberger³⁴ investigated optimum formation shapes and D. Hummel extended the method of H. Schlichting to arbitrarily shaped formations of wings having different span, weight, and aspect ratio. Other investigations have been carried out by Hummel and Bock²⁶ that include aileron deflections to compensate for the rolling moment due to the asymmetric lift distribution instead of using a horseshoe vortex, which uses a mean value of upwash. For this study, the improved spanwise analysis led to slightly larger values of the calculated power reduction, especially for low spanwise distances. Because of this reason, they conclude that results obtained from a simple horseshoe vortex model can be regarded as the minimum values of power reduction. More recently, lifting surface theory was applied for unrolled vortex sheets behind the wings by Hummel²⁵, and for rolled-up sheets by Beukenberg and Hummel⁶.

One of the first analytical studies to examine the theory of formation flight was performed by Max M. Munk. Written in 1921, his report titled “The Minimum Induced Drag of Aerofoils” outlines many of the theoretical methods still applied today. Using a purely mathematical approach Munk introduces the lifting straight line and the Biot-Savart law and then uses them to derive statements about lift and drag for various configurations of multiple lifting surfaces. More specifically, he determines the necessary locations of multiple lifting surfaces so that a minimum induced drag for the entire system is achieved. For all of the configurations examined, a lifting line is always applied to represent the airfoils. A statement is made that this lifting line is a perfect approximation and suitably sufficient for the investigation. This methodology makes Munk’s theoretical work a valuable resource for establishing general parametric studies for formation

flight, but is perhaps not especially relevant for determining formation flight benefits with a high degree of accuracy. In different sections of his paper, he derives fundamental expressions for various airfoil positions. The first section determines the distribution of lift that produces the least drag for a single lifting line and the magnitude of the drag. In section two he analyses both a transverse straight line of multiple parallel airfoils and multiple parallel airfoils lying coincident in a single 2D plane (but not in a straight line). Section three introduces a three-dimensional parallel arrangement, and in section four, he analyses non-parallel two-dimensional in-plane arrangements. Several notable conclusions are drawn. The induced drag reductions are independent of the streamwise locations of the aircraft. It is also shown that the minimum induced drag for a planar wing of fixed total lift and span was achieved with a distribution of wake-induced downwash that was constant in the far wake. This condition required an elliptical lift distribution.

During the early 1950's increased interest in obtaining aerospace superiority resulted in numerous formation flight investigations. This decade was one of the first for formation flight research therefore, the primary objective of many of these investigations was to examine the results of placing one aircraft in close proximity to another, without concern for optimization. The ability to transport aircraft effectively was one such initiative. In late 1950 Franklin Diederich and Martin Zlotnick published a confidential research memorandum to determine the lift distributions for a Consolidated Vultee B-36 and two Boeing B-47 airplanes coupled at the wingtips¹⁷. Theoretical calculations were performed for the coupled flight configuration. Assumptions included incompressible, inviscid flow, thin airfoil sections, and no stall. According to the research proposal, viscosity would tend to decrease lift while compressibility would tend to increase lift, having a negating effect. For the theoretical models the lift was carried by an array of horseshoe vortices distributed spanwise and centered on the quarter-chord line. A total of 21 horseshoe vortices were used (11 for each side). With this configuration, ten equations for 11 unknowns were present. A corrector vortex was used to eliminate the last unknown. The wing of the B-47 was assumed to have zero twist and camber, whereas the wing of the B-36 had both twist and camber. The induced flow angle was equated to the angle of attack at the three-quarter chord line. The lift, induced drag, bending moment, and rolling moment, were presented for the aircraft flying alone and then in combination. The aerodynamic parameters for the combination of aircraft were expressed as increments to be added to the already known data for the uncoupled aircraft. It was suggested that the developed equations could be used for performance analysis, but no attempt to do this was made. For calculations, the chord discontinuity caused by different sized wingtips was small and was neglected during modeling of the system. Lift distributions were integrated to obtain total lifts, bending moments, and rolling moments. The induced drags were obtained by using Munk's stagger theorem. A total of four AOA conditions of individual planes were tested. The combination flight tests resulted in a maximum 10% lift increase for the B-36 and a 20% lift increase for B-

47. The research did not account for changes in the predicted formation results due to fuselage effects, tail effects, or aeroelastic effects.

During the next couple decades, the preliminary research of the 1950's was becoming more established and a greater interest in more efficient formations, in terms of fuel efficiency and transport methods, was beginning to appear. In 1970, using observations of birds that fly in formation to achieve performance benefits, P.B.S. Lissaman and C. A. Shollenberger, performed a range analysis of the flight of migratory birds. By applying simple theory to V-shaped formations consisting of a varying number of birds, substantial performance benefits were predicted. Preliminary results showed that elliptic loading was always very near optimal. As an example, an investigation studying a formation of 25 birds had approximately a 71% greater flight range than a lone bird. The calculations performed by Lissaman and Shollenberger assumed a straight plane trailing vortex wake. It was after these early (and often qualitative) studies, that future research in the area of formation flight became to mature into a more disciplined science.

Two computational studies were performed in the late 70's by Feifel¹⁹ and Maskew³⁷. For both of these studies, a vortex lattice method was used. Feifel's VLM was performed with calculations done in the near-field for five aircraft of known geometry, although only one formation test case was studied. The close formation flight led to an asymmetric load distribution due to the upwash. To counteract the rolling moment Feifel used aileron deflections. Similar to the work of Feifel, Maskew used a quadrilateral vortex lattice method, but with calculations performed in the Trefftz plane. Aircraft were of known geometry and included a trim procedure which held lift constant with zero roll. Used both echelon formation and two parallel rows of aircraft with similar drag reductions for the systems as a whole. A reasonable tradeoff between a practical formation size and range benefit seemed to lie between three to five aircraft. Range increases for the entire formation were about 46% to 67%.

Using the fact that birds fly in formation to gain a performance benefit, D. Hummel²⁵ also published a report in 1983 in the *Journal of Theoretical Biology* discussing the theoretical aerodynamics behind formation flight power reductions. Hummel uses two different lifting element representations. Both single horseshoe vortices and lifting surface theory are used to represent the wings for both homogeneous and inhomogeneous formations (wings of different span, AR, and weight). The horseshoe vortex method attempted to account for the narrowing of the trailing wake by using an approximate width for the horseshoe vortex, as opposed to a vortex of equal width to the wing. For the lifting surface theory approach, the leading wing was replaced by a 2D distribution of bound vortices (similar to a simple vortex lattice approach) for the wings' surface and an unrolled, plane free vortex sheet for the wake. An important aspect of his lifting surface approach was that the spanwise and the chordwise distributions of the upwash

were accounted for at each lifting surface. This provided for an asymmetric load distribution that was accounted for with flap deflection. A comparison of the horseshoe vortex method and the lifting surface method showed no difference in flight power reduction for large spanwise separations. For small spanwise separations, the power reductions calculated with the lifting surface theory were greater than with the horseshoe vortex method. This conclusion establishes the horseshoe vortex method as a conservative estimate. In addition, the lifting surface theory tended to overestimate the power reductions because it did not account for the narrowing of the trailing wake. As a final conclusion, Hummel determines that the horseshoe vortex method was not only easier to apply, but most likely produced more accurate results.

In 1990 Beukenberg and Hummel⁶ extend Hummel's previous theoretical work of 1983 to include aircraft flight tests and introduce the concept of an automatic aircraft position control system. Again, the investigation is for arbitrarily shaped flight formations of different wings. The previous theoretical calculations utilizing horseshoe vortices and unrolled vortex sheets were revisited with the addition of a model that used a rolled-up vortex sheet behind each wing. The rolled up sheet caused a narrowing of the wakes' width, more accurately representing the actual expected flowfield. Assumptions were that all aircraft fly in a single horizontal plane and that the formation shape remains constant with respect to time. It was found that the total power reduction for the entire formation is independent of the shape of the formation (a conclusion consistent with Munk's well known displacement theorem). Using the horseshoe vortex approach, Beukenberg and Hummel determine that total power reductions as high as 10% could be realized for a formation of just two wings. As the number of wings is increased the total power reduction increases and eventually asymptotically reaches a theoretical maximum. The *distribution* of the power reduction, however, depended heavily on the shape of the formation. Both the unrolled and rolled-up vortex sheet methods resulted in greater power reductions for the formation. Using the rolled-up sheet method, they determine that a maximum power reduction occurred when the trailing wing was at a lower elevation than the leading one. The ratio of height difference to spanwise gap, $\Delta\zeta = \Delta z/s$, for maximum power reduction was 0.155. Also performed were two flight tests to demonstrate power savings under realistic conditions.

In a 1998 study, Blake and Multhopp¹⁰ consider a new problem not considered previously. To better represent actual flight situations, their formation model allowed for the change in aircraft weight due to fuel consumption, and for a nonuniform distribution of weight across the span. Theoretical calculations utilizing horseshoe vortices are used. The accuracy in maintaining the lateral position of each aircraft in addition to the effects of rotation of lead aircraft (used to increase range) was studied. They concluded that large reductions in induced drag are possible and that an elliptical lift distribution for the entire formation maximizes drag reduction. This was easily achievable using their formation model because each aircraft did

not need to be identical. A 60% range increase was possible for a formation of five aircraft. To achieve substantial drag reductions, the positions of the aircraft must be maintained to within 10% of a wings' span. It was hypothesized that aircraft trimming to account for wake-induced forces and moments may reduce the drag benefit. An interesting and rarely investigated aspect related to flight formations was introduced. Because placing an aircraft's wingtip within the trailing vortex of a leading aircraft tends to disrupt the structure of the vortex, additional trailing aircraft will not "feel" the same effect from that vortex as they normally would if it had been disturbed.

For a single wing, Iglesias and Mason²⁸ claim that the generally accepted minimum induced drag spanload is not necessarily the best choice for an aircraft. Purportedly, weight savings resulting in performance benefits can be achieved using load distributions other than the elliptic distribution. The use of this idea was extended to groups of aircraft flying in formation. A constrained minimization approach is used to determine the optimum downwash distributions for wings in formation flight. The optimum spanloads for a group of aircraft flying in an arrow formation were found using a discrete vortex method and a Trefftz plane analysis under constraints of lift, pitching moment, and rolling moment. Airfoil twist and camber were solved for after the required load distributions were found. They found that the ideal wing planform shapes for load distributions necessary to provide the maximum drag reduction for the formation did not correspond to ideal shapes for individual aircraft flying alone.

In 2001 Chichka, Speyer, Fanti, Park¹⁶, interested in the development of a peak-seeking control system for a pair of aircraft in close formation, applied many of the previous theoretical principles to develop their model. They use two bound horseshoe vortices to represent the aircraft. According to the Helmholtz vortex laws, the bound vortex cannot end at the wingtip and vortex "tails" must be present, which are not necessarily fixed in space. For simplification, the "tails" were fixed in space, and lost none of their strength. The Kutta-Joukowski theorem was used to calculate the strength of each of the bound vortices. Their model also assumed inviscid, incompressible, constant, straight flow. Using the Biot-Savart law they calculated the induced velocities, then calculated the local angle of attack due to the upwash. In actuality, the trailing vortex tends to descend slightly, but due to the follower's upwash, the vortex is actually approximately in the same horizontal plane as the follower.

Using a VLM to perform the analytical calculations, Wagner, Jacques, Blake, and Pachter⁵¹. Optimal positions for the wingman of a tight formation flight of T-38 Talon aircraft were determined using the induced drag as the main parameter to be reduced. A second objective involved the addition of a second wingman (two following aircraft). Both 2-D and 3-D wing models were tested, but in the end 2-D models were used to make calculations easier.

Similar to the work of Iglesias and Mason²⁸, Frazier and Gopalarathnam²⁰ proceed to determine the optimum downwash distributions for wings in formation flight. Normally, the wing geometry described by an optimum downwash distribution is less than practical therefore adjustments to the configuration must be made before application to actual flight test conditions. The reasoning behind their desire to determine the optimum downwash condition spawns from the prospective future ability of lifting surfaces that are adaptive. Frazier and Gopalarathnam propose an alternate approach that first determines the optimum downwash in the Trefftz plane. The advantage of their approach is that it results in a simple analytical result. They use a modified version of the analytical calculus-of-variation approach of Jones²⁹ that allowed for lift and rolling moment constraints of each individual wing. Whereas Jones's method is for a single wing, Frazier and Gopalarathnam modify the method for application for a formation of multiple wings. After determining the optimum downwash in the Trefftz plane, a vortex lattice formulation is used to determine the optimum lift distribution from the downwash. This VLM approach is similar to that of Blackwell⁷. Additionally, the induced drag for each aircraft is determined using the near field approach of Iglesias and Mason. The wake was assumed to be rigid and invariant with the streamwise direction. Results for several formation positions of seven equally spaced wings showed that a maximum induced drag reduction of 71% could be achieved with a lateral centerline-to-centerline spacing of $0.89b$, where b was the wingspan. Larger lateral spacings showed an increase in the total induced drag. They show that the lift distributions for wings in formation flight with constraints on the lift and rolling moment on each wing must result in a linear variation in the Trefftz plane downwash for the total induced drag of the formation to be a minimum. They compare their research to results obtained using the constrained-minimization approach of Iglesias and Mason and find the results to be identical. A constraint of their method, similar to the constrained-minimization approach, is that it is only valid if the wake traces of each individual aircraft do not intersect or overlap. They conclude that there isn't any additional benefit to their optimum downwash approach over the constrained-minimization approach, but it does provide a simple closed-form expression for the optimum downwash.

To summarize, most of these theoretical and computational investigations ignored the affects of compressibility, viscosity, and flow separation. The theoretical investigations used either horseshoe vortices (regular width, or narrowed) or a simple single lattice of horseshoe vortices. The computational methods used several variations of a vortex lattice method with calculations being performed in either the near field or the Trefftz plane. Additionally, these methods often assume a rigid, invariant wake in the streamwise direction. It could be stated that most of the theoretical or computational techniques available provide results representing formation flight based on ideal flow principles. Incorporating viscous and compressible effects into theoretical or computational studies while producing accurate and realistic results is a difficult

proposition to satisfy. While the numerous assumptions make computational formation flight research practical, and theoretical research possible, there is no exact substitute for actual flight tests or other experimental work. During actual real-life tests all physical laws are automatically satisfied and present, yielding a naturally accurate flowfield. This is not to say experimental research is the best method for all situations. The same reasons that make experimental research attractive can complicate testing methods creating additional error. The only way to validate computational results is through direct experimental measurement.

1.3.2 Experimental Research

Formation flight has only very recently been studied experimentally in detail^{22, 35, 50}. The most valuable experimental tests performed were those that used actual aircraft for tests. Although only a very limited number of configurations have been tested this way, the results are indispensable. Experimental flight tests, performed by Beukenberg and Hummel⁶, used two Dornier Do-28 aircraft. The results were compared to theory to see if there was correlation. A third aircraft, a Dornier 27, was used to fly above the formation pair to monitor the formation positions. During the tests the front aircraft flew at a constant flight level and a constant velocity to provide a wake for the trailing aircraft. The two aircraft were first flown with a large spanwise separation to establish a baseline for interference free flight. Then, the outer aircraft moved to a smaller wingtip spacing where there was greater interference from the leading aircraft's wake. To establish the position of maximum interference, the trailing aircraft moved to a position where throttle position became a minimum and aileron deflections became a maximum. Flight tests were carried out for two different speeds ($V=80$ kts. And 100 kts.) in order to vary the lift coefficient. They concluded that maximum flight power reductions of about 15% could be realized for practical flight. For low lift coefficients and small longitudinal separation distances, the results from the experimental flight tests correlated very closely to the theoretical calculations. For higher lift coefficients and larger longitudinal distances, there were larger differences between the experiments and theory. At NASA-Dryden Iannotta²⁷ also performed tests on actual aircraft to explore the feasibility of achieving formation flight benefits. 45 flights by specially equipped F/A-18's were made. Pilots explored the best places to fly within the vortex and found the safest, most efficient distances for following. Drag was reduced by up to 20% and used 18% less fuel.

