Introduction to aeronautics: a design perspective

INTRODUCTION TO AERONAUTICS: A DESIGN PERSPECTIVE CHAPTER 3: AERODYNAMICS AND AIRFOILS “ Isn’t it astonishing that all these secrets have been preserved for so many years just so that we could discover them!! " Orville Wright 3. 1 DESIGN MOTIVATION The Physics of Aerodynamic Forces Figure 3. 1 shows a cross section view of an aircraft wing. A wing cross section like this is called an airfoil. Lines drawn above and below the airfoil indicate how the air flows around it. The shape of the airfoil and the pattern of airflow around it have profound effects on the lift and drag generated by the wing. Aircraft designers choose a particular airfoil shape for a wing in order to optimize its lift and drag characteristics to suite the requirements for a particular mission. It is essential that an aircraft designer understand how the changes that occur in air as it flows past a wing create lift and drag, and how airfoil shape influences this process. Figure 3. 1. Flowfield Around an Airfoil The Basis for Airspeed Indication The changes which occur in the properties of moving air as it encounters obstructions provide the basis for the airspeed indicating systems used on most aircraft. An understanding of how these systems work is essential to anyone who designs, builds, or operates aircraft. 3. 2 BASIC AERODYNAMICS The Language A number of terms must be defined to facilitate a discussion of aerodynamics. The lines in Figure 3. 1 which indicate how the air flows are known as streamlines. Each streamline is drawn so that at every point along its length, the local velocity vector is tangent to it. A tube composed of streamlines is called a stream tube. In a steady flow, each streamline will also be the path taken by some particle of air as it moves through the flowfield (a region of air flow). A steady flow is defined as one in which the flow properties (pressure, temperature, density and velocity) at each point in the flowfield do not change with time. If, as in Figure 3. 2, a streamline runs into an obstruction, the airflow along the streamline comes to a stop at the obstruction. The point where the flow stops is called a stagnation point, and the streamline leading to the stagnation point is called a stagnation streamline. Figure 3. 2. Stagnation Point and Stagnation Streamline If, at each point along a streamline, there is no variation in the flow properties in a plane perpendicular to the flow direction, the flow is said to be one-dimensional. Figure 3. 3 illustrates a flow that is one-dimensional at stations 1 and 2. Station 2 Station 1 Figure 3. 3. Flow Which is One-Dimensional at Station 1 and Station 2 The Continuity Equation Figure 3. 3 depicts a flow in a stream tube. Because the walls of the stream tube are composed of streamlines, the velocity vectors are everywhere tangent to the walls of the tube, so no air can pass through the tube walls. The rate at which mass is flowing through a plane perpendicular to a one-dimensional flow is given by: (3. 1) where is the mass flow rate and A is the cross-sectional area of the stream tube. In nature, in the absence of nuclear reactions, matter is neither created nor destroyed. Therefore, mass which flows into the tube must either accumulate there or else flow out of the tube again. The case where matter is accumulating in the tube like air filling a balloon is an unsteady, time-varying flow. If the flow is a steady flow, then the rate at which mass is flowing into the tube at station 1 must just equal the rate at which mass is flowing out if the system at station 2: (3. 2) Equation 3. 2 is known as the continuity equation. It is a statement of the law of conservation of mass for fluid flows. Applying this equation to the flowfield shown in Figure 3. 3 reveals a phenomenon which is very important to the production of aerodynamic forces. If we assume that the flow is incompressible (density is constant everywhere in the flowfield) or at least that the changes in air density are small, then (3. 2) makes it obvious that the reduction in stream tube area at station 2 will produce an increase in the velocity there relative to the velocity at station 1. A simple demonstration of this effect occurs when an obstruction such as a person's thumb is placed over the end of a garden hose which has water flowing out of it. The obstruction of the flow reduces the area of the stream tube and forces the fluid to accelerate in order to maintain the mass flow rate. Figure 3. 4 shows a stream tube in a portion of the flowfield around an airfoil. The airfoil is an obstruction to the flow. It reduces the area of the stream tube and forces the flow to speed up as it flows around it. The changes which occur in the properties of the air as it flows past the airfoil produce aerodynamic forces. Figure 3. 4 A Stream Tube in Air Flowing Past an Airfoil Example 3. 1 Air flows through a tube which changes cross-sectional area similar to the one illustrated in Figure 3. 3. At a point in the tube (Station 1) where the cross-sectional area is 1 m2, the air density is 1. 2 kg/m3 and the flow velocity is 120 m/s. At another point in the tube (Station 2) the cross sectional area is . 5 m2 and the air density has decreased to 1. 0 kg/m3. What is the mass flow rate through the tube and what is the flow velocity at station 2? Solution: Using (3. 1), the mass flow rate is: Then, solving (3. 2) for V2 : Euler's Equation To understand the changes which occur in the flow properties of a fluid as its velocity changes, consider an infinitesimally small particle of air moving along a streamline in a steady flow, as shown in Figure 3. 