The acquisition of load, moment, and pressure data within a wind tunnel test environment are the most common types of experimental tests. Using a loadcell balance is the most direct way to obtain total lift and drag values for models in formation flight although this method provides no information about the nature of the flow surrounding the models. Magill et al.³⁵ carried out wind tunnel tests using two models;

they modeled the mothership with just a swept wing and the hitchhiker with a 1/32 scale F-84E model. The forces and moments on the hitchhiker were measured directly with a loadcell balance system. A closed circuit wind tunnel with a 6 x 6 x 24 foot test section at Virginia Tech was used for experimental tests. Computational aerodynamic analysis using a vortex lattice method (VLM) and a desktop aircraft model were also used to determine the best location for a hitchhiker aircraft and analyze the stability of the formation. Three flight configurations were highlighted: wingtip-docked, close formation, and towed formation. The data showed very little change in the aerodynamic forces of the mothership, and possibilities of large benefits in lift and drag for the hitchhiker when located slightly aft and inboard with respect to the mothership. The wingtip-docked configuration had a 20-40 percent performance benefit for the hitchhiker compared to solo flight. The close formation configuration had performance benefits for the hitchhiker approximately 10 times that of solo flight, and the towed formation was approximately 8 times better than solo flight. The vortex lattice results showed the same trends as seen in the wind tunnel data, but did not predict benefits as large as seen in the wind tunnel. They showed that the vortex lattice method underestimated the induced drag benefit for the trailing wing because it used assumptions of a flat wake and inviscid flow. They concluded that the vortex lattice method was useful for performing a quick analysis of position and planform effects in formation flight. Gingras²¹ also performed wind tunnel tests on a 1/10th scale model of a modern military fighter aircraft. Force, moment, and surface pressure measurements were acquired. The wind tunnel testing was performed to investigate the effect of a lead aircraft on a trailing aircraft flying in close formation. Later Gingras et al.²² acquired both static and dynamic force and moment data as well as surface pressures and wake data. Two goals were to find positions which maximized L/D increases while minimizing force and moments that make precise station keeping difficult, and to determine the control power required to enter, exit, and remain in beneficial formation positions. They showed that the aforementioned test methods can adequately predict benefits, and be used for control law design for automated systems. At the University of Stuttgart, Institute of Aerodynamics and Gasdynamics, laminar wind tunnel facility in Stuttgart, Germany, a test to experimentally determine the sectional lift coefficient for a two-dimensional wing as performed³². The wind tunnel was a 46m open circuit wind tunnel. The lift was determined by measuring and integrating the pressure distribution present across the top and bottom walls of the wind tunnel, instead of using the pressure distribution over the models' surface. The difference of the pressures determined on both walls is proportional to the lift of the wind tunnel model. This technique only applies for two-dimensional wings that span the width of the tunnel. Although this particular test was not directly relevant to formation flight research, it provided an alternate way to measure the lift without the using a loadcell balance or wing surface pressure measurements and was a valuable reference for the development of the experimental methods contained in this work. The results agreed very well with other measuring methods. The drag was determined by integrating the total pressure loss in the wake profile

obtained with a wake rake. During testing, the rake was traversed at a constant speed in the spanwise direction and the mean airfoil drag was determined. The wake rake was mounted at a distance of 30% chord behind the wing.

Ever since the use of automated five and seven-hole velocity probes has become more common, wake surveys have been increasingly performed to provide additional information about the drag due to lift. Maskew³⁷, Wu et al⁵², and Brune¹² tried to separate the contributions of viscous and vortex drag using wake surveys. Although some success has been achieved, Spalart⁴⁹ mentions there is some tradeoff between apparent induced drag and the inferred viscous drag that varies with the streamwise location of the survey. The problem is due to the difficulty of finding a suitable definition of induced drag in real, viscous flows, particularly those with separation. For simple attached flows, experimental techniques can provide some insight into and rather accurate estimates of the vortex drag⁴⁸.

Vlachos and Telionis studied the formation flight problem with a slightly different method. They used exclusively a digital particle image velocimetry system (DPIV) to obtain time-resolved velocity distributions in a Trefftz plane for close formation flight and docked flight. Detailed velocity measurements in the wake of wing configurations were made using two NACA 63-420 wing models by Vlachos and Telionis⁵⁰. For these tests each wing model was mounted to an endplate, therefore only one wingtip of each wing was free. It is essential that both the tips of the trailing model are free. It was reported that for very small gaps between the wing tips, the total shed vorticity was reduced. In these cases, severe unsteadiness also developed. In the interest of docked flight, this would make the docking operation more difficult. It was also found that a discrete frequency appears, which was measured to be equal to $fc/U \sim 2$, with f the frequency, c the chord and U the free-stream velocity.

We want to develop a method that is capable of both, measuring loads for wings in formation flight, in addition to providing information about the flowfield surrounding the lifting surfaces without influencing the flow or requiring the installation of surface pressure taps.

1.4 Objectives

The main objective of this study was to develop a method that used experimentally obtained near-field velocity measurements to determine the total lift and the lift distributions for the trailing wing of a two-wing formation. Additional applications of the method would allow for the determination of lift and drag benefits for various wing formations. After understanding the mechanics of flow around a single three-

dimensional wing, close proximity configurations of multiple wings can be examined. For the current research a pair of wings are used. A swept NACA 0012 wing represents the leading aircraft. A cambered three-dimensional NACA 63-420 wing is used to represent the trailing aircraft. The process of designing a suitable experimental setup and test method is described. A final configuration and method is chosen that provides a novel method to determine in-flight loads for wings flying in close proximity. By using this method, which only relies on data taken within the downstream flow, the models are free from interference flow effects due to bulky load measurement devices and mounts.

Two main approaches for manipulating the near-field plane data obtained from the wind tunnel tests are used, and the results from each method are compared. The first method performs a mathematical integration of the velocity data around a closed circuit, yielding the circulation for each trailing vortex in the plane and the total circulation (or lift) for each wing. The second method first calculates a two-dimensional vorticity field from the velocity field then integrates the vorticity over an area to obtain the circulation. Several vorticity calculation schemes are compared before choosing a final method. To validate the accuracy of these two methods, comparisons of the lift benefits for the trailing airfoil are made to direct measurements taken using a NASA six-degree-of-freedom GA10 strain gage balance and to results published in existing literature.

To obtain an appreciation of the complexity of the ensuing flow for compound aircraft flight, a seven-hole probe was used to acquire velocity data in locations other than the near-field plane. The distribution of circulation for the trailing airfoil was determined by taking velocity measurements at various cross-sectional positions of the wing. Details related to the design of these tests and their assumptions are described.

In order to provide the greatest benefit for certain aircraft, or for the formation as a whole, their optimal formation position must be found. Much research has already been performed for various aircraft models to determine the optimal formation positions. This is a typical objective of many formation flight studies that usually entails trying a large number of positions and then measuring the resulting benefit for each (usually with a loadcell instrumented balance system to measure the loads directly). Because of the extensive amount of existing research in this area, it is not one of the primary topics of investigation.

Chapter 2

2 EXPERIMENTAL SETUP AND PROCEDURE

Introduction

The purpose of this study was to develop an alternative method to experimentally determine the distribution of circulation in the wake, and across the surface, of wings in close proximity flight. The downstream velocity distribution was used to obtain estimates of the circulation present for various single and two-wing configurations. Knowing the circulation distribution enabled the determination of lift forces experienced by a given wing. It perhaps becomes apparent what some of the main components of the setup should include. A wind tunnel, two wing models, and a means to measure the downstream velocities were needed. These are, of course, only the most basic necessities for the research. The need for other specific equipment and tests was continuously reevaluated throughout the duration of the study. The two wing models were mounted into the wind tunnel test section. One wing, representing the leading aircraft, was mounted directly to the side wall of the wind tunnel test section. The second wing, representing the trailing aircraft, was mounted on a sting located aft, and either inboard, inline with, or outboard of the leading wing. The design of the mounting sting, in addition to its location, for the trailing wing was modified part way through the tests and will be discussed later. To obtain velocity data (or more accurately, pressure data) in the wake of the wings, a small diameter seven-holed probe mounted on a computer-controlled, motor-driven three-dimensional traversal system was used. There were actually more components to the entire system than just the aforementioned, and their purpose and implementation are discussed in greater detail shortly.

Before proceeding directly into a discussion of the key components of the experimental setup, a look at several important issues relevant to experimental wind tunnel testing are briefly discussed. Any experimental study includes several important parts: Project evaluation, facility and equipment selection, model fabrication, setup integrity, repeatability/data quality control and troubleshooting are some of the most notable. The first task is to determine whether or not there is the opportunity or the need for a project. Is there a need or desire for research in this area? The idea of formation flight has been a topic of interest for many decades, and much research has been performed in this area. Although there are more straightforward or traditional methods for the determination of the circulation (lift) distributions of wing models, there was an interest to devise an alternative method equally capable of acquiring accurate results without the need for instrumentation placed directly on the models. After determining that there was a

need, a project plan was made. The first step was to define the project as well as possible. This included defining project goals, the methods required to accomplish them, and potential obstacles. Until there was a good grasp on the necessary details, it was not possible to choose specific equipment and procedures to compliment the research. During any experimental inquiry, especially one delving into previously uncharted areas, there exists a great possibility for unforeseen setbacks. Usually, the more well thought out and structured the project plan is, there exists less probability for inefficient use of time. Sufficient time was budgeted into the project timeline to account for any obstacles.

For this particular research several issues of concern included setup integrity, test repeatability and data quality control. Although there was always a level of uncertainty present for the experimental tests, they were designed to avoid as much experimental error as possible. Solid model mounting devices were used to minimize unwanted vibration that could affect the surrounding flow. The ability to quickly and accurately reconfigure the test apparatus was also of importance to minimize downtime and assist with the repeatability of results. Choosing suitable data acquisition parameters and post-processing techniques for the experiment at hand also help minimize unnecessary error. A thorough analysis of possible error sources, in addition to the accurate determination of test parameters (i.e. temperature, humidity) is essential. General methods to help minimize error include the usage of properly calibrated sensors, the proper implementation of instrumentation.

2.1 Wing Models

One necessary requirement for the wings was that they needed to be three-dimensional models to enable the measurement of lift distributions due to formation and end effects. Vorticity must be allowed to shed from both wingtips of the trailing wing with minimal interference from the supporting sting to obtain an accurate distribution of lift. The design and implementation of three-dimensional models without significant interference from stings or other supporting braces has always been one of the greatest challenges of experimental wind tunnel testing. The research presented here is of no exception, and perhaps presents an even greater challenge. Because the primary data for the research was taken with a flow measurement probe at a location downstream from the formation, any disturbance caused by the model mounting devices would also be measured, thereby introducing error. By utilizing minimization and streamlining techniques, stings were designed that would reduce the interference within the flow. A more detailed discussion of the sting designs and their implementation within the wind tunnel are included in the wind tunnel section.

A total of two wing models were fabricated and tested throughout the duration of the research. Depending on each particular test, either one or both models were used at a single time. There were two reasons to use only two wings: A pair of wings would provide a simpler flowfield than three or more wings

in addition to fitting within the confines of the ESM wind tunnel. Criteria for choosing the airfoil sections for the wings were based on two main factors. All of the models chosen for tests needed to be easy to obtain or inexpensive to fabricate and preferably have a large amount of existing literature published about them. A significant amount of existing literature enabled easier verification of results.

The leading wing was an aluminum NACA-0012 wing with a 15° sweep angle, mounted from the side wall of the wind tunnel. This wing had zero camber and a thickness of 12% with respect to the chord. The span was 8" and the chord, C , was 4" making the aspect ratio, $AR = 2$. The position of this wing was permanently fixed although it could be rotated to adjust the angle of attack.

The second wing chosen was a cambered rectangular wing with a NACA 63-420 cross-section. This wing model was fabricated by laminating a machined core of balsa wood with a carbon fiber shell. A steel rod with a length equal to that of the wings span was drilled and threaded at its midspan and at its ends. The rod was inserted into the wing, coincident with the quarter-chord line to allow for several mounting locations. Because of the extreme forward location of the maximum camber for these airfoils, a five-digit classification system was adopted versus the four-digit classification¹. This is a 6-series designation airfoil section with a thickness ratio of 20% and a minimum surface pressure location at 0.3 chords behind the leading edge. The AR was 4 and the chord, c , was 2". This airfoil section was chosen because past experiments have shown that extensive low Reynolds number laminar boundary layers (similar to those experienced in the current wind tunnel tests) could be sustained for wings with smooth surfaces¹.

2.2 Wind Tunnel and Model Implementation

There are several types of wind tunnels in which experimental research can be performed. The two basic types are open and closed circuit. The open circuit tunnel is less costly to construct, although it has a higher operating cost compared to a closed circuit tunnel. Open circuit tunnels require more screening at the inlet to obtain high quality flow (low turbulence) within the test section than do closed circuit tunnels. Regardless of what type of tunnel is chosen, the test section size and turbulence levels must be within the desired ranges. For the current research, the Engineering Science and Mechanics departments' low-speed wind tunnel facility at Virginia Polytechnic Institute and State University was utilized. The wind tunnel is an open-circuit low-speed tunnel, constructed in 1983. Figure 9 shows a schematic of the wind tunnel. Driven by a single 15hp axial fan with stators, the maximum sustainable wind tunnel speed is approximately 30m/s, more than adequate for the tests performed. Actual tests were performed at 20m/s. Although a larger wind tunnel would have also sufficed, the size of the ESM low-speed facility was an ideal low-cost solution. By using a smaller wind tunnel and smaller models, costs could be minimized while still providing an adequate testing facility. The test section dimensions were 0.51m by 0.51m and 1.27m long. For easy access to

models mounted in the tunnel, and for setup visualization during testing, a removable Lexan sheet was used as one of the side walls of the test section. Both the fan and motor were mounted on a concrete isolation block to reduce the transfer of vibration to the tunnel walls. To control turbulence in the flow, a honeycomb lattice and five nylon flow conditioning screens were used in the settling chamber. The generated velocities were very stable over time having a maximum turbulence intensity of 1.25%. The flow across the test section is reasonably uniform with a variation smaller than 2.5%. The wind tunnel diffuser was angled at 5° and carried an area ratio of 3.7. More details regarding the design, construction and calibration of this facility can be found in the report by Seider⁴⁷.

Figure 9: ESM Low-Speed Wind Tunnel Schematic.

To monitor the wind tunnel velocity, a pitot-static tube was mounted on the far wall. The pitot tube was mounted far enough from the side wall of the tunnel to provide accurate air velocity measurements without interfering with the airflow impinging on the test models. The pitot-static tube was connected to an Edwards-Datametrics Barocell precision pressure transducer (Model 590D-100T-3Q8-H5X-4D) to measure the free stream dynamic pressure reading of the test section. The Barocell has a range of 100 Torr and a resolution of 0.01% of full scale. The specified accuracy of the Barocell is 0.05% of the reading plus 0.001% of the full scale. To display the pressure reading measured by the Barocell pressure transducer a BOC Edwards LED readout display was used.

Mounting the airfoil models into the wind tunnel required several minor modifications of the wind tunnel. The larger swept model was permanently mounted to a circular template that was then fitted into a circular cutout in the side wall of the tunnel. By rotating the circular template, the angle of attack for the leading airfoil model could be adjusted. It should be noted that the axis of rotation for adjusting the AOA

was not coincident with the axis of the swept airfoil, but with an axis perpendicular to the side wall of the tunnel and to the oncoming flow. This geometry is illustrated in Figure 10. At a zero degree AOA the airfoils leading edge is horizontal. When the airfoil is adjusted to greater angles of attack the leading edge is no longer horizontal; the tip of the airfoil becomes lower than the base. Rotation about axis “A-A” (as opposed to axis “B-B”) more accurately represents the physics encountered when an actual aircraft increases its angle of attack, and is referred to as the pitch angle.

Figure 10: Pitching axis for the leading wing (view from above).

The global and local coordinate axes, the spacing of the two wings, and the location of the near-field data plane are all shown in Figure 11. The parameters $\eta=Y/c$ and $\xi=X/c$ are nondimensional variables measured using the global coordinate system $\{X,Y,Z\}$. Also referenced to the global coordinate system origin is the data plane. The location of the data plane was kept a constant $3c$ from the trailing edge of the leading wing. The local coordinate system is used when describing positions in reference to the trailing wing. The points “a” and “b” are located at the trailing edge wingtips of the trailing wing and are projected straight downstream to the data plane for comparative purposes between the different test cases.

Figure 11: Relative positions and coordinate axes for the leading wing, trailing wing, and data plane.

For a majority of the tests, which utilized the NACA 63-420 wing as the trailing airfoil, the supporting sting was attached to the model at the midspan of the quarter-chord line and extended downward towards the floor of the wind tunnel at a 45° angle. Because the loadcell balance system (to which the sting was attached) was mounted external to the wind tunnel test section, modifying the floor of the tunnel test section was necessary to allow for the penetration of the trailing airfoils' mounting sting. This was accomplished by simply cutting out a portion of the test section floor to accommodate all the desired positions of the trailing airfoil relative to the leading airfoil. After positioning the sting, the remainder of the opening was covered with a thin steel plate and fastened to the floor. The design of the sting was governed by two main parameters. Because the sting was fairly long (0.45m) it had to be rigid enough to prevent itself from flexing significantly under the wind load, otherwise undesirable vibration of the model would occur. Additionally, if too bulky a sting were used, it would interfere with the flowfield introducing error in velocity measurements. These are two common obstacles of sting design for experimental wind tunnel testing. By streamlining the shape of the sting a balance between size and strength was achieved. Figure 12 shows a cross-section of the sting and its implementation into the wind tunnel for tests using the NACA 63-420 wing.

Figure 12: Sting cross-section and wind tunnel implementation.

2.3 Loadcell Balance

Although primary results were obtained using a flow measurement probe technique, a means to verify these results was also desirable. For this task, a strain gauge balance mounted to the base of the trailing wings' sting provided a direct measurement of the forces and moments experienced by the wing. The balance used was a NASA Langley GA10 with six degrees of freedom. Two sets of four active strain gages were wired into a full Wheatstone bridge circuits. All six degrees of freedom were measured. Before using the balance for tests, a calibration was performed. To obtain lift and drag from the six measured forces and moments, general statics principles were applied. Figure 13 shows the six forces and moments measured. Vishay Measurements Group strain gauge conditioners (Model 2120A) were used to amplify the signals. These particular conditioners accept connections for quarter, half, or full Wheatstone bridges and internal half-bridge, dummy 350Ω and dummy 120Ω dummy completion gages. Output gain is continuously adjustable between 1 and 2100. It should be noted that because four gauges were available for use in each bridge, there was no need for the integrated bridge completion circuits of the signal conditioners. The Wheatstone bridge excitation voltage was set to a constant 10 volts and the amplifier gain was set to X200. The common mode rejection ratio for the X200 gain multiplier was specified as 100dB in the literature.

Selected Wing Loads and Resultant Forces

Figure 13: Forces and moments on trailing wing with the resulting forces and moments on the load cell.

2.4 Flow Measurement Probe

A convenient way to obtain velocity measurements within a flow field is through the use of a flow measurement probe. A 7-hole probe, produced by the Aeroprobe Corporation was used to take velocity measurements in the wake of the wings. Multi-hole probes can measure all three components of the velocity, the total pressure, and the static pressure at any given point. The probe is constructed from several different machined parts and then assembled. Below, Figure 14 shows the external geometry and components of the probe. The probe tip is made from brass, and has seven holes drilled into it. The probe used for this research had a tip with a conical shape and a diameter of 1/8". Care was taken to choose a probe with a suitable diameter for the flow field structures being studied. If the probe diameter were too large, it would have interfered with the vortical flow structures being measured. The first extension is the same diameter as the probe tip and its length was approximately 20 probe tip diameters long. To increase the overall length of the entire probe, a ferrule and second extension are added. The diameter of the second extension was about 1.5 times that of the tip. The larger diameter of the second extension gives the probe structural rigidity. The overall length of the 7-hole probe used was 6". At the end of the probe, there is a brass mount. The shape of the mount is hexagonal, allowing it to be mounted into a collar with the same shape cutout, and then secured with a setscrew. Each of the holes in the probe tip leads to a stainless steel tube, with its inside diameter matching the diameter of the holes. As the probe shaft diameter increases, each

tube is telescoped into a larger tube, which finally protrudes from the back of the mount enabling the connection of flexible Tygon tubing. Each connection is soldered and tested for strength and leakage. The final assembly is also tested for pressure “cross-talk”, i.e. pneumatic communication between two or more probe holes and their associated tubing. Figure 15 presents structural details of a typical probe tip and sample dimensions for a 0.125” (1/8”) tip diameter probe tip. Because no two probes are identical after fabrication, and because small amounts of damage can occur to the probe over time, it must be cleaned and calibrated often to provide accurate results. Before tests were run, the probe was cleaned by drawing isopropyl alcohol through the ports and then dried. The calibration procedure performed was previously developed by the Aeroprobe Corporation and involves placing the probe into a known flowfield and rotating the probe through many different angles while acquiring data. The principle idea of operation is the following: The pressure over a bluff body is the highest at the stagnation point and lowest near separation. If the flow direction forms small angles with the axis of the probe (below 2°), the center hole registers the highest pressure. If however, the flow is steeply inclined with respect to the probe then one of the peripheral holes on the windward side of the probe tip registers the highest pressure, while on the leeward side of the probe, the flow is separated. The pressure information provided by the three holes in the separated region is not used. This type of careful calibration allowed the instrument to measure the direction and the magnitude of the velocity accurately.

Figure 14: External geometry of a 7-hole probe.

Figure 15: Probe tip dimensions and tubing connections for a 7-hole probe.

2.5 Data Acquisition System and Post Processing

To obtain actual velocities from the probe, several pieces of equipment must be used in addition to the probe itself. Tygon tubing was attached to the leads of the probe. It was important to label the ends of the tubing to determine which tube went to which port of the probe. The tubing was then connected to a Pressure Systems ESP (model ESP-32HD). The ESP is a compact device containing multiple pressure transducers; each transducer converts an applied pressure to a voltage. This particular model ESP contained 32 separate ports with a usable pressure range of $\pm 10''$ of water. The first seven ports were connected to the seven Tygon tube leads from the probe. The eighth port on the ESP was connected to the total pressure side of the wind tunnel pitot tube. This enabled the measurement and recording of the wind tunnel velocity at the same instant data were recorded from the probe, important for the nondimensionalization of velocities to compare test cases later. To interface the ESP to the data acquisition board in the computer an Aeroprobe Corporation system was used. The system includes hardware that interfaces with the ESP, and motor controllers to drive the stepper motors used for the 3-axis probe traversing scales. Figure 16 shows the three-axis motorized traversing scales with the probe mounted to the z-axis. The scales allowed the probe to be moved to the desired locations in the downstream flowfield.

Figure 16: Motorized 3D traversal system and probe.

A Measurement Instruments CIO-DAS08/AOL data acquisition board was installed into the computer system used for data collection. This board has eight 12-bit differential ended analog inputs. Using differential inputs was ideal because it helps eliminate electromagnetic interference (EMI) or radio frequency interference (RFI) that may be present in the path of the signal wires. EMI and RFI can induce voltages on both signal wires and the effect on single ended inputs is generally a voltage fluctuation between signal high and signal ground. A differential input is not affected in this way. When the signal high and signal low of a differential input have EMI or RFI voltage induced on them, that common mode voltage is rejected. Because the Aeroprobe system was designed to be used with this particular board, there was no rewiring necessary when taking data from the various components of the test setup (i.e. strain gauge balance, ESP pressure transducer) or when driving the traversing scale motors. Before installing the board into the computer, InstaCal (initialization software provided for the board) was installed. The D/A range for the board was set to ± 5 volts. In Figure 17 the block diagram outlines the data acquisition system. During the data acquisition process, the pressures acquired with the probe and ESP were reduced to velocities using Aeroprobe's built in velocity reduction software. For further information related to the reduction scheme consult the Aeroprobe multi-hole probe manual².

Figure 17: Data acquisition system layout.

Once the acquisition system was designed and assembled, obtaining useful data depended on its correct implementation. Here we consider specific parameters such as sampling rate, period, and spatial resolution. Several types of tests were conducted, each requiring slightly different acquisition of data. Regardless of which test was performed, the time requirement to acquire all the data was large. For each point in the acquisition grid, time was needed to traverse the probe, acquire data, and then store the data. Because of the dynamic nature of the flow conditions and the need to traverse the probe, the data were time averaged. Data were acquired for a long enough time period and at a sufficient sampling rate to obtain a characteristic sample. Factors that contributed to the dynamic nature of the flow included Strouhal shedding from the wing models, trailing vortex position instability, and tunnel turbulence. To determine a sufficient period and rate, an estimate of the various frequencies of the dynamic flow factors needed to be made. Wind tunnel tests were performed at 20m/s. Using Equation 8 to determine the Strouhal shedding frequency, an estimate of 630 Hz was made for both the leading and trailing wings. In Equation 8, St is the Strouhal number, U is the free stream velocity, and d is the characteristic diameter. Estimates were calculated using $St=0.2$, and $d=0.25''$ (an approximate thickness for either wing model).

$$f = \frac{St * U}{d} \quad (8)$$

The frequency of the trailing vortex position oscillations (which only occurred for certain formation positions) could be determined from the time record of pressures acquired with the velocity probe. For all cases the frequencies were an order of magnitude lower than the Strouhal shedding frequency. Although limited amounts of data were available to study the trailing vortex oscillations, it was determined that the frequencies of the oscillations were slightly dependent on the formation geometry. The wind tunnel turbulence factor was substantially large (TF=1.2) and needed to be accounted for. Frequencies of flow turbulence fluctuation present within the tunnels' free stream flow were caused by several factors, the most significant being axial fan effects and vibration of the tunnel walls and support bracing. Once all of the dynamic flow frequencies were determined a suitable sampling rate and period were chosen. In order to obtain an accurate sampling of the dynamic flow effects, and avoid sampling error, the sampling rate is usually set to twice that of the highest expected frequency. Then, in order to ensure the acquisition of a statistically representative sample, the period was set to a minimum of the longest dynamic flow factors' period. Due to the limitations of the ESP and velocity probe system, the upper limit of the frequency response was only 50Hz. Although this was well below the maximum frequencies expected, an accurate sample average could still be obtained by increasing the acquisition time duration significantly. By using an acquisition time orders of magnitude longer than the lowest frequency's' period, any aliasing of the data could be compensated for. Although this is method does not avoid aliasing, the extremely long acquisition time period will help to reduce its effect. Probe data were taken at 256Hz for a period of four seconds for all gridpoints.

As was mentioned, acquiring data with a velocity probe required a significant amount of time, and was an important limiting factor. Once the minimum sampling rate and time was chosen, the total acquisition time was only dependent on the time required to traverse the positioning motors plus the time required to acquire and save data for each gridpoint. To keep tests to a reasonable amount of time, the number of data points in each plane of interrogation could not be too large. In contrast, too coarse a spatial resolution of gridpoints would increase the error of any interpolated values (i.e. velocity, vorticity) between measured points and thereby affecting the final circulation calculations and/or lift calculations. The calculation of the vorticity fields utilized a derivative method, therefore, too large a spatial resolution would result in a large amount of error. A spatial resolution of 0.5 x 0.5cm was used for total lift determination tests. Tests for any wing formation required an overall grid size of 11 x 30cm to capture the downstream

flowfield effects. For other tests a spatial resolution of 1mm was needed. Tests with only one wing present could be performed with a smaller grid, shortening acquisition time.

During post processing of the data, calculations were performed on the wake data to determine the strengths of the trailing vortices. As will be seen in Chapter 3, careful choice of integration boundaries was needed to minimize data processing error. In Chapter 1.3 the differences between real and ideal vortices were mentioned. For real vortices, a viscous core is always present within the center of each trailing vortex. Acquiring velocity data within the core, while using an ideal flow assumption (as given by Equation 3) to calculate circulation, will yield errors in the magnitude of the circulation. Because the lower velocities within the core are not predicted by ideal flow theory, the use of Equation 3 to calculate circulation from velocities measured while within the core region, will underestimate the amount of circulation for the vortex. Similarly, although to a lesser extent, calculating the vortex circulation for a real vortex while using velocity data far outside the core region will also slightly underestimate the total circulation because the lower velocities far from the core are somewhat reduced by viscous effects and interaction with surrounding flowfield disturbances. To minimize error, circulation calculations utilizing Equation 3 were only performed using velocity data acquired just outside of the core region.

Chapter 3

3 WAKE DATA METHOD

The purpose of the research was to develop a method for the determination of lift for wings experimentally without the need for a direct load measurement device. The method chosen would rely on data taken downstream from a wing, eliminating the need for bulky mounting devices that could interfere with the nearby flow and potentially introduce additional experimental error. Two slightly different methods to determine the lift from downstream near-field data planes are proposed. Both approaches are based on ideal fluid mechanics principles. In the following sections, the general theory for each method is described, followed by their application to the cases of a single wing, and to a pair of wings in close proximity. Results from the application of the initial methods for the two test cases are shown within their respective sections.