5. A number of forces may act on this particle. Gravity and magnetic fields may exert body forces on it. Viscous shear forces may retard the particle's motion. Pressure imbalances may also exert a net force. If we consider only flows of relatively lightweight gases which do not have large vertical components and no strong magnetic attractions, then the effects of body forces may be ignored. If we consider only inviscid (frictionless) flows, then viscous shear forces can also be ignored. For such a situation, the only significant forces remaining are due to pressure imbalances along the streamline. Figure 3. 5. Forces on a Fluid Element Applying Newton's second law to the motion of the particle along the streamline, the sum of the forces in the streamwise direction, , is equal to the mass of the fluid particle multiplied by the rate of change of its velocity: now the volume of the fluid particle is the infinitesimal streamwise distance, ds, multiplied by the area of the perpendicular face, dA, so: m = r ds dA Also, since the velocity vector is everywhere tangent to the streamline, the direction of ds is everywhere parallel to the local velocity, so: which yields: or: (3. 3) Equation (3. 3) is called Euler's equation, after the eighteenth-century Swiss mathematician, who first derived it. The differential equation is a statement of Newton's second law for a weightless, inviscid fluid. It essentially states that for any increase in a fluid's velocity, there must be a corresponding decrease in its pressure. Because it relates the rate of change of a fluid's momentum to the forces acting on it, (3. 3) is also known as the momentum equation. Bernouilli's Equation For many purposes, the integral form of (3. 3) will be more useful to us. For a compressible fluid, the integral of the right-hand side requires a relationship for density. However, many useful flow problems can be solved with reasonable accuracy by assuming density has a constant value throughout the flowfield. This is an extremely accurate assumption for liquids. It also gives reasonable results for air if the velocities throughout the flowfield remain below 100 m/s or 330 ft/s. With r assumed constant (incompressible flow,) integrate (3. 3) from some arbitrary point along the streamline, station 1, to another point, station 2 to yield: or: (3. 4) Equation (3. 4) is known as Bernouilli's Equation after another eighteenth-century Swiss mathematician, Daniel Bernouilli. The two terms on each side of Bernouilli's equation are given descriptive names. The pressure term is called the static pressure. The velocity squared term is called the dynamic pressure, and is often identified by the symbol q. (3. 5) The sum of static pressure and dynamic pressure is called total pressure. It is identified by the symbol Po. Total pressure in a flow governed by (3. 4) is invariant along a streamline. When using (3. 4), it is important to remember that it is only valid for the steady flow along a streamline of an inviscid, incompressible fluid for which body forces are negligible. Together with the continuity equation, Bernouilli's equation provides the key to understanding such diverse concepts as how wings generate lift and how airspeed indicating systems work. 3. 3 BASIC AERODYNAMICS APPLICATIONS Airspeed Indicators One of the simpler applications of the aerodynamic equations developed to this point is the analysis and design of common airspeed indicating systems. These systems function by using the relationship between pressure and velocity described by Bernouilli's equation. Figure 3. 6 shows a schematic Figure 3. 6. A Pitot-Static Tube and Manometer The system consists of a Pitot tube, one or more static ports, and a device for indicating differential pressure (a manometer in Figure 3. 6.). The Pitot tube is named for its inventor, Henri Pitot, an eighteenth-century French scientist. It is placed in a flowfield with its opening perpendicular to the flow velocity so that if its opposite end were open, air would flow directly through it. Since the opposite end of the Pitot tube is blocked by the differential pressure indicator, the air in the tube cannot flow, and a stagnation point exists at the entrance to the tube. We assume that if we look far enough upstream in the flowfield, the flow becomes essentially undisturbed by the Pitot tube and any shape to which it is attached. The undisturbed flow is called the free stream, and the properties of this undisturbed state are called the free stream conditions. Free stream conditions are usually identified by a subscript infinity, e. g. . Since total pressure is constant along a streamline, the total pressure for the stagnation streamline leading to the stagnation point at the entrance to the Pitot tube is: (3. 6) Velocity is zero at the stagnation point, so (3. 4) requires that the static pressure there is equal to the total pressure. The Pitot tube therefore measures the total pressure of the flow and transmits it to one side of the manometer. The static ports are oriented parallel to the flow velocity so that no stagnation point develops and the pressure they measure is as close to the free stream static pressure as possible. Aircraft designers use great care in placing static ports, and they often use multiple ports in order to get good approximations to the free stream static pressure. The static ports in Figure 3. 6 are placed on the sides of the Pitot tube to form a Pitot-static tube. The static pressure is transmitted through the connecting tube to the other side of the manometer. Solving (3. 6) for yields: (3. 7) Example 3. 2 A manometer connected to a Pitot-static tube as in Figure 3. 