3.1 Determination of Total Lift

To determine the lift generated by a wing, Equation 9 from ideal flow theory can be applied. Γ is the total amount of circulation bound on the surface of the wing per unit span, U is the free stream velocity, b is the wingspan, and ρ is the local flow density.

$$L = \rho U \Gamma b \quad (9)$$

This is a general expression from fluid mechanics and applies to any solid body immersed in a fluid flow that is generating lift, although its application to wings is most prominent. Equation 9 can be rewritten in a more functional form and is shown below.

$$L(y) = \int \rho U \Gamma(y) dy \quad (10)$$

Here Γ is allowed to vary in the spanwise direction leading to a lift that also varies as a function of the spanwise position. There are several assumptions made when stating either of the above expressions. The density and free stream velocity are taken to be constants, implying incompressible and steady flow. For the current problem of lift determination, the density and free stream velocity are known quantities, and the total amount of circulation bound on the wing is left to be determined. The simplest approach is to assume a constant distribution of circulation across the wings' surface. This assumption is best suited for

representing two-dimensional models where vorticity is generated uniformly across the lifting surface, and similarly, shed uniformly from the trailing edge across the span of the wing. The application of this expression to obtain an estimate of lift for three-dimensional wings can also be suitable, but only for certain conditions. It is known that for any three-dimensional wing, most of the vorticity is shed from the wingtips. Still, some vorticity is shed from the trailing edge just inboard of the tips, and to a lesser extent, along the trailing edge near the midspan. Because the total amount of vorticity for any lifting surface must be conserved, the amount of vorticity bound on the wings' surface must decrease as it is simultaneously shed from the trailing edge in the spanwise direction. As the aspect ratio becomes larger, a smaller percentage of the total vorticity is shed inboard of the tips. Many previous experimental tests have shown that the distribution of bound vorticity for a three-dimensional wing decreases in a monotonic fashion toward the wingtips. As the wings' aspect ratio approaches infinity the vorticity shed inboard of the tips become negligible, and therefore, calculating the total lift for a three-dimensional wing can be done as if it were two-dimensional with little to no significant error. Even for wings with aspect ratios as low as four, all of the vorticity shed near the wingtips will have rolled into the tip vortices a short trailing distance downstream of the wing. For very short aspect ratio wings, using Equation 9 to calculate total lift can lead to significant error because the amount of bound vorticity is not approximately constant across a majority of the span. For these situations the differential form of the expression, Equation 10, can be used with better accuracy.

The total lift for a single wing was calculated using an ideal flow approach. The three-dimensional rectangular wing models were assumed to be of large enough aspect ratio so that practically all of the vorticity generated on over the wing was shed at the wingtips. Even for cases with more than one wing in formation, where an asymmetric upwash distribution was present across the wings, all of the vorticity was assumed to shed from the wingtips. This was a significant assumption because it implied there was a constant distribution of circulation across each wings' span. This assumption considerably reduced the amount of required computation to obtain results because only the circulation contained within the tip vortices needed to be considered.

For the current research, the determination of total wing lift is approached assuming a constant distribution of circulation across the span of the wing, and that all vorticity is shed from the wingtips. By performing an integration of the velocities present in the wake the total amount of circulation bound on the wing, and therefore total lift generated by the wing, was determined. Figure 18 shows a theoretical representation of this configuration. The view is from above the wing and the free stream velocity approaches from the top of the figure. Each horizontal line represents a separate vortex line of unspecified strength, the properties of which have been discussed in Chapter 1.3.

Figure 18: A constant distribution of bound vorticity with all trailing vorticity shed from the wingtips for a three-dimensional wing.

When viewing the wing in Figure 18 from a downstream position, the individual vortex lines that shed from the trailing edge near the wingtips will begin to roll around each other and form wingtip vortices. The assumption that all vorticity is shed exactly from the tips implies the immediate formation of wingtip vortices. Figure 19 shows a theoretical representation of the velocity field present in the wake behind a single rectangular wing that is generating lift. The velocity vectors for this figure were computed using two Oseen vortices according to Equation 7. A vortex is shed from each tip, each having equal strength but opposite sign. Because of the previous two assumptions, the amount of circulation present at any cross section of the wing is equal to the circulation present within one vortex.

Figure 19: Two-dimensional theoretical representation of real wake velocity using two superimposed Oseen vortices.

To determine the amount of circulation present on a single rectangular wing, we must only measure the strength of one of the trailing vortices. The total circulation for a vortex can be found using one of two methods. The circulation for a single vortex can be found by either performing a line integration of velocities tangential to a closed circuit lying in a plane perpendicular to the axis of the vortex, or by performing an integration of the two-dimensional vorticity over a specified area. The theory behind each method was briefly discussed in Chapter 1.3. Here both methods are described in detail, and then applied to the cases of a single rectangular wing and a formation of two wings (Chapters 3.2 and 3.3).

3.1.1 Velocity Integration Method

For the first method, velocities are integrated around a closed contour encompassing the center of the vortex. According to theory, the value of circulation calculated is not dependent on the shape or the size of the contour. As long as the contour encloses the vortex origin the result will be accurate. Because of this reason a circular path is chosen to simplify calculations. Although we are employing an ideal flow approach,

the viscous effects within the vortex cannot be neglected completely. The circulation calculation is based directly on the magnitude of the tangential velocity therefore, choosing a suitable radius for the circular integration path becomes essential. Choosing too small of a radius will underestimate the value of circulation. During the test case for a single rectangular NACA 63-420 wing (see Chapter 3.2) we experimentally investigate the effect of the circles' radius on the circulation calculation.

Additionally, it is known, that as the distance from the wings' trailing edge to the data plane is increased, the vortex begins to increase in diameter slightly with a simultaneous reduction in velocity magnitude. Viscous effects cause this vorticity diffusion. For a more fully diffused vortex, an integration path of greater radius is needed to capture the total vortex circulation. By integrating the lower velocities around a longer contour, the net result is an approximately constant circulation for near field measurements. For this reason, the effects due to viscosity in the streamwise direction are negligible, assuming the contour does not pass through, or very near, the core region of the vortex. For an Oseen vortex, which represents a real vortex suitably, a convenient cut-off point containing 99.34% of the original circulation is to select an integration path having a radius of twice the radius of maximum velocity⁴.

Now that we have established the theories behind the calculation of lift for a wing and of circulation for a vortex, we can identify the specific process used to determine circulation and lift from our wake data. As described in detail in Chapter 2.4, a flow measurement probe connected to an ESP pressure scanner was implemented to obtain pressure measurements in the wake of the wings. By performing a reduction of pressures measured with the probe, the three-dimensional magnitude and direction of the flow at various locations within a two-dimensional plane was established. The three components of the measured velocity and their orientation with respect to the mean flow direction are shown in Figure 20. Velocity components v and w lie within the plane of interrogation while the u -component is oriented perpendicular to the plane.

Figure 20: Orientation of the u , v , and w -velocity components within a near field plane.

The implementation of a computer program to perform the velocity integration is now discussed. Looking at Figure 19, it is apparent that v and w -velocity values are only known at a limited number of locations (or gridpoints) in the plane. The gridpoints are equally spaced and arranged in horizontal rows and vertical columns. Figure 21 is a depiction of the path of integration that was used by the computer program. Because the velocity grid had a coarse resolution it was necessary to implement an interpolation scheme for locations between gridpoints.

Figure 21: The applied path of integration.

A computer code was written in Matlab to import the data saved by the AeroAcquire acquisition software and calculate the circulation strengths of each of the trailing vortices. After manually specifying the center of the vortex in question, and the desired radius for the circular path of integration, the program generated points along the circular path. The path was always discretized using 200 points for radii ranging

from five to twenty millimeters, yielding a dS interval between 0.157 and 0.628mm. At each of the 200 points, a linearly interpolated velocity was calculated. If the point in question coincided exactly with an existing gridpoint, the velocity was taken as measured. If the point lay directly between two gridpoints in either the x or y -directions, the velocities to either side were used to interpolate the new velocity. For all other cases, the four surrounding known velocities were used for the interpolation. Summing the velocities around the circular contour yielded the circulation for the vortex. The velocities were nondimensionalized by the free stream velocity, U_∞ , and the distances were nondimensionalized by the chord of the small wing, c .

3.1.2 Vorticity Integration Method

A second way to experimentally determine the vortex circulation strength was investigated. As mentioned in Chapter 1.3 the circulation strength for a vortex can be determined by integrating the vorticity over an area. Application of this concept required that the two-dimensional vorticity field be calculated from the velocity field. To get the vorticity, the velocity field is simply differentiated. Comparisons of several vorticity calculation schemes are described.

Previous research by Etebari and Vlachos¹⁸ was used to compare several vorticity calculation schemes for their applicability to the current problem. They applied several different methods of vorticity estimation to simulations of known flow fields to quantify errors. Of greater importance to this research, they determined the effects of spatial sampling resolution on the bias errors of the various methods. Because of the large spatial resolution between the gridpoints of the current research, this comparison provided most helpful. From their work, seven different schemes are compared. Figure 22 shows the percentage error incurred by using each of the methods applied to cases for a single Oseen vortex, and a general velocity field with wave numbers 1-7 respectively. It was apparent that the Compact-6, Chapra, and the Richardson-6 methods performed well at high spatial resolutions for both cases. Note that any of the methods can yield substantial error when applied to a typical velocity field (Figure 22) if the spatial resolution is not considered.

Figure 22: Normalized average error % versus spatial resolution for various schemes: a) applied to an Oseen vortex, b) applied to a velocity field containing wave numbers 1-7.

According to Etebari and Vlachos, “The higher-order, less smoothing schemes (sixth-order compact and Richardson extrapolation and the Chapra fourth-order formulation) exhibit error levels that are dominated by random noise at high spatial sampling resolutions, while the schemes that demonstrate lower-order accuracy and more smoothing (second-order central finite difference, 8-point circulation, and the fourth-order noise optimized Richardson schemes) exhibit error levels that are very sensitive to poor spatial sampling resolution.”

The spatial resolution for the current research varied approximately between $\delta/L=0.35$ and 0.5 , depending on the particular situation. This value is based off the diameter of the vortex and the resolution of the grid. A case where three gridpoints (two intervals) span the width of the vortex would correspond to a $\delta/L=0.5$. Based off the above analyses, and consideration for the amount of computing time required to process the data, the 6th order compact scheme to calculate the vorticity fields from the velocity fields was chosen and implemented. According to Etebari and Vlachos, “The sixth-order accurate compact scheme delivers significantly less bias error compared with previously used methods, particularly when the domain is spatially undersampled and in the presence of strong velocity gradients. The compact scheme demonstrated less than 0.3% bias error throughout the core of the vortex, resolving flow structures as small as the Nyquist sampling frequency of the system.” Equation 11 shows a generalization of the Pade scheme, the scheme used for the 6th order compact method. The constants α , β , a , b , and c are solved for by substitution of Taylor series coefficients, with f and f' representing the values of the velocity and implicitly determined local derivative values. In particular, $\alpha=1/3$, $\beta=0$, $a=14/9$, $b=1/9$, and $c=0$.

$$\beta f'_{i-2} + \alpha f'_{i-1} + f'_i + \alpha f'_{i+1} \beta f'_{i+2} = c \frac{f_{i+3} - f_{i-3}}{6h} + b \frac{f_{i+2} - f_{i-2}}{4h} + a \frac{f_{i+1} - f_{i-1}}{2h} \quad (11)$$

Figure 23, also from the work of Etebari and Vlachos, depicts a visual representation of the points required to perform the calculation of the vorticity. Eight neighboring gridpoints are used to determine the derivative at the central location.

Figure 23: Eight neighboring gridpoints (circled) are used to enable the calculation of the vorticity at the center location with the 6th order compact scheme.

The acquired velocity data were processed using this 6th order vorticity calculation scheme. By doing this, 2D vorticity fields were generated from the 2D velocity fields. Before the application of Equation 5 to determine the circulation for a vortex, the area for integration must first be defined. Because of the coarse spatial sampling of the data, there were a limited number of data points involved in the calculation. For this reason, the process of defining an area and performing the integration was not automated, but done manually within a spreadsheet program.

3.2 Test Case For a Single Wing

In this section we apply the methods previously described in Chapters 3.1.1 and 3.1.2 to the case of a single rectangular NACA 63-420 wing. A constant free stream velocity of 20m/s and a nominal angle of attack of eight degrees were used. First we present the near field velocity field. In Figure 24 we see the velocity field obtained with the velocity probe. The distance between the trailing edge of the wing and the data plane was three chord lengths, 3c. From a cursory visual inspection alone, there appears to be two vortices of equal and opposite strength, each shed from opposite wingtips. Note the flow disturbance in the lower center portion of the flowfield. This was caused by the mounting sting, which was attached to the wing at the midspan location of the quarter chord line. The integration paths that were used to calculate the circulation of each vortex, also indicated in Figure 24, were circles.

Figure 24: Experimentally measured velocity flowfield for a single rectangular wing.

Next the strengths of the trailing vortices were calculated using Equation 3, according to the velocity integration method described previously in Chapter 3.1.1. Because the general shape of the tip vortices was circular, circular integration paths could be used to simplify calculations. Figure 25 shows the strengths of the tip vortices as a function of circle radius. Note that when the circle radius is very small, a smaller circulation is obtained. In theory, for an ideal vortex, the strength of the vortex is independent of the integration path chosen as long as it encompasses the origin of the vortex. The reason for the lower values at smaller radii was due to the lower velocities within the vortices viscous core. This was already discussed in Chapter 1.3 and shown in Figure 2. Also note the slight oscillation of the curve. This effect is due to the choice of velocity interpolation scheme implemented within the circulation calculation program. Depending on whether the point for which the velocity was being calculated lay directly on an existing gridpoint, between two gridpoints in either the x or y -directions, or between four gridpoints in both the x and y -directions, the number of terms used in the approximation was different. For all cases, the final value for the circulation was taken to be the maximum calculated value. It is noted that the strengths of the two vortices are of opposite sign and approximately equal, as expected for a single rectangular wing. The strength of the left tip vortex was $-0.2285\text{m}^2/\text{s}$ while the right tip vortex strength was $0.2474\text{m}^2/2$. The small deviation may be due to a slight misalignment or wind tunnel variation.

Figure 25: Circulation strength as a function of integration circle radius for left and right tip vortices.

Next the vorticity integration method was applied for the same case. The velocity field was first processed with the 6th order vorticity calculation scheme described in Chapter 3.1.2. Figure 26 shows this vorticity field.

Figure 26: Vorticity field for a single rectangular wing calculated using a 6th order compact scheme.

After the vorticity field was calculated, the strengths of the tip vortices were determined using the vorticity integration method. By extracting the vorticity values from the gridpoints corresponding to the

location of the vortices within the spreadsheet program, multiplying each value by its associated elemental area, dA , and then summing, the total circulation for the vortices was obtained. Using this method, values of $-0.2416\text{m}^2/\text{s}$ and $0.2595\text{m}^2/\text{s}$ were obtained.

Comparing the circulation results from both methods for the single wing case yielded a small difference. Using either case, the right side vortex was between 7-9% greater in strength than the left. The circulation values calculated using the vorticity integration method were 5.7% and 4.9% (left/right) larger than those calculated using the velocity integration method. The larger percentage error corresponded to the vortex with the smaller overall strength. A summary of the circulation strengths is shown in Figure 27.

Figure 27: Comparison of circulation values calculated using both the velocity integral method and vorticity integral method.

Using the assumption that all vorticity generated on the surface of the wing is shed directly from the wingtips, the total lift for the wing can be determined from the trailing vortex strengths. Since the amount of vorticity shed from each tip is equal, the circulation contained within one tip vortex equals the amount of circulation around the wing section. The lift can then be calculated using Equation 9.

3.3 Application to Formation Flight

In this section we again apply the methods described in Chapters 3.1.1 and 3.1.2 but for the case of a pair of wings. A NACA 63-420 wing was placed $0.5c$ downstream and $0.25c$ outboard of a NACA 0012 wing. The free stream velocity was 20m/s . The NACA 63-420 angle of attack was set to an eight degrees and the NACA 0012 was set to 12 degrees angle of attack. Probe data were acquired in a manner similar to the single wing case. The post-processing techniques were identical also. The data plane was set at a distance of $3c$ from the trailing edge of the leading wing. Below are the velocity and vorticity plots along with the corresponding circulation strengths for the formation case. Note that there are three tip vortices present. There is one shed from the leading wing (far left) and two shed from the trailing wing.

Figure 28: Experimentally measured velocity flowfield for a two-wing configuration.

Circulation strengths calculated using the velocity integration method are shown in Figure 29 below. The NACA 0012 tip strength was $0.4816\text{m}^2/\text{s}$. The vortex strengths for the NACA 63-420 wing were $-0.3626\text{m}^2/\text{s}$ and $0.2243\text{m}^2/\text{s}$.

Figure 29: Circulation strengths as a function of integration circle radius for leading and trailing wings.

Figure 30: Vorticity field for a two-wing formation calculated using a 6th order compact scheme.

Applying the vorticity integration method, circulation strengths for the three vortices from left to right were $0.47548\text{m}^2/\text{s}$, $-0.37207\text{m}^2/\text{s}$, and $0.24386\text{m}^2/\text{s}$ respectively. When the NACA 63-420 wing was placed in the wake of the leading wing, a large difference in vortex strength is seen when comparing the left and right vortices. Comparing both methods, there was a -0.8% difference between the leading wings' vortex strength, a 2.7% difference between the left trailing wings' vortex strength, and a 8.4% difference between the right trailing wings' vortex strength. As before, the larger error percentages corresponded to vortices with smaller overall strengths. Compared to the NACA 63-420 wing flying solo, there was an average circulation increase of 23% in the trailing vortex strengths for this particular formation case.

Figure 31: Comparison of circulation values calculated using both the velocity integral method and vorticity integral method.

Additional tests were performed in which both the spanwise and streamwise gaps were varied. Results for these cases can be found in the Results chapter (Figure 55).

As shown, both methods produce similar results for the calculation of circulation. Without a known case to compare the two methods, there was no concrete way to judge the accuracy of either method, although several theories and options can be considered. First, the generation of the vorticity field from the velocity field is yet one additional required calculation. In no way can performing additional calculations reduce the amount of error present, unless by chance. The scheme used to calculate the vorticity from the velocities had to be chosen carefully; using a first or second order derivative method (as commonly done) would have resulted in error as large as 100% for the given spatial resolution of the grid. As implemented, the summation of the vorticity values was performed over a very coarsely spatially sampled grid. To increase the accuracy of this method, additional vorticity values would need to be interpolated over the entire region, greatly increasing the computational complexity. Using the velocity integration method to calculate circulation only required the calculation of interpolated velocities about a line contour.

To summarize, the best choice for the determination of the tip vortex strengths would be to implement the velocity integration method, but to also include a higher order approximation when performing the velocity interpolation. Because of the above considerations, the velocity integration method was employed for all further tests to decrease the complexity of calculations, minimize the time requirement, and reduce potential error.

Chapter 4

4 METHOD DEVELOPMENT

The initial methods, described in the previous chapter, were developed to enable the determination of the total lift for a wing using solely the strength of its trailing vortices. For the case of the single NACA 63-420 wing, all the vorticity generated on the surface of the wing was assumed to shed directly from the wingtips. Had the amount of vorticity shed from each tip been equal, the circulation contained within one tip vortex would have equaled the amount of circulation around any given cross-section of the wing. The implication of this would have meant a constant distribution of circulation across the span of the wing enabling the use of Equation 9 to calculate the lift. From the data presented above for the formation case, one problem became apparent: The strengths of the trailing wings' tip vortices were not of equal and opposite strengths. This was seen in Figure 29 or Figure 31 at the end of Chapter 3.3. In this particular case, the strength of the trailing wings' inboard tip vortex was greater than the outboard vortex. Vorticity must be conserved; this implies that vorticity must be shed nonuniformly from the trailing edge of the wing, and not only at the tips (the initial assumption), to account for the difference. Acceptance of this fact led to an asymmetric distribution of circulation across the wings' span and therefore, an asymmetric distribution of vorticity in the wings' wake. This asymmetry was most probably due to the upwash caused by the leading wing. No longer was the assumption of a constant circulation distribution across the trailing wings' span valid! Because of this unexpected result, a modification to the method was required in order to provide an accurate distribution of circulation across the wings' span. Knowing the circulation distribution across the span of the wing would enable the calculation of the wings' rolling moment in addition to its lift distribution and total lift.

Determination of the Spanwise Circulation Distribution

Because of the significant amount of vorticity shed from the wings' trailing edge inboard of the wingtips, and because of the asymmetric distribution of vorticity in the spanwise direction, it was more accurate to take these effects into account before further study. Figure 32 shows approximate circulation distributions calculated using an ideal flow horseshoe vortex upwash approach for several different aspect ratio wings at a constant angle of attack. Notice as the aspect ratio becomes smaller, a greater amount of the total vorticity is shed inboard of the tips. Not until the aspect ratio becomes very large does the circulation distribution approach a constant across the span of the wing. It should be noted that the distributions in the

figure are for fully attached turbulent flow over the airfoil. The trailing wing used for this research had an aspect ratio of four. Although the vorticity shed inboard from the tips for this wing was not substantial, not accounting for it would have increased the error in the circulation calculation.

Figure 32: Distribution of circulation for a rectangular wing of various aspect ratios.

In addition to the non-constant spanwise circulation distribution, the wake shed from the trailing edge that is initially flat, tends to roll-up at the edges due to its self-induced velocity. This rollup begins as soon as the vorticity is shed from the trailing edge. At locations farther downstream the wake can be seen to narrow in width. An accurate determination of the lift distribution across the span of the wing is not possible from a far downstream position because of the redistribution in space of the trailing vorticity. For example, a vortex line shed directly from a wingtip of the wing will begin to roll into the tip vortex, inherently changing its location in both the spanwise and chordwise directions with respect to its original position. To determine the spanwise lift distribution for the wing, a more direct approach to the measurement of circulation was taken. Instead of taking measurements in a downstream plane perpendicular to the direction of the flow as for earlier tests, velocity measurements for the calculation of circulation were taken in planes that were perpendicular to the span of the wing at various spanwise stations and very near the wing. Several of these arbitrary planes within which data were acquired are depicted in Figure 33.

Figure 33: Depiction of arbitrarily placed spanwise data planes used for the determination of the circulation distribution.

To introduce the theory behind this improved method, the case of fully attached potential flow over an airfoil section is considered. Figure 34 shows this flow for an airfoil at an arbitrary angle of attack. An arbitrary path of integration is applied that encompasses the airfoil, shown with a dashed line, and then Equation 3 is used to calculate the circulation for the section. This arbitrary path will be called a “full contour,” because it is fully enclosed.

Figure 34: Potential flow streamlines over an airfoil at an angle of attack (with randomly shaped path of integration).

Because data were being acquired with a flow measurement probe, the ability to traverse the tip of the probe to all locations along the contour was not possible without disturbing the flow. To prevent this, and at the same time minimize the quantity of data that needed to be acquired and processed, we can apply several simplifying assumptions to the choice of integration path assuming certain criteria can be met. According to ideal flow theory, the choice for the path of integration for the circulation calculation is independent of the path of integration as long as the path is a closed circuit that encloses the origin. The application of Equation 3 is the equivalent to the summation of the tangential component of the velocity vectors along the path of integration. Figure 35 shows the same airfoil section as Figure 34, but with a

rectangular path of integration. This rectangular path is still considered a “full contour”. In this instance, the top and bottom portions of the path are parallel to the direction of the free stream while the right and left sides are perpendicular to the free stream. The velocities near the surface of the airfoil are tangential to the surface of the airfoil due to the no-separation assumption. Depending on the proximity of the rectangular path to the airfoil, the velocities may or may not be in line with the free stream. Velocity vectors closer to the surface of the airfoil experience a greater deviation from the free stream direction than vectors farther away from the surface. As the distance from the surface approaches infinity, the directions of the streamlines become identical to the direction of the free stream. At these locations there are no disruption in the direction of the velocities from the free stream direction.

Figure 35: Potential flow streamlines over an airfoil at an angle of attack (with randomly shaped path of integration).

The wake and trailing vortices persist for an infinite distance downstream according to ideal flow. Even for experimental conditions, these structures exist for extremely long distances behind the wing. To simplify the circulation calculation, three of the four sections (top, bottom, and left sections) of the rectangular path can be moved to infinity, causing the direction of the velocity vectors to be either parallel or perpendicular to the flow along these sections of the path. For a real flowfield having viscous properties, we need travel much less distant than infinity to achieve undisturbed velocities. Far from the airfoil, the velocities become perpendicular to the direction of the path of integration, therefore this section does not contribute to the circulation integral. On the top and bottom portions, the velocity vectors are perfectly in line with the direction of the free stream in addition to having a magnitude equal to that of the free stream. Because the integration path has a positive convention that is counterclockwise, the vectors on the bottom of the path are in the opposite direction of the integration direction. Because of equal and opposite contributions on the top and bottom portions of the path, they cancel each other out, and therefore the sum can be neglected. Only the right section of the path in Figure 35 must be included in the circulation calculation to obtain the total circulation for the section. This is of immense significance when defining the

data necessary to be acquired with the flow measurement probe, as the acquisition time can be drastically reduced. This single line section will be called a “partial contour.” For reference, two different lengths of partial contours were used during tests. The short contour will be called a “short partial contour,” and the longer one a “long partial contour.” It is important to remember that because the top and bottom portions of the path are assumed to be a very far distance from the airfoil section, the length of the right section must be large to avoid error. In the wind tunnel test environment, the flow is parallel to the top and bottom walls very near the walls. Error can be introduced if the walls are too close to the wing. According to previous research, the effects of the wing are most significant to within ± 2 -3 chords distance from the wings’ surface. The total height of the test section was 10 chords. Traversing the velocity probe to the full vertical extents of the traversal system would have been sufficient, but to reduce acquisition time shorter partial contours were used. Figure 36 below depicts the three types of contours used. Vertical gridpoint spacings of 1mm were implemented for the determination of the spanwise lift distributions. Because a resolution of 1mm was used in the vertical direction, greater distances between spanwise columns of data were used to minimize total acquisition time.

Figure 36: Three different contour types used to obtain spanwise circulation values.

To summarize, the theories applied for the calculation of circulation were essentially the same for both this and the initially proposed method, except that the data planes were oriented in different directions. The improved method provides a more accurate determination of circulation because of the closer proximity of the measurement probe to the wing. Most importantly, the ability to determine the circulation as a function of spanwise position can allow for a more accurate determination of lift for the wing, as the circulation can be integrated over the wings’ span. As an added benefit to the improved method, less post-processing of data is required. To further minimize the effect of the trailing wings’ mounting sting (evident in the velocity and vorticity plots), a thinner sting was made and attached to its outboard wingtip.

Chapter 5

5 RESULTS AND DISCUSSION

As an introduction to the *Results and Discussion* section, a brief summary of the preliminary results presented in Chapters 3 and 4 is provided. These results provided a framework on which subsequent tests were based, and on which the current chapter is organized. In addition to reiterating the previous preliminary data, new data will be presented here. All results from Chapter 3 were based on the initially proposed method. The goal was to determine the total lift experienced by the wing solely through the measurement of the trailing tip vortex strengths. Velocity and vorticity fields, taken with a flow measurement probe, were shown for a single wing and for a two-wing formation. Two methods to calculate the circulation in the wake were described and applied to the cases of a single wing and a pair of wings. Based on preliminary results obtained from the tests in Chapter 3, deficiencies in the method were identified, and modifications and improvements were made to enhance the method. Chapter 4 described these changes and outlined a new, improved method, for the determination of the circulation distribution. Data acquired using the improvements are included in this chapter. Like the initial tests, the improved method was applied to the cases of a single wing and formation flight. Lastly, loads measured directly with the loadcell are presented.

5.1 Initial Results

In Chapter 3, the wake velocity and vorticity fields for the single NACA 63-420 wing were shown (Figure 24, Figure 26). The wingtip vortices for this wing were shown to be approximately of equal strength and opposite sign (Figure 25, Figure 27), as expected. Two different approaches for the circulation calculation, described in Chapters 3.1.1 and 3.1.2, were applied to this single wing case (Figure 27). There was shown to be a 5.3% average difference in the circulation calculation between the two methods for the single wing case, with the vorticity integration method producing slightly larger circulation magnitudes. Wake velocity and vorticity fields were also shown for a two-wing formation (Figure 28, Figure 30). Again, the two different circulation calculation methods were applied. There was shown to be a 4.33% difference in the circulation calculation between the two methods for the two-wing configuration with the vorticity integration method producing slightly larger circulation magnitudes.

Additional wake velocity and vorticity fields, similar to those in Chapter 3, were acquired for the formation cases. The leading wing remained fixed while the trailing wing was moved in a horizontal plane. The location of the data plane remained fixed at $3.0c$ behind the trailing edge of the leading wing. Probe data were collected for three streamwise separations ($0.5c$, $1.0c$, and $1.5c$) and three spanwise separations (0 , $\pm 0.25c$), where c was the chord of the smaller trailing wing. In addition, two cases representing docked flight were tested; the leading edges of both wings were aligned for one case, and the trailing edges were aligned for the other case. For these docked cases, there was zero gap between the wingtips of the two wings. Figure 37 through Figure 54 show the velocity and vorticity fields for the 11 cases mentioned above, including the docked flight results. The formation case from Chapter 3.3 is shown again in Figure 41 and Figure 42.

Figure 37: Velocity field for $0.5c$ trailing distance, $-0.25c$ spanwise gap ($0.5c, -0.25c$).

Figure 40: Vorticity field for $0.5c$ trailing distance, $0c$ spanwise gap ($0.5c, 0c$).

Figure 41: Velocity field for $0.5c$ trailing distance, $+0.25c$ spanwise gap ($0.5c, +0.25c$).

Figure 42: Vorticity field for $0.5c$ trailing distance, $+0.25c$ spanwise gap ($0.5c, +0.25c$).

Figure 43: Velocity field for $1.0c$ trailing distance, $-0.25c$ spanwise gap ($1.0c, -0.25c$).

Figure 44: Vorticity field for $1.0c$ trailing distance, $-0.25c$ spanwise gap ($1.0c, -0.25c$).

Figure 45: Velocity field for $1.0c$ trailing distance, $0c$ spanwise gap ($1.0c, 0c$).

Figure 46: Vorticity field for $1.0c$ trailing distance, $0c$ spanwise gap ($1.0c, 0c$).

Figure 47: Velocity field for $1.0c$ trailing distance, $+0.25c$ spanwise gap ($1.0c, +0.25c$).

Figure 48: Vorticity field for $1.0c$ trailing distance, $+0.25c$ spanwise gap ($1.0c, +0.25c$).

Figure 49: Velocity field for $1.5c$ trailing distance, $-0.25c$ spanwise gap ($1.5c, -0.25c$).

Figure 50: Vorticity field for $1.5c$ trailing distance, $-0.25c$ spanwise gap ($1.5c, -0.25c$).