7 has a difference in the height of the two collumns of water of 10 cm when the Pitot-static tube is placed in a flow of air at standard sea level conditions. What is the velocity of the airflow? Solution: In a normally functioning Pitot-static tube, the pressure measured at the static port will always be lower than or equal to the total pressure measured at the stagnation point, so the collumn of water connected to the static port will be higher than the other. Using the manometry equation, with the subscript o identifying total pressure and the subscriptidentifying the freestream static pressure approximated at the static port: Then, substituting the required values into (3. 7): The manometer or other differential pressure device measures the difference between the total pressure and the static pressure of the free stream. According to (3. 6), this difference is the dynamic pressure. If the air density is known, then the dynamic pressure is a direct indication of the free stream velocity. In aircraft, a differential pressure gauge is normally used instead of a manometer. In the differential pressure gauge, the static and total pressure lines are connected to opposite sides of a metal diaphragm. The pressure difference causes the diaphragm to deflect. A linkage connected to the diaphragm moves a needle on the gauge dial when the diaphragm moves. By calibrating the dial scale in terms of velocity instead of pressure, the differential pressure gauge becomes an airspeed indicator. Figure 3. 7 shows a schematic of an airspeed indicator connected to a Pitot-static tube. Figure 3. 7. Schematic of an Airspeed Indicating System ICeT The airspeed which the needle on the airspeed indicator points at for a given set of flight conditions is called the indicated airspeed. If the airspeed indicator is geared and calibrated based on (3. 6), then it is accurate only at speeds below 100 m/s or 330 ft/s where the flow is incompressible. Aircraft built prior to around 1925 operated exclusively at incompressible airspeeds and had incompressible airspeed indicators. Incompressible flow indicators are inaccurate for high speed flight and are no longer used. The Euler equation may be integrated without assuming incompressible flow. The details of this integration go beyond the scope of this text, but the result is a compressible form of Bernouilli’s equation. Virtually all modern airspeed indicators are geared and calibrated to represent the compressible analog of (3. 7) which is: (3. 8) Note that (3. 8) is not a simple equation to engineer into a mechanical instrument. In addition, values of r are difficult to measure accurately in flight. For these reasons, it is difficult to build a simple and reliable airspeed indicator based on (3. 8). Engineers surmounted this problem, however, by simplifying the equation. Airspeed indicators are manufactured with gears calibrated to use sea level standard atmospheric values of P and r. In effect, an airspeed indicator is calibrated to solve the expression: (3. 9) where Vc is called the calibrated airspeed. Yet, this is still not what is indicated on the airspeed indicator. The static ports on the aircraft may be located such that they do not accurately measure the freestream static pressure. This is referred to as position or installation error. Additionally, there may be small inaccuracies in the machining of the instrument. To account for these discrepancies, errors are quantified during flight testing and equated to a velocity change (Vp) called position error. The relationship between what is displayed on the airspeed indicator (indicated airspeed - Vi) and the calibrated airspeed is given as: (3. 10) On a perfect airspeed indicator, with zero position error, a pilot reading indicated airspeed would also be reading calibrated airspeed. However, in most cases DVp does not equal zero, and indicated airspeed will be slightly greater or less than calibrated airspeed. In order to obtain true airspeed (3. 8) from calibrated airspeed (3. 9), two corrections must be made, one for the actual existing pressure and the other for the actual existing density. Making the pressure correction yields equivalent airspeed which is defined as: (3. 11) Note that the actual static pressure is used in (3. 11), as opposed to the sea level values in (3. 9). The ratio between Ve and Vc is generally called the compressibility correction factor and is given the symbol f : (3. 12) where: (3.

13) Note that f varies only with (Po -) and. All other variables in (3. 13) are constant. can be obtained by setting a standard sea level reference pressure in the aircraft altimeter, and (Po -) can be obtained from knowing the calibrated airspeed. In this manner, a table of f factors such as Table 3. 1 can be produced which apply for any aircraft. It is normally more convenient to find a value for f from the table than to evaluate (3. 13). Table 3. 1. Compressibility Correction f Factors PressureAltitude(ft)| Calibrated Airspeed (knots)| | 100| 125| 150| 175| 200| 225| 250| 275| 300| 5000| 0. 999| 0. 999| 0. 999| 0. 998| 0. 998| 0. 997| 0. 997| 0. 996| 0. 995| 10000| 0. 999| 0. 998| 0. 997| 0. 996| 0. 995| 0. 994| 0. 992| 0. 991| 0. 989| 15000| 0. 998| 0. 997| 0. 995| 0. 994| 0. 992| 0. 990| 0. 987| 0. 985| 0. 982| 20000| 0. 997| 0. 995| 0. 993| 0. 990| 0. 987| 0. 984| 0. 981| 0. 977| 0. 973| 25000| 0. 995| 0. 993| 0. 990| 0. 986| 0. 982| 0. 978| 0. 973| 0. 968| 0. 963| 30000| 0. 993| 0. 990| 0. 986| 0. 981| 0. 975| 0. 970| 0. 963| 0. 957| 0. 950| 35000| 0. 991| 0. 986| 0. 981| 0. 974| 0. 967| 0. 959| 0. 951| 0. 943| 0. 934| 40000| 0. 988| 0. 982| 0. 974| 0. 966| 0. 957| 0. 947| 0. 937| 0. 926| 0. 916| 45000| 0. 984| 0. 976| 0. 966| 0. 956| 0. 944| 0. 932| 0. 920| 0. 907| 0. 895| 50000| 0. 979| 0. 969| 0. 957| 0. 944| 0. 930| 0. 915| 0. 901| 0. 886| 0. 871| For the density correction, observe that: and (3. 14) Since the density ratio is usually less than or equal to 1, is usually Ve. Notice that when flying at sea level on a standard day = 1, and = Ve. Recall that dynamic pressure is given by (3. 5) So that: (3. 14) Equivalent airspeed may be alternately defined as that airspeed that would produce the same dynamic pressure at sea level as is measured for the given flight conditions. It will become apparent later on in this chapter and in Chapter 4 that, in the absence of compressibility effects, aircraft with identical configurations and orientation to the flow will produce