Figure 51: Velocity field for $1.5c$ trailing distance, $0c$ spanwise gap ($1.5c, 0c$).

Figure 52: Vorticity field for $1.5c$ trailing distance, $0c$ spanwise gap ($1.5c, 0c$).

Figure 53: Velocity field for $1.5c$ trailing distance, $+0.25c$ spanwise gap ($1.5c, +0.25c$).

Figure 54: Vorticity field for 1.5c trailing distance, +0.25c spanwise gap (1.5c, +0.25c).

For all cases there were three tip vortices present, one from the leading wing and two from the trailing wing. Near the bottom center portion of the velocity and vorticity plots the disturbance caused by the original mounting sting can be seen. An additional observation of interest was the smearing of the velocity and vorticity vectors for certain test positions due to trailing vortex instabilities. In all three of the plots where the trailing wing was slightly inboard ($-0.25c$) of the leading wing, there was seen instability of the inboard vortex of the trailing wing. This vortex tended to oscillate about the larger neighboring vortex shed from the leading wing, causing the smearing.

By processing the velocity fields, the strength of each of the trailing vortices was calculated using the integral method described in Chapter 3.1.1. Figure 55 shows wingtip vortex strengths for the nine different positions of the trailing wing relative to the leading wing. Each plot represents a different downstream position showing data for the inboard, aligned, and outboard positions. Although lines are drawn connecting the points of the graph, there is a considerable chance that the actual values would deviate from the lines had data been acquired for additional intermediate spacings. In either of these plots of trailing vortex strengths, it was difficult to identify any trends or establish a balance of vorticity shed from the trailing wing until the vorticity shed from inboard of the tips was accounted for. Of the three plots in Figure 55, the data for the $-0.25c$ spanwise gap has the greatest probability of error because of the inability of the circular integration contours used to accurately encompass the area of velocity smearing. Analyzing the tip

vortex strength data, one trend can be identified. In general, as the streamwise spacing of the two wings was increased for the aligned ($\xi=0c$), the individual vortex strengths and the total amount of vorticity shed from the wingtips were greater. Data for additional spacings would be required to make accurate statements about the overall trend as a function of spanwise tip spacing.

Figure 55: Trailing vortex strengths versus spanwise separation for inboard, aligned, and outboard positions of the trailing wing.

The docked flight cases used the NACA 0012 wing as the larger leading aircraft (the mothership, MS), and the trailing NACA 63-420 wing for the smaller trailing aircraft (the hitchhiker, HH). Both of the cases simulated a configuration in which the hitchhiker aircraft would be physically attached to the mothership aircraft. The combination of both wings acted like a single wing with varying chord. Configurations of this type have been studied before and employed in real life applications as well. For the first docked case the leading edges of the wings were aligned. For the second docked case the trailing edges

of the wings were aligned. It is important to note that the wings had different chords and camber; the mothership had a chord twice that of the hitchhiker. Because of this, the elevation of the hitchhiker wing was slightly less for the case when the trailing edges were aligned. Figure 56 shows how the two airfoil sections were aligned for leading edge-to-leading edge (LE-LE) and trailing edge-to-trailing edge (TE-TE) tests.

Figure 56: Alignment of mothership (gray) and hitchhiker (black) wings for docked flight.

For the docked flight cases, only two vortices were observed, one from the leading wing, and one from the outboard wingtip of the trailing wing. Only because the leading wing had a larger chord was there any vorticity shed at the interface between the two. There was more vorticity shed from the leading wings' tip for the case when the leading edges were aligned. For these tests, there was zero gap present between the wings while acquiring velocity probe data.

Figure 57: Velocity field for docked flight, leading edges aligned (LE-LE).

Figure 58: Vorticity field for docked flight, leading edges aligned (LE-LE).

Figure 59: Velocity field for docked flight, trailing edges aligned (TE-TE).

Figure 60: Vorticity field for docked flight, trailing edges aligned (TE-TE).

Wingtip vortex strengths for the docked flight tests are shown in Figure 61 and Figure 62. The results were as expected. For the case of the trailing edges aligned, there was only minimal vorticity shed from the tip of the mothership. Most of the vorticity shed in this case was from the outboard tip of the hitchhiker. For the case of the leading edges aligned, there was still a significant amount of vorticity shed from the tip of the mothership, and the amount of vorticity shed from the tip of the hitchhiker was reduced.

Comparing the total amount of trailing vortex circulation present in the wake, values of $0.6098\text{m}^2/\text{s}$ and $0.5821\text{m}^2/\text{s}$ were obtained for the LE-LE and TE-TE tests respectively.

Figure 61: Trailing vortex strengths for docked flight: Leading edges aligned.

Figure 62: Trailing vortex strengths for docked flight: Trailing edges aligned.

An important observation noted in every tested case, was the presence of vorticity of opposite sign shed from the inboard portions of the trailing wing to the midspan. Figure 63 is shown to illustrate this occurrence, although it can be seen in any vorticity plot in the Results chapter. In Figure 63 the three tip vortices can be seen, in addition to some vorticity shed from the trailing wings' sting (near the bottom of the figure). The vorticity of opposite sign can be seen between the 1.3 and 4 x/c locations. The vorticity centered at about $x/c=2.2$ is of opposite sign than the left tip vortex at $x/c=0.85$. The vorticity centered at about $x/c=3.1$ is of opposite sign than the right tip vortex at $x/c=4.3$. In some cases, it is even possible to visualize the presence of this negative vorticity from the corresponding velocity fields. This effect was also noticeable in the results from the work of Vlachos and Telionis⁵⁰. Their work was performed in a water tunnel and used a noninvasive particle image velocimetry (PIV) system to acquire data.

Figure 63: Vorticity plot showing vorticity of opposite sign present on the midspan ($1.0c$ trailing gap, $0c$ spanwise separation).

As mentioned, the midspan sting generated a significant flow disturbance even though it was streamlined. Again this can be seen at the bottom of Figure 63 near the $x/c=2.5$ location. The initial method only utilized data from the cores of the tip vortices, and didn't require the data near the midspan. Because of this, the disturbance created by the sting wasn't a large problem. After the initial method was modified and the circulation distributions became required, the disturbance became an issue, and a new sting was implemented.

5.2 Final Results

Data acquired using the improved method described in Chapter 4 are now shown. This allowed us to determine the circulation distribution. A new, thinner end-mount sting was made to eliminate the interference near the midspan before proceeding with tests. First data is shown for a single wing, and we see that it is almost symmetric. In theory, this distribution should be perfectly symmetric, but due to experimental error, wind tunnel free stream variance, and the endmount sting, there was a slight deviation. Afterwards, data is shown for a formation flight configuration. A discussion of the shape and magnitude of the circulation distributions is performed.

Figure 64 shows the circulation distribution, calculated using full contours, for the NACA 63-420 wing flying solo. In Figure 65 and Figure 66 we again see the circulation distributions for the NACA 63-420 wing flying solo, but long and short partial contours were used for the acquisition and calculations instead of full contours. Both figures showing results for the partial contours present lower circulation values than for the full contour case in Figure 64. The length of the partial contour used for Figure 65 was 140mm (approximately 2.8 chord lengths), while the length used for Figure 66 was only 60mm (approximately 1.2 chord lengths). It was important to note the difference in circulation magnitude measured using the different contours.

Figure 64: Circulation distribution calculated using full contours for the NACA 63-420 wing flying solo.

Figure 65: Circulation distribution calculated using long partial contours for NACA 63-420 wing alone (2.8c contour height).

Figure 66: Circulation distribution calculated using short partial contours for NACA 63-420 wing alone (1.2c contour height).

Next data for the formation flight case of $0.5c$ trailing distance and $+0.25c$ spanwise gap ($0.5c, +0.25c$) using full contours is shown in Figure 67. For this, and all formation cases, the circulation biased towards the inboard (left) side of the trailing wing. This was due to the upwash provided by the leading wing. The effects are similar to what would be obtained if there were only one wing, but it was twisted, with a higher AOA near the inboard side.

Figure 67: Circulation distribution for NACA 63-420 wing in close proximity to the NACA 0012 leading wing. Calculated using full contours ($0.5c, +0.25c$ case).

Here we discuss the previous plots showing the circulation distributions. There were two main contributing factors that affected the *shape* and *magnitude* of the measured distributions. 1) The shape of the circulation curve was influenced by the physics of the flow present. 2) The average magnitude of the circulation curve (for the same test case) was influenced by the various testing techniques used.

1) Evident in the plots, the amount of circulation calculated near the midspan is less than towards the wingtips. This effect is contrary to the typically expected distribution. Usually the distributions for unseparated flow over a rectangular wing behave as shown in the below figures. Figure 68 and Figure 69 show the expected shapes of the circulation distributions for a rectangular wing without and with twist. The case of a single wing with twist is provided to represent the trailing wing in close proximity to the leading

wing. The upwash from the leading wing causes the effective angle of attack of the trailing wing to be greater towards the inboard (left) side.

Figure 68: Approximate ideal flow distribution of circulation for a 3D wing with zero twist.

Figure 69: Approximate ideal flow distribution of circulation for a 3D wing with twist.

Theoretical representations of the expected vorticity distribution for the two above cases are now shown. Scenario I corresponds to Figure 68 and Scenario II corresponds to Figure 69. In the spanwise direction, as vortex lines leave the wings' surface, the amount of circulation present for that particular section is reduced. The vortex line figures allow us to predict the relative amount and position of the shed vorticity in the wake of the wing for each circulation distribution present across the wing.

Scenario I

Scenario II

These same ideas, presented above, can be applied to the actual data obtained from experimental tests to qualitatively compare the obtained circulation distributions to the distribution of vorticity in the wake. Two experimental cases were analyzed. Studying the distribution and magnitude of vorticity in the 2D vorticity plots (Figure 26, Figure 42) for the test cases shown in Figure 64 and Figure 67 (both the single wing and the $0.5c + 0.25c$ formation case), the distributions of vortex lines can be established as before. These theoretical depictions are provided in an attempt to account for the vorticity of opposite sign inboard of the wingtips. Because vorticity must be conserved, and vortex lines cannot end within the interior of the

flow, the vorticity of opposite sign and of smaller magnitude inboard of the tip vortex (seen for all test cases) must be explained somehow. Scenario III approximately represents the case observed for the single wing. The tip vortices are of approximately equal strength. The vortex lines that bend and exit the wings' surface between the tip and midspan account for the vorticity of opposite sense seen in Figure 26. Additionally, a lower value of circulation is present around the wings' midspan because of this situation. Drawing the spanwise distribution of circulation corresponding to Scenario III would yield the same distribution as the experimentally determined distributions in Figure 64 or Figure 65. As more vortex lines leave the wings' surface, the value of circulation decreases. Similarly, Scenario IV represents the distribution that existed for the formation case of $(0.5c, +0.25c)$ seen in Figure 42 and Figure 67.

Scenario III

Scenario IV

2) Now that we have accounted for the shape of the observed circulation distribution curves, we can analyze the reasons for the magnitude of the circulation that was measured. The average magnitude of

the circulation was apparently influenced by the testing technique used. More specifically, there was a difference in the overall magnitude of the measured circulation depending on whether a full contour, a long partial contour, or a short partial contour was used. In theory, as long as none of the assumptions related to the development of the contours were broken, integrating the tangential velocities for each type of contour should yield the same circulation magnitude. In actuality, it was extremely difficult to determine exactly how long each contour needed to be for each test case. Too short a contour failed to capture all of the disturbed velocities present near the section and caused a smaller than actual circulation value to be measured. Because of this, using a full contour yielded the largest circulation magnitude. The long partial contour (2.8c) yielded the next largest value of circulation, while the short partial contour (1.2c) yielded the smallest value.

5.3 Confirmation of Flow Separation

In the previous section we qualitatively justified the measured circulation distributions using vortex line depictions from fluid mechanics theory. Here we attempt to determine why the unusual circulation distributions existed. It was hypothesized that flow separation was occurring and causing large-scale flow disruptions. We show u and w -velocity profiles near the midspan for the single NACA 63-420 wing. Flow visualization utilizing a PIV system was also implemented to visually confirm the separation of the flow for all the tested angles of attack. Figure 70 shows the u -velocity deficit and the downwash velocity directly behind the wing at a location near the midspan at $AOA=8$. The velocity probe was traversed vertically while data were acquired. The minimum velocities near 70mm are coincident with the location of the trailing edge of the wing. In addition to the velocity deficit, the vertical component of the velocity (the w -component) along the same acquisition line is plotted. Note the negative w -velocity above the trailing edge of the wing, and the switch to a positive w -velocity below. Normally the downwash created by unseparated flow over a wing always stays negative directly behind the wing, although some variation may still be seen near the trailing edge. To verify this behavior for other angles of attack further tests were performed. Figure 71 shows the same data but for $AOA=4$. In Figure 72 the velocity deficit is shown for -4 , 4 , and 8 degrees angle of attack. As the angle of attack was increased, the width of the velocity deficit region increased with a simultaneous increase in the magnitude of the w -component of the velocity just above and below the deficit region. These trends can be seen clearly in the figure. After flow separation was suspected for the NACA 63-420 wing, PIV flow visualization was performed for confirmation. At all angles of attack the flow visualization showed separation of the flow directly from the leading edge when placed at an angle of attack and from the maximum section thickness location when placed at zero degrees (both actual or effective) angle of attack. The Reynolds number for the water tunnel tests was 25,000.

Figure 70: Streamwise velocity deficit and downwash for AOA=8.

Figure 71: Streamwise velocity deficit and downwash for AOA=4.

Figure 72: Streamwise velocity deficit for several angles of attack.

5.4 Load Measurements

Load measurements were taken for various relative streamwise (ξ) and spanwise (η) spacings of the two wings in addition to two cases representing docked flight. As explained earlier, load measurements were taken using a 6-DOF balance to compare to existing literature and to serve as a means to verify the accuracy of near field probe tests. For all load measurement cases the non-cambered swept NACA 0012 airfoil, used as the leading airfoil, was set to a 12° angle of attack. Similarly, the cambered symmetric NACA 63-420 airfoil was set to a nominal 8° angle of attack. Due to the airfoil camber, the zero lift angle of attack for this airfoil was -4° , resulting in a total effective angle of attack of 12° . All angles were measured using a digital inclinometer aligned with the mean chordline. For each streamwise separation ($\xi=0.5c, 1.5c, 2.5c$, and $2.5c$ with a 10% height reduction) spanwise separations of $\eta=-0.5c, -0.25c, -0.05c, 0c, +0.05c, +0.25c$, and $+0.5c$

were tested. For docked flight the leading wing was termed the mothership (MS), while the trailing wing was termed the hitchhiker (HH). During docked flight, only positive values of $\eta=0c$, $0.05c$, $0.25c$, and $0.5c$ were tested. For the first docked case the leading edges of the wings were aligned ($LE_{MS}-LE_{HH}$). For the second docked case the trailing edges of the wings were aligned ($TE_{MS}-TE_{HH}$). A total of 36 formation positions were measured including the docked configurations. Data for these cases are shown in the figures below. Figure 73 through Figure 76 show load lift, drag, and rolling moment (C_L , C_D , and C_m) values for the formation flight cases. Figure 77 shows lift, drag, and rolling moment data for the $LE_{MS}-LE_{HH}$ docked case. Figure 78 shows the lift, drag, and rolling moment for the $TE_{MS}-TE_{HH}$ case. The parameters C_D and C_L have been nondimensionalized by the tunnel static pressure and the planform area. The rolling moment, C_m , has been nondimensionalized by the tunnel static pressure, the planform area, and the wingspan.