the same aerodynamic forces if the dynamic pressures they are exposed to are the same. Since Ve is a direct measure of dynamic pressure, it is a very useful indicator of an aircraft's force generating capabilities. This fact is very useful to both engineers and pilots. Groundspeed It is worthwhile at this point to recapitulate the process for correcting an indicated airspeed. The steps are as follows: (3. 10) (3. 12) (3. 14) The result, , is called true airspeed. The series of corrections from indicated to calibrated to equivalent to true airspeed is often called an ICeT (" ice tee") problem, with the lower case e being used as a reminder that equivalent airspeed is usually less than the other airspeeds. However, true airspeed is frequently not very useful until another correction is made. is the magnitude of the aircraft's true velocity relative to the air mass. However, the air mass itself may be moving relative to the ground. The velocity of the air mass relative to the ground is the wind velocity. This must be added vectorially to the true velocity relative to the air mass in order to determine the aircraft's ground speed, Vg. Ground speed is the magnitude of the aircraft's velocity relative to the earth's surface. To help distinguish between true airspeed and groundspeed, consider the following example: Example 3. 3 An aircraft flying at 300 knots true airspeed has a 50 knot tailwind. What is its groundspeed? Solution: To obtain groundspeed, use vector addition: (3. 15) This example illustrates the important concept that an aircraft's motion relative to the earth may be significantly different in both direction and magnitude from its motion relative to the air mass. Whereas motion relative to the air mass is most important for generating sufficient aerodynamic forces to sustain flight, it is usually motion relative to the earth that allows an aircraft to fulfill its mission. In situations where headwind velocities approach the same magnitude as an aircraft's true airspeed, its usefulness compared to surface transportation can be greatly diminished. The following example gives a complete demonstration of the ICeT (actually ICeTG) method: Example 3. 4 An aircraft flying at 20, 000 ft pressure altitude has an indicated airspeed of 205 knots. If the outside air temperature is -20oF, position error is -5 knots, and there is a 40 knot headwind, what is the aircraft's groundspeed? Solution: Using (3. 10): Then, from Table 3. 1, for this altitude and calibrated airspeed, f = . 987 and using (3. 12): Now, from the standard atmosphere table in Appendix B, the pressure at a pressure altitude of 20, 000 ft is 973. 3 lb/ft2, so solving (2. 1) for the density: Note that it is not a standard day for the given conditions because the temperature is colder and therefore the density is higher than in the standard atmosphere at 20, 000 ft. Using r = 0. 001289 slug/ft3 in (3. 14): Now for the " G" step in ICeTG, the correction for wind velocity to determine groundspeed. The aircraft has a direct headwind of 40 knots, so its groundspeed is calculated from (3. 15) as: Low-Speed Wind Tunnels Wind tunnels are devices used to study the aerodynamics of aircraft and other shapes in a laboratory environment. The object to be studied is mounted in the test section of the wind tunnel as shown in Figure 3. 8. A fan or pump at one end of the tunnel creates a flow of air. Air flows into the tunnel through an inlet or settling chamber, accelerates through the nozzle, flows through the test section, and decelerates in the diffuser. The velocity of the air changes as it flows into sections of the tunnel with different cross-sectional areas as required by the continuity equation. The pressure of the air changes with changing velocity in accordance with Bernouilli's equation. Of course, the velocities and pressures predicted by these equations will only be correct if the assumptions made in deriving them are satisfied. For wind tunnels which operate at maximum test section velocities below 100 m/s or 330 ft/s (so the incompressible assumption is valid), these predictions are reasonably accurate. Figure 3. 8. Low-Speed Wind Tunnel Schematic The velocity of the air in a wind tunnel's test section is usually measured either by a Pitot tube placed in the test section or by two static ports, one in the settling chamber and one in the test section. The second method has the advantage that static ports do not intrude into the test section and therefore are less likely to interfere with the mounting of a model to be tested. Assuming incompressible flow, (3. 3) can be solved for V1 to yield: (3. 16) Substituting (3. 16) for V1 in (3. 4) and rearranging to collect like terms yields: which can be solved for V2 to yield: (3. 17) Since the required measurement is a differential pressure, the two static ports may be connected to the two sides of a manometer to create a test section velocity indicator. Example 3. 5 A low-speed wind tunnel similar to the one shown in Figure 3. 8 has a settling chamber cross-sectional area of 10 m2 and a test section cross-sectional area of 1 m2. When the wind tunnel is run at its maximum velocity in standard sea level conditions, a manometer connected between static ports in the walls of the settling chamber and the test section as shown in Figure 3. 8 has a difference in the heights of its fluid columns of 50 cm. What is the maximum test section velocity and the mass flow rate through the test section for this tunnel and these conditions? Solution: The manometry equation is used to determine the static pressure difference between the settling chamber and the test section. Since the velocity in the test section must be higher than the velocity in the settling chamber, the pressure in the test section will be lower and the height of the manometer fluid column which is connected to the test section will be higher: Once the pressure difference is known and the air density is obtained from the standard atmosphere table, (3. 17) may be used to determine the test section velocity: Since the test section velocity is below 100 m/s and the settling chamber velocity must be even slower, the assumption of incompressible flow is confirmed as valid and the analysis may proceed. The density in the test section is therefore approximately the standard sea level density, and (3. 1) may be used to predict the mass flow rate: Airfoils The continuity equation and Bernouilli's equation may also be used to explain how airfoils generate lift. Consider the steady, inviscid, incompressible flow of air past an airfoil as shown in Figure 3. 9. Figure 3. 9. Flow Past an Airfoil The entire flowfield is not shown in Figure 3. 9, only two stream tubes; one which passes above the airfoil and one passing below it. At Station 1, which is far upstream of the airfoil, the flow is one-dimensional. As the flow moves downstream, the orientation of the airfoil causes more of an obstruction to the flow above it than it does to the flow below it. This obstruction to the flow causes the stream tube above the airfoil to be constricted. The stream tube below the airfoil, on the other hand keeps a nearly constant cross sectional area all along its length, and in fact expands slightly as it approaches the underside of the airfoil leading edge. The continuity equation requires that the flow in the upper stream tube must accelerate to get past the airfoil while the flow in the lower stream tube does not and may even decelerate. Because the flow is one-dimensional far upstream of the airfoil, the same flow conditions, and therefore the same total pressure, will exist on every streamline at Station 1. We have made the appropriate assumptions so that Bernouilli's equation will apply along each streamline. Therefore, total pressure will be the same everywhere in the flowfield. Since, to satisfy continuity, the air will be moving faster at 2a than at 2b, the static pressure will be lower at 2a than at 2b. This pressure difference produces lift. Pressure, Shear, Lift, and Drag There are only two ways in which a fluid can impart forces to a body immersed in it. The first way, as just described, is by exerting pressure perpendicular to the body’s surface. If the pressures on opposite sides of a body are not equal, then a net force such as lift is exerted on the body. A portion of the drag on a moving body likewise results from pressure imbalances, but a significant portion also results from shear stresses exerted parallel to the body surface due to the viscosity (resistance to flowing) of the fluid. In reality, lift and drag are components of a total aerodynamic force on the body which is a sum of the net force due to pressure imbalances and the net force due to shear stresses. We have arbitrarily chosen to define lift as that component of the total aerodynamic force which is perpendicular to the free stream velocity direction and drag as that component which is parallel to the free stream. Figure 3. 10 illustrates pressure, shear stresses, lift, drag, and the total aerodynamic force on an airfoil. Figure 3. 10. Pressure, Shear, and Total Aerodynamic Force on an Airfoil Pressure and Lift A more detailed analysis of Figure 3. 9 gives further insight into the distribution of the pressure over the surface of the airfoil. If the continuity equation is applied at many points along the stream tubes in Figure 3. 9, a plot of velocity vs chordwise distance in each tube similar to Figure 3. 11 may be generated. Note that in Figure 3. 11, zero velocity is assumed to exist at the front and rear stagnation points on the airfoil, even though the stream tubes do not have infinite area at those points. This is possible because the stagnation points are on the side walls of the stream tubes. Applying Bernouilli’s Equation to these velocity plots yields plots of surface pressure distribution such as Figure 3. 12. Figure 3. 11 Velocity Distributions in Stream Tubes Above and Below Airfoil Figure 3. 12 Typical Airfoil Surface Pressure Distribution Note that Figure 3. 12 is for an airfoil with a chord length of 1 meter. If the airfoil span is also 1 meter, then since the pressure distributions are plotted vs chordwise location, the area between the upper and lower surface pressure curves is the net force due to pressure perpendicular to the airfoil chord line, the normal force. Figure 3. 13 shows the relationship between normal force and lift. The angle between the chord line and the free stream direction is called angle of attack, and is given the symbol a . Figure 3. 13 Normal Force and Lift on an Airfoil Figure 3. 14 shows the pressure distribution as arrows drawn perpendicular to the surface of the airfoil. Arrows drawn outward from the surface indicate pressures lower than free stream static pressure, while arrows drawn in toward the surface indicate pressures higher than free stream static. Figure 3. 