Figure 73: Trailing wing loads ($\xi=0.5c$, $\eta=-0.5c$ through $+0.5c$).

Figure 74: Trailing wing loads ($\xi=1.5c$, $\eta=-0.5c$ through $+0.5c$).

Figure 75: Trailing wing loads ($\xi=2.5c$, $\eta=-0.5c$ through $+0.5c$).

Figure 76: Trailing wing loads ($\xi=2.5c$ with 10%c height reduction, $\eta=-0.5c$ through $+0.5c$).

Figure 77: Trailing wing loads (leading edges aligned, $\eta=0.0c$ through $+0.5c$).

Figure 78: Trailing wing loads (trailing edges aligned, $\eta=0.0c$ through $+0.5c$).

A note with respect to the docked flight cases: For both the docked flight cases where $\eta=0.0c$ deviating slightly from the zero gap arrangement while acquiring loads was required. It was important for the hitchhiker not to contact the mothership while acquiring the load data, else the measurement of extraneous loads would occur. To accurately represent the docked flight cases while acquiring load data, an extremely small gap was allowed between the mothership and the hitchhiker wings. A gap of no greater than 0.5mm, set between the two wings, satisfied this requirement and still produced accurate flow geometry.

Chapter 6

6 CONCLUSIONS AND FUTURE WORK

Methods for the determination of the trailing vortex strengths and the lift distribution for the trailing wing of a two-wing formation have been proposed and tested. Although further tests utilizing larger wings tested at higher velocities are still required to minimize low Reynolds number effects on the results, a good basis for further work has been established. To restate, the original goal was to use velocity or vorticity data acquired within a near-field downstream plane to determine total lift for the trailing wing in a two-wing formation by measuring only the strengths of the trailing vortices. In the end we found that near-field data (between $0c$ and $3c$ downstream), while useful for determining the strengths of the tip trailing vortices, do not provide enough information to determine the distribution of circulation across the wings span to enable the calculation of lift. A combination of effects contributed to this conclusion for the tested configurations. First, the low aspect ratio of the wings caused a sizeable amount of vorticity to be shed asymmetrically inboard of the wingtips causing a lift distribution that did not approximate a constant across the span as it would for longer aspect ratio wings. Not accounting for the inboard vorticity would introduce error in the lift calculation. By the time the wake reached the location of the data plane it had already rolled up at the edges, redistributing the location of the vorticity from where it had originally been shed at the trailing edge. Basing the bound vorticity distribution off of the vorticity distribution in the wake would have been incorrect. Had the aspect ratio of the wings been very large, the circulation distribution would have been approximately constant and all vorticity would have shed directly from the wingtips. This ideal scenario would have resulted in equal strengths of the trailing tip vortices, an effect that was not observed. Secondly, due to the very low Reynolds number of the flow ($\sim 67,000$), flow separation and trailing vortex wakes were always present. Although these occurrences did not compromise the validity of the results obtained with the final developed (spanwise) method, they made comparisons to existing literature difficult. Experimentally determined lift distributions for wings in close formation flight and low Reynolds numbers are almost nonexistent, especially when the data is acquired with a velocity probe as opposed to surface pressure taps. The interaction of the flow separation with other three-dimensional effects contributed to the presence of vorticity of opposite sign in the wake located directly between each tip vortex and the midspan locations. If the wake shed from the trailing wing had been symmetric, even though separated, a parametric study of the lift could have been performed by exclusively measuring the strengths of the tip vortices. Following is a

discussion of the most important conclusions relevant to the research. A mild attempt is made to discuss the conclusions in the same order that they were ascertained while conducting out the research.

Chapter 3 outlined the initial method developed. The method consisted of the acquisition of two-dimensional planes of data downstream of the wing formations. Wingtip vortex strengths were to be calculated from the near-field planes using either a velocity contour integration approach or a vorticity area integration approach. Accuracy of either method relied on the sufficient spatial resolution of the points within the data planes. As expected, there was a tradeoff between spatial sampling resolution and total acquisition time. Because time was an important limiting factor, establishing the correct balance between both was important. For each gridpoint time was needed to traverse the positioning motors, acquire the data, and save the data. An evaluation of the required spatial resolution was performed. The limiting factor determining the spatial resolution was the error produced when calculating the vorticity field from the velocity field. Calculating the vorticity from the velocity is a derivative process sensitive to the spatial sampling and may produce considerable error for poorly sampled regimes. For this reason both the method and the spatial sampling must be chosen carefully. By comparing the amount of error produced by several vorticity calculation schemes at different spatial resolutions, a minimum resolution of 5mm x 5mm was chosen. Using this resolution with a compact 6th order vorticity scheme allowed for a maximum of 20% error in the vorticity calculation for a general sample flowfield. It was discussed earlier how higher spatial sampling resolutions or lower order vorticity calculation schemes, as often used in research, can result in error as large as 100%.

Given an experimentally measured velocity field containing concentrated wingtip vortices, the use of the velocity integral and the vorticity integral methods yielded similar results despite the relatively coarse spatial sampling resolutions, with the vorticity integration method producing slightly larger circulation magnitudes for most cases. According to theory, both methods should yield exactly the same values. For the test case involving the single NACA 63-420 wing (Chapter 3.2), the circulation strengths for each wingtip vortex calculated using the vorticity integration method were 5.7% and 4.9% (left vortex/right vortex) larger than those calculated using the velocity integration method. The larger percentage error always corresponded to the vortex with the smaller overall strength. For the formation of both wings (Chapter 3.3), the vorticity integration method produced circulation values that were -0.8%, 2.7%, and 8.2% (leading wing vortex / inboard trailing wing / outboard trailing wing) greater than the velocity integration method for the three tip vortices present. Again, the largest percentage error corresponded to the vortex with the smallest strength, in this case the outboard vortex of the trailing wing. As applied, the vorticity integration method may have potentially introduced more error into the final circulation calculation than the velocity integration method, especially for the poorly spatially sampled velocity fields. The reason for the difference was

attributed to both, error in the vorticity calculation (the vorticity calculation scheme used tended to underestimate the value of vorticity near the center of each vortex), and the inability of the method to define the correct area needed for the area integral. The second issue could potentially be rectified by using a more robust area-defining method. Results calculated using the velocity integration method used linearly interpolated velocities in addition to directly measured velocities whereas, vorticity values were not interpolated between the coarsely sampled data plane points for the vorticity integration method. For application of the velocity integration method to calculate circulation of a single trailing vortex (Chapter 3.1.1), care must be taken to make sure points outside the viscous core region of the vortex are used. If the path is placed within the core region, where lower than expected velocities are present, the total amount of circulation for the vortex will be underestimated. For future comparisons between the two methods, the utilization of a higher resolution vorticity field (either measured or interpolated) for the vorticity integration method should be considered a necessity.

According to fluid mechanics theory for incompressible flow, the net amount of vorticity being generated on the surface of the wing must equal the amount of vorticity being shed in the wake of the wing for the steady state. Also the sum of the negative and positive vorticity shed from a single lifting surface must be zero. During close proximity tests of the two wings, the magnitudes of the tip vortices from the trailing wing varied with formation position and were always of significantly different strengths. The implication of this was that additional vorticity must be shed from elsewhere along the wing (i.e. the midspan). If all vorticity present in the downstream flowfield was only concentrated into the tip vortices, then the tip vortices should have been of equal strength. Closer inspection of the vorticity in the wake showed that vorticity of opposite sign of the tip vortices was shed inboard from the tips. Estimating the strength of the inboard vorticity and combining it with the vorticity from the tips resulted in an equilibrium of positive and negative vorticity shed from the trailing edge of the trailing wing. It was concluded that the inclusion of the inboard vorticity for any further analyses would be mandatory. The reason for the vorticity of opposite sign was perhaps one of the most important observations/conclusions. Although it was not expected at the beginning of the study, the flow over the wings was either separated or possessed a trailing vortex wake due to low Reynolds number effects. The interaction of the separated or wake region with the vorticity shed near the tips caused the unexpected regions of vorticity near the midspan. In general, any Reynolds number below 500,000 can be considered laminar. Any effects caused by the laminar flow will only be accentuated at Reynolds numbers below 100,000. Because of limitations on model size and wind tunnel velocity, the chord Reynolds number for all of the current tests was approximately 67,000.

Certain formation positions caused an unsteady motion of the trailing vortices to occur. For these cases, averaging the data in time caused a “smearing” of the tip vortices’ usual circular shape into more of an

elongated elliptical shape. Because the circulation calculation scheme used a circular integration path, it was not able to entirely encompass the required area, leading to error. Figure 79 shows this situation. Note how circular integration paths, of arbitrary radius C_1 or C_2 , cannot be placed such that the entire area of negative vorticity (blue region) is included without also including a portion of the neighboring vortex above.

Figure 79: Illustrating the insufficient circular integration path for the case of $(1.5c, -0.25c)$.

The averaging and smearing of the data due to vortex instability was not necessarily one of the main contributions of error for the time-averaged circulation calculations in and of itself. The error occurred due to the inability of the chosen method to sufficiently encompass the non-circular region of vorticity. A method utilizing non-circular integration boundaries could be implemented to better surround areas of arbitrarily shaped vorticity in the wake. Figure 80 shows one possible solution to accurately identify and measure a region of non-circular shape. To implement this scheme a computer program of greater complexity must be written to account for the arbitrary boundary shape. One proposed idea calls for a scheme that would identify contours of constant vorticity for use as the boundary for the area integral. For data planes with coarse spatial sampling, greater consideration for the choice of vorticity interpolation scheme should be applied.

Figure 80: Proposed solution for cases with smeared distributions due to vortex instability.

Another potential way to solve the problem of smearing of data due to averaging could be to acquire the entire downstream velocity field in a time-resolved manner. This would require instantaneous and repeated acquisition of data at all locations in the plane. A method to acquire data in this way for further study is through the use of a particle image velocimetry, or PIV, system. A PIV system would enable the acquisition of data in a more efficient manner, significantly reducing the acquisition time. Because this method greatly decreases the time requirement, more cases could have been studied. Although not the primary objective of the research, a better idea of the best location for the trailing airfoil could have been found using this alternative method. A PIV system works by illuminating a single flat plane within a particle impregnated flowfield with a laser sheet. Because an entire plane of data is illuminated at once, time dependent data for all locations in the flowfield can be acquired. Most of the readily available PIV systems are 2D, or single plane, systems. That is, only velocities that lie within the single illuminated plane can be resolved at one time. Application of this method for the current study would require the acquisition of data within two different perpendicular planes. For a downstream flowfield, with velocities in the v and w -directions as designated by Figure 20 only a single plane perpendicular to the direction of the mean flow need be employed. From this data, time resolved velocity vectors in the v and w directions could be used to study time-dependent vorticity and circulation distributions in the wake of the wing formations. The benefits of this technique include reduced acquisition time and the ability to obtain an entire 2D flowfield in a time resolved manner instead of an averaged sense. A drawback to this method is that the third velocity component, perpendicular to the plane of interrogation, cannot be resolved simultaneous to the other two components. To obtain the third component, the plane must be rotated 90 degrees so that the third velocity component lies within the plane. Whereas all three components can be simultaneously obtained with a velocity probe over the entire Trefftz plane (but requiring time averaging), multiple planes parallel to the mean flow direction must be taken to obtain the same data with a PIV system. For example, a velocity probe can acquire u , v , and w -velocity components for a 10 x 10 grid all within a single plane, requiring traversing of the probe between each of the 100 gridpoints. Using a 2D PIV system to obtain the same u , v , and w velocities would require 10 planes parallel to the mean flow direction, with traversing of the plane between each of the 10 planes. Depending on the particular situation, one disadvantages of a PIV system could be the loss of illumination intensity loss for highly diffused laser sheets. If large planes of data are required the choices are to either spread the light into a wide sheet, reducing the intensity, or acquire the plane in multiple sections. If multiple sections must be used, the data from one section to the next is no longer correlated in time, and dynamic effects that cross between sections cannot be studied. Application of a PIV system for the determination of the spanwise distribution of circulation would most likely suffer the same averaging effects as a velocity probe system due to the fact that the laser sheet would need to be traversed in a spanwise direction for each new position.

Drawing from the issues presented previously, the total lift for a mid-aspect ratio wing (in this case, $AR=4$) cannot be obtained from the strengths of the trailing vortices within a near-field velocity plane alone because doing so fails to account for all of the shed vorticity. Only for very large aspect ratio wings and chord Reynolds numbers over 500,000 may this be possible. To determine the lift for a wing, the circulation distribution must be obtained first, and then integrated across the span. At first it was proposed to determine the circulation distribution across the wing by dividing the already acquired 2D wake data into neighboring sections and finding the vorticity contained within each section. After further investigation, it became apparent that this was no longer a valid option because of the rollup of the wake at downstream positions.

After the initial method (Chapter 3) was developed and improved (Chapter 4), it was possible to determine the circulation distribution for wings with separated or unseparated flow. The improved method, developed as part of this research in Chapter 4, was able to accurately measure the circulation distribution given that it was correctly applied. Several types of contours were applied for the determination of the circulation around various spanwise wing sections. Either full or partial contours used by the improved method should have been able to accurately determine the circulation profiles of the rectangular three-dimensional wings, but discrepancies in the obtained circulation values depending on whether a full contour, long partial contour, or short partial contour was used were observed. To alleviate these discrepancies, improvements to the implementation of both the full and partial contours are proposed. It is the author's opinion, that either a full contour or a long partial contour will yield the same circulation value assuming some simple changes are made. If using a full contour, care should be taken to ensure a true "full" contour that does not miss any portion. If any portion of the contour is to be left out, then that entire section must be very far from the wing where the flow is effectively the freestream value. Using a long partial contour should be sufficient assuming data can be taken in a single line that extends to at least ± 2 chord lengths above and below the trailing edge.

A majority of the tests performed employed a mounting sting that was fixed to the model at its midspan. Once it was determined that the circulation distribution was needed, as opposed to only the strengths of the trailing vortices, obtaining accurate velocity measurements directly behind the entire span of the wing became of greater importance. The flow disruption caused by the mounting sting near the center of the span, which was insignificant while only measuring the tip vortices, became problematic when applying the modified method. For the final tests, a thinner and shorter mount was made and attached to one of the models' wingtips at the quarter chord location.

Using direct measurements from the loadcell balance, we were able to make several observations. Varying the spanwise separation did not seem to affect the drag values significantly for any of the tested streamwise separations. As the trailing wing was brought closer to the leading wing in the spanwise direction, both the lift coefficient and rolling moment increased. The load data was found to be in very good agreement with that of Magill et al.³⁵ Of the limited formation positions tested, the greatest increase in lift for the trailing wing was found for the case of $\xi=2.5c$, $\eta=-0.5c$, and a 10% c reduction in elevation of the trailing wing. This position was downstream, slightly inboard (overlapping wingtips), and slightly below, with respect to the position of the leading wing.

The most significant challenge of the current research was determining that the flow was separated. Without a means of flow visualization, the only way to determine whether an unsteady vortex wake or separated region was present was with several types of time consuming tests utilizing the velocity probe.