14 Surface Pressures on an Airfoil The net normal force on a portion of the airfoil surface is the pressure on that portion multiplied by its area. Because the airfoil surface is not, in general, parallel to the chord line, then if ds is the length of an infinitesimally small portion of the surface and dx is the length of the component of ds along the chord line (see Figure 3. 15), the contribution of its surface normal force to the total force normal to the chord line for an airfoil of unit span is: (3. 18) Figure 3. 15 The Component Normal to the Chord Line of the Force Due to Surface Pressure So the magnitude of the total normal force on the airfoil is: (3. 19) which is exactly the same as the area between the two pressure lines on Figure 3. 12. As shown in Figure 3. 13, the lift on the airfoil is the component of normal force perpendicular to the free stream velocity vector (plus a negligible component of the chordwise force on the airfoil which will be ignored): (3. 20) Figure 3. 14 shows an interesting situation which is commonly achieved by many airfoils. The very low pressures on the rounded leading edge of the airfoil produce a net force in the chordwise direction which is positive forward. This effect is known as leading-edge suction or leading-edge thrust. On airfoils which are fairly thick and have relatively large leading-edge radii, leading-edge suction frequently has a significant component opposite the drag direction for a range of useful angles of attack. This reduces the net drag on these airfoils, making it a very desirable feature. One of the advantages of the relatively thick airfoil used by the Fokker DVII in World War I over the thinner airfoils on fighters of the Allies was greater leading-edge suction and therefore less drag. 3. 4 VISCOUS FLOW Viscosity is the tendency for a fluid to resist having velocity discontinuities in it. Viscosity in a liquid results from strong intermolecular forces which resist the motion of molecules relative to each other. The intermolecular forces between faster-moving molecules and slower ones cause velocity differences to be quickly equalized in a viscous liquid. As a liquid heats up the individual molecules have more energy relative to the intermolecular forces, so the viscosity of the liquid decreases. In a gas, on the other hand, viscosity results from the diffusion of momentum. Since a gas is composed of free-moving molecules with relatively weak intermolecular forces, the excess velocity of a faster-moving portion of a flowing gas is spread to the slower portions by collisions between faster and slower molecules and by actual movement of the higher-energy molecules into the slower-moving regions. As a result, when a gas heats up, the average speed of its molecules increases, and the rate at which momentum diffuses does also. Hence, a gas becomes more viscous as its temperature increases. But aside from these differences, the actions of viscosity in gases and liquids are quite similar. Viscous effects are most important when a fluid is in contact with and moving relative to a solid body such as an aircraft. That portion of the fluid which is in direct contact with the solid body cannot move relative to it. This is due to the fact that on a molecular scale, even the smoothest polished surface is very rough and full of peaks and valleys. The sides of these peaks and valleys are barriers to the motion of the fluid molecules which are flowing next to the surface. The molecules strike these barriers and impart their excess momentum to the body, so that the fluid closest to the body must move at the same speed as the body. The exchange of momentum between the fluid and the body is the actual mechanism of viscous shear stress. Viscosity causes the velocities of fluid layers further from the body to also be reduced. This reduction in velocity decreases with increasing distance from the body. The Boundary Layer The region next to a body in which the flow velocities are less than the free stream velocity is know as the boundary layer. Figure 3. 16 shows a velocity profile for a typical boundary layer. The edge of the boundary layer is normally defined as the point where the velocity reaches 99% of the free stream velocity. Boundary layers on modern aircraft can be from a few millimeters to several meters thick. Table 3. 2 indicates typical values of boundary layer thickness for a variety of objects. Virtually all important viscous effects occur in the boundary layer. As a result, the rest of the flowfield can be treated as inviscid. This greatly simplifies the aerodynamic analysis task. Figure 3. 16 Boundary Layer Velocity Profile Table 3. 2 Typical Boundary Layer Thicknesses Object| Flowing Fluid| Flow Velocity| Order of Boundary Layer Thickness| Supersonic Fighter Aircraft Wing| air| 500 m/s| a few millimeters| Glider Wing with 1 m Chord Length| air| 20 m/s| a few centimeters| Ship 200 m Long| water| 10 m/s| 1 m| Smooth Ocean| air (wind)| 10 m/s| 30 m| Land| air (wind)| 10 m/s| 100 m| Skin Friction Drag Several viscous effects in the boundary layer are very important to the aircraft designer. The first is the production of viscous drag, which is also called skin friction drag. Skin friction drag typically comprises about 50% of the total drag on a commercial airliner at its cruise condition. Since drag must be overcome by thrust, reducing viscous drag will reduce the amount of thrust needed and hence the fuel burned. A designer has several methods for reducing viscous drag. One method is to reduce the surface area of the aircraft which is in contact with the air. This area is called the wetted area, a term borrowed from ship designers. Design engineers pay a great deal of attention to minimizing an aircraft’s wetted area while keeping enough internal volume so that everything which the airplane must carry will fit. A second method for minimizing skin friction drag is controlling the shape of the boundary layer profile. Figure 3. 17 shows the changes a boundary layer undergoes as it flows over a surface. The initial boundary layer which forms at the front or leading edge of the surface is very orderly, with all velocity vectors parallel and only the velocity magnitudes decreasing with proximity to the surface. This is known as a laminar boundary layer, because it is composed of orderly layers. As the flow moves further down the body, the orderly flow breaks down and transitions into a swirling, mixing flow known as a turbulent boundary layer. The turbulent boundary layer is thicker than the laminar boundary layer. Figure 3. 17 Boundary Layer Transition and Separation Figure 3. 18 compares the profiles of the turbulent and laminar boundary layers. Note that, though it is thicker for the same conditions than the laminar boundary layer, velocities in the turbulent boundary layer are higher closer to the surface. This is due to the fact that the swirling flow in the turbulent boundary layer allows large quantities of faster-moving air to travel en masse down close to the surface, a much more effective way of transferring momentum than diffusion in the orderly laminar boundary layer. Because the velocities in the turbulent boundary layer are higher close to the surface, more momentum is transferred to the body, hence more skin friction drag. Figure 3. 18 Laminar and Turbulent Boundary Layer Velocity Profiles The actual mathematical expression for the shear stress, t, is: (3. 21) where m is the fluid viscosity, and y is the direction perpendicular to the body surface. The rate of change of velocity with y distance, , is called the velocity gradient, and the subscript y= 0 indicates that the gradient of interest is the one at the body surface. The skin friction drag for a body is given by: where Df is the skin friction drag, dS is a differential surface area, and Swet is the total wetted area of the body. The skin friction drag is often expressed as a dimensionless coefficient, Cf , which is defined as: (3. 22) where is the free stream dynamic pressure. Equation (3. 21) shows the same difference between laminar and turbulent boundary layers in the shear stress they produce as was described above. Since the turbulent boundary layer profile has a higher velocity gradient at the body surface than the laminar boundary layer, it produces greater shear stress and hence more skin friction drag. Smooth body surfaces tend to delay transition from laminar to turbulent flow. If the pressure in the flow is gradually decreasing with distance along the surface (corresponding to a gradual increase in flow velocity outside the boundary layer,) this also tends to delay transition. The condition of decreasing pressure with distance is called a favorable pressure gradient, because such a pressure field will help the flow accelerate. Designers can achieve favorable pressure gradients over a large part of a body by placing the point of maximum thickness of the body as far aft (to the rear) as possible. Of course, a body must eventually end, and the part of the body downstream of the point of maximum thickness will necessarily have an adverse pressure gradient as the pressure returns from it's low values to freestream pressure. Figures 3. 12 and 3. 14 both show that on the upper surfaces of airfoils at moderate angles of attack, the region of adverse pressure gradient begins upstream of the point of airfoil maximum thickness. The sloping part of the surface in Figure 3. 17 represents a region of adverse pressure gradient. The flow around the body reaches its maximum speed as it passes the body's point of maximum obstruction to the flow. The adverse pressure gradient on the rear of the body is just enough to slow the flow back down to free stream velocity at the rear end of the body. The flow in the boundary layer has lost momentum compared to that outside the boundary layer. However, the boundary layer flow still faces the same adverse pressure gradient. Therefore, at some point prior to the trailing edge (rear) of the body, the flow in the boundary layer slows to a stop, and then reverses. Stagnant or reverse flow acts like an obstruction to the rest of the normal forward flow, so it must detour around the obstruction. Since the reverse boundary layer flow is next to the body surface, the detouring flow moves away from the body, a condition called separation or separated flow. Notice the third boundary layer profile, the one just downstream of the beginning of the sloped part of the surface. The velocities in the boundary layer close to the surface at this point are zero, but no reverse flow has started. The velocity gradient at the wall for this profile is also zero, so there is no skin friction drag. This condition signals the beginning of separation. However, for very controlled conditions, a carefully designed airfoil can maintain a zero-gradient velocity profile from its point of maximum thickness all the way to its trailing edge. Since the pressure on the rear of the airfoil is returning to free stream values, airfoil designers call this area the pressure recovery region. The zero velocity gradient, zero shear stress pressure recovery is called a Stratford recovery after B. S. Stratford, the first engineer to study such a phenomenon1. Pressure Drag The static pressure at the forward stagnation point on a body is free stream total pressure. There is an aft stagnation