One initially proposed solution to alleviate the flow separation was to use a wing model having a smaller section thickness. An additional trailing wing, having a NACA 0012 cross-section, was manufactured using a rapid prototyping process. This wing had similar dimensions to that of the other NACA 63-420 trailing wing used for earlier tests except there was zero camber and only 12% thickness with respect to the chord. Because of its smaller thickness, compared to the NACA 63-420 wing model, there was less possibility of flow separation during the tests. Even using this thinner wing, the separation or vortex wake was seen distinctly. Based on this reason, and additional reviews of existing literature, it was concluded that there would always be a separated region accompanied by a vortex wake for the very low Reynolds number flows. To compliment this finding, several figures from the work of previous authors are presented. Figure 81, from the research of Mueller⁴¹, shows the unsteady wake present behind a two-dimensional NACA 66-018 wing at a six-degree angle of attack. This two-dimensional wing is free from end effects, but should produce similar results to our tests because of the similar geometry and Reynolds number. Figure 82, also from the work of Mueller⁴¹, shows smoke visualization over a Smooth Miley airfoil at a chord Reynolds number of 150,000. Even for Reynolds numbers as high as 150,000 do see the tendency of the flow to separate at low angles of attack.

Figure 81: Unsteady vortex wake behind a 2D NACA 66-018 wing at 6-degree AOA and $Re_c=40,000$ (sketch made from smoke visualization photograph) (Mueller⁴¹)

Figure 82: Smoke visualizations for a Smooth Miley airfoil at $Re_c=150,000$. (Mueller⁴¹)

Future studies intending to further develop formation flight load determination using a velocity probe should anticipate flow separation and/or trailing vortex wakes. Care should be exercised if surveying these types of wakes with traditional pitot or pitot-static probes, as the oscillating nature of the unsteady wake will cause fluctuating velocity directions that are outside the useable range of the probes. The use of a five or 7-hole probe would be required in the minimum. Because these effects are not fully avoidable, a fundamental understanding of airfoil performance within a laminar flow regime is required. Understanding the limiting angle of attack before separation for particular Reynolds numbers is important and could even be a topic of research. Using larger models and a higher test velocity are essential to increase the Reynolds number, thereby reducing the possibility of separation. The small size and low speed of the wind tunnel used for this investigation were contributing factors to the flow separation. The requirement to have fully three-dimensional wing models in a formation flight system requires either very small models, or a large test section.

The turbulence level of the wind tunnel used was of an undesirable level for the study of the low Reynolds number flow. Laminar flows are greatly susceptible to the effects of turbulence. This conclusion was based solely off further literature review completed near the end of the current research. In order to obtain significant and accurate formation flight results, further attention should be paid to the influence of the tunnel turbulence level, model surface roughness, and vibration issues. The effects on the results due to the experimental technique, instrumentation, data processing and procedures used, need should be evaluated further for quality control purposes. As a final concluding remark, it is the author's opinion that practical formation flight tests are not possible in a small, low-speed wind tunnel with high turbulence characteristics. If at all possible, larger models tested at a higher velocity would be ideal to provide more accurate results.

Future work

In addition to the several suggestions for future improvements to the developed methods interspersed throughout the conclusions section, additional suggestions are given here. The first suggestions discuss changes or improvements directly related to the completed research. The last few suggestions discuss other related topics of interest for future investigations.

- Mounting the models vertically within the wind tunnel test section would simplify mounting and traversing issues. Additional attention to model mounting methods when designing tests will help minimize flow interference. A thin, wingtip-mounted sting, as used for the final tests, is suggested for all tests.
- Using an even thinner (1/16") flow measurement probe will also reduce the disruption of the flowfield. An even better idea would be to use a fast-response velocity probe with embedded pressure sensors. Because this type of probe is capable of resolving oscillations with frequencies up to the kilohertz range, a more accurate average of the flow effects can be made. The frequencies present in the wake could also be resolved.
- For the acquisition of any near-field data planes, positioning the velocity probe a constant downstream distance from the trailing wing instead of a constant distance from the leading wing would be advisable for consistency. By doing this, any dissipation of the wake vorticity would be accounted for and direct comparisons from case to case would be possible.

- Calculations that could be performed, but were not addressed in this research include the calculation of the total kinetic energy in the wake or the measurement of the u -velocity deficit in the wake to determine form drag.
- Using a higher order method to interpolate velocities for the velocity integration method (Chapter 3.1.1) is suggested instead of the current linear approximation. The current approximation utilized a different number of gridpoints for the linear interpolation at different locations causing the value of circulation to oscillate slightly as the radius was varied instead of reaching a steady state (Figure 25). The use of a higher order approximation instead of a linear one may rectify this.
- With respect to the vorticity integration method (Chapter 3.1.2), the implementation of a computer program to automatically locate the centers of vorticity, determine an applicable boundary contour, and calculate the circulation would yield far more accurate results than the general approach taken here. Such a program would locate the centers of vorticity and define a suitable area boundary, even for cases when irregularly shaped areas are present.
- After perfecting the method to determine the spanwise circulation distribution, lift calculations for the trailing wing could be calculated for different formation positions and then compared to loadcell measurements. L/D vs. spanwise separation plots for each trailing distance could be made. If more circulation distributions had been measured and calculated, a comparison between the measured loadcell rolling moment and the moment calculated from the asymmetric circulation distribution could have been performed. Before this is accomplished, the circulation distribution method must be “fine tuned” to ensure its accuracy.
- A more rigorous investigation of separation hysteresis effects or the response of the separation to turbulence levels for low Reynolds numbers would be interesting. Spanwise and chordwise separation and transition to turbulence for wings in close formation flight could be analyzed.
- The use of alternate models for formation flight research could be performed. The use of flat plates or cambered flat plates as suggested by Laitone³¹, may yield higher L/D ratios for very low Reynolds number tests. In his study showed that the use of sharp edged models or flat plates actually increased the lift to drag efficiency for low Reynolds numbers when compared to a standard NACA 0012 airfoil. Figure 83 shows the variation of the L/D ratio versus angle of attack for several airfoil shapes at a low Reynolds number. Note that using both the sharp edged wedge and the cambered flat plate resulted in higher L/D values than the NACA 0012 wing, even for larger angles of attack.

An additional curve was plotted (dashed line) for the NACA 0012 wing to show the effect of a large increase in freestream turbulence.

Figure 83: Lift-drag ratio (L/D) variation versus angle of attack for several $AR=6$ rectangular planform wings at $Re=20,700$ (Laitone³¹)

- It was shown that the stability of the trailing vortices with respect to their position was affected by the relative position of the two wings in the formation. During several tests, oscillations of the smaller vortex about the larger one were observed for cases when the wingtip overlapped in the spanwise direction. Topics that could be addressed include the determination of positions for which instabilities occur, the frequencies and amplitudes of the oscillations, or whether the oscillations contribute to trailing vortex breakdown. To better study this phenomenon, an acquisition system able to resolve the entire flowfield in a time-resolved manner (such as particle image velocimetry) would be useful.
- A more suitable research topic for the low Reynolds achievable in the ESM low-speed wind tunnel, would have been to experimentally study the onset and position of the laminar separation bubble

and/or the transition to turbulence for a single two or three-dimensional wing. To simulate the effects of formation flight, a single wing with twist could be used.

By using a combination of experimental methods, a better understanding of three-dimensional flow effects at low Reynolds numbers, and for wings in formation, can be established. Also, through experimental research, the accuracy of mathematical or computational fluid dynamics methods can be verified, helping to develop better computational models.

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Appendix

%Program to generate circles for velocity interpolation and circulation calculation

```
clear vel1 vel2 height span inc xx yy th r npts mx my m Vx Vy Vxy Xcirclepoints Ycirclepoints v circ slope  
tanslope velslope radius circulation
```

```
inc=5; %gridspacing  
span=51; %grid width (no. points)  
height=23; %grid height (no. points)
```

```
%Read In Velocities  
for i=4*span+1:5*span  
    for j=1:height  
        vel1(i-4*span,j)=data(i,j); %x velocities  
    end  
end  
for i=5*span+1:6*span  
    for j=1:height  
        vel2(i-5*span,j)=data(i,j); %y velocities  
    end  
end
```

```
%Generate 2D Space For Plotting  
xx=0:inc:(span-1)*inc;  
yy=0:inc:(height-1)*inc;
```

```
%Plot Velocity Grid  
vel1=vel1';  
vel2=vel2';  
figure  
quiver(xx,yy,vel1,vel2), xlabel('position, mm'), ylabel('position, mm'), title('Velocity Vectors, (test34)'), axis  
equal  
hold on
```

```
%Define Vortex Center  
mx=37;  
my=52;  
m=[mx,my];
```

```
%Specify Circle  
minrad=5; %Specify minimum radius  
maxrad=25; %Specify maximum radius  
npts=200; %number of points for circle (first/last point are same)  
s=1;  
for rad=minrad:maxrad  
    clear Xcirpts Ycirpts xpos ypos Vx Vy Vxy a b c d v angle ds velslope tanslope  
    % Plots a circle with center in m = [mx, my] and radius r and with npts supporting points  
    th=linspace(0,2*pi,npts);  
    plot(m(1)+rad*cos(th),m(2)+rad*sin(th)),axis equal,axis([1 span*inc 1 height*inc])
```

```
%circle points are:  
%th=linspace(0,2*pi-2*pi/inc,npts);  
Xcirpts=[m(1)+rad*cos(th)];
```

```

Ycirpts=[m(2)+rad*sin(th)];

%Linearly Interoplate Velocities
for k=1:npts
    if ((Xcirpts(k)/inc-floor(Xcirpts(k)/inc)==0) & (Ycirpts(k)/inc-floor(Ycirpts(k)/inc)==0)) %calc using 1 point
        only
            xpos=Xcirpts(k)/inc+1;
            ypos=Ycirpts(k)/inc+1;
            Vx(k)=vel1(ypos,xpos);
            Vy(k)=vel2(ypos,xpos);
        elseif Xcirpts(k)/inc-floor(Xcirpts(k)/inc)==0 %calc using case 1
            ypos=floor(floor(Ycirpts(k))/inc)+1;
            xpos=Xcirpts(k)/inc;
            Vx(k)=vel1(ypos,xpos)*abs(Ycirpts(k)-(ypos+1)*inc)/inc+vel1(ypos+1,xpos)*abs(Ycirpts(k)-
ypos*inc)/inc;
            Vy(k)=vel2(ypos,xpos)*abs(Ycirpts(k)-(ypos+1)*inc)/inc+vel2(ypos+1,xpos)*abs(Ycirpts(k)-
ypos*inc)/inc;
        elseif Ycirpts(k)/inc-floor(Ycirpts(k)/inc)==0 %calc using case 2
            xpos=floor(floor(Xcirpts(k))/inc)+1;
            ypos=Ycirpts(k)/inc;
            Vx(k)=vel1(ypos,xpos)*abs(Xcirpts(k)-(xpos+1)*inc)/inc+vel1(ypos,xpos+1)*abs(Xcirpts(k)-
xpos*inc)/inc;
            Vy(k)=vel2(ypos,xpos)*abs(Xcirpts(k)-(xpos+1)*inc)/inc+vel2(ypos,xpos+1)*abs(Xcirpts(k)-
xpos*inc)/inc;
        else %calc x and y velocity components using neighboring 4 points
            ypos=floor(floor(Ycirpts(k))/inc)+1;
            xpos=floor(floor(Xcirpts(k))/inc)+1;
            a=abs(Ycirpts(k)-(ypos+1)*inc)/inc;
            b=abs(Ycirpts(k)-ypos*inc)/inc;
            c=abs(Xcirpts(k)-(xpos+1)*inc)/inc;
            d=abs(Xcirpts(k)-xpos*inc)/inc;

            Vx(k)=(vel1(ypos,xpos)*(b/(a+b))+vel1(ypos+1,xpos)*(a/(a+b)))*(d/(c+d))+(vel1(ypos,xpos+1)*(b/(a+b))+
            vel1(ypos+1,xpos+1)*(a/(a+b)))*(c/(c+d));

            Vy(k)=(vel2(ypos,xpos)*(b/(a+b))+vel2(ypos+1,xpos)*(a/(a+b)))*(d/(c+d))+(vel2(ypos,xpos+1)*(b/(a+b))+
            vel2(ypos+1,xpos+1)*(a/(a+b)))*(c/(c+d));
        end
        Vxy(k,:)= [Vx(k) Vy(k)];
    end
end
Vxy;
%figure
%quiver(Xcirpts,Ycirpts,Vx,Vy),axis equal

%Calculate Circulation: SUM(V.ds)
ds=sqrt((Xcirpts(1)-Xcirpts(2))^2+(Ycirpts(1)-Ycirpts(2))^2)/1000;
tanslope=atan2((Ycirpts(2)-Ycirpts(npts-1)),(Xcirpts(2)-Xcirpts(npts-1)));
velslope=atan2(Vxy(1,2),Vxy(1,1));
angle=(velslope-tanslope);%angle between velocity and tangent line
circ=sqrt(Vxy(1,1)^2+Vxy(1,2)^2)*cos(angle)*ds;
for k=2:npts-1
    tanslope=atan2((Ycirpts(k+1)-Ycirpts(k-1)),(Xcirpts(k+1)-Xcirpts(k-1)));
    velslope=atan2(Vxy(k,2),Vxy(k,1));
    angle=(velslope-tanslope);%angle between velocity and tangent line
    v=sqrt(Vxy(k,1)^2+Vxy(k,2)^2)*cos(angle)*ds;

```

```

    circ=circ+v;
end
circulation(s)=circ;
radius(s)=rad/1000;
s=s+1;
end
radius'
circulation'

```

%Program to extract velocity and vorticity data from 3pi program output file

```

clear i j span height vel1 vel2 velmag vort
span=61; %no. points
height=23; %no. points
inc=5; %distance between gridpoints (mm)

```

```

xx=0:inc:(span-1)*inc;
yy=0:inc:(height-1)*inc;

```

```

for i=4*span+1:5*span
    for j=1:height
        vel1(i-4*span,j)=data(i,j);
    end
end

```

```

%contour(vel1',20) %this is the U/Uo-velocity!!!

```

```

for i=5*span+1:6*span
    for j=1:height
        vel2(i-5*span,j)=data(i,j);
    end
end

```

```

%figure
%contour(xx,yy,vel1',vel2',20) %this is the V/Uo-velocity!!!

```

```

%Plot u and v velocity field
%%figure
%%quiver(xx,yy,vel1',vel2',1.3), xlabel('Position, mm'), ylabel('Position, mm'), title('Time-Averaged Velocity Vectors'), axis equal, axis([1 span*inc 1 height*inc])

```

```

%for i=8*span+1:9*span
%    for j=1:height
%        velmag(i-8*span,j)=data(i,j); %this is velocity magnitude
%end
%end
%figure

```

```

%contourf(xx,yy,velmag',20), xlabel('position'), ylabel('position'), title('Velocity Magnitude'), axis equal, axis([1 span*inc 1 height*inc])

```

```

for i=9*span+1:10*span
    for j=1:height
        vort(i-9*span,j)=data(i,j); %this is x-vorticity
    end
end
figure

```

```
contour(xx,yy,vort',40), title('Time-Averaged Vorticity, 1/s (Trailing Edges Aligned)'), xlabel('Position,  
mm'), ylabel('Position, mm'), axis equal, axis([1 span*inc 1 height*inc])  
caxis([-6000 6000]),colorbar;
```