point on the body as well. For inviscid flow, the static pressure at the aft stagnation point would also be free stream total pressure, and there would be no net drag. When the flow in the boundary layer loses momentum, it also loses total pressure. The static pressure in the flow outside the boundary layer is transmitted to the boundary layer and through it to the body surface. Therefore, when the boundary layer separates, its pressure is generally less than or equal to free stream static pressure. This is always less than total pressure at the front stagnation point. The difference in pressures at the front and rear of the body produces a net force in the drag direction which is called pressure drag. This is also called drag due to separation. Pressure drag can be reduced by delaying separation. The turbulent boundary layer has higher velocities close to the wall and a more effective mechanism for replacing low momentum fluid with faster-moving molecules from outside the boundary layer. A turbulent boundary layer is therefore more resistant to separation, more able to maintain forward velocity for a longer distance against an adverse pressure gradient. Therefore, designers will sometimes use a bumpy surface near the front of a body in order to force boundary layer transition. The higher-energy turbulent boundary layer which results, although it has greater skin friction drag, will separate further aft on the body, reducing pressure drag. A golf ball is a good example of this design decision. The round shape of the golf ball results in very high adverse pressure gradients on the rear surfaces, compared to a more tapered, streamlined, rear section. The high adverse pressure gradient causes separation to occur very early, just aft of the point of maximum thickness, for a laminar boundary layer. This results in very high pressure drag. Figure 3. 19 illustrates how the bumpy surfaces of golf balls cause earlier transition to delay separation, reducing pressure drag and allowing the balls to fly farther. (a) Smooth Surface (b) Dimpled Surface Figure 3. 19 Effect of Dimpled Surface on Separation Point on Golf Balls Reynolds Number Separation on a smooth golf ball occurs so early partly because the momentum of the air flowing past the ball is relatively low compared with the viscous shear which tends to slow it down. A non-dimensional parameter called the Reynolds number is used as a measure of these relative magnitudes of momentum and viscous forces. It is named for Osborne Reynolds, a pioneer researcher in viscous flow phenomena. The parameter is given the symbol Re and defined as: (3. 23) where x is a characteristic reference length or distance (such as the chord length of a wing or the distance from the leading edge of a surface to a particular point in a boundary layer) which describes a particular body or surface. The terms in the numerator of the expression for Reynolds number indicate the magnitude of the momentum of the flow, while viscosity in the denominator is a measure of the viscous forces present. Research has shown that the characteristics of a boundary layer can be described as functions of Reynolds number. This means that two bodies with the same shape and orientation to the flow, but with different sizes and in different flow conditions will have the same type and shape of boundary layer profile and the same transition and separation characteristics if they have the same Reynolds number. This type of relationship is called a similarity rule. It provides an important basis for wind tunnel testing, since the flowfields around small wind tunnel models will match those around large aircraft if the Reynolds numbers and other relevant similarity parameters are matched. Wind tunnel testing of this sort inspired and proved design concepts such as the Stratford pressure recovery. The critical Reynolds number is used to predict transition. Critical Reynolds number is defined using the distance from the start of a boundary layer as the reference length. When a distance (e. g. x coordinate) rather than a characteristic length (e. g. chord length) is used to define a Reynolds number, it is sometimes referred to as a local Reynolds number. To see how critical Reynolds number is used, consider the boundary layer for air flowing over a flat plate, similar to the left half of the surface in Figure 3. 17. The critical Reynolds number for such a body might be around 500, 000, depending on the surface roughness. If the flow velocity and density are high and the viscosity is low, critical Reynolds number will be reached and transition will occur only a short distance from the start of the boundary layer. On the other hand, if the flow is slow-moving, more viscous, and less dense, it will take a much larger value of the distance from the start of the boundary layer before local Reynolds number equals the critical Reynolds number. Look again at the equation defining the Reynolds number to see why this is so. In this second case, the boundary layer will remain laminar much further along the surface. This will have a profound effect on drag and separation characteristics of the boundary layer. This is one of the primary reasons why engineers conducting wind tunnel tests attempt to match Reynolds numbers with the full-scale flight conditions they are modeling. Laminar boundary layers cover only approximately the first 5-15% of a typical aircraft's wing. Example 3. 6 An airfoil in a wind tunnel test section has a critical Reynolds number of 600, 000. If the wind tunnel is operating in standard sea level conditions with a test section velocity of 90 m/s, how far aft of the airfoil's leading edge will transition occur? Solution: Solving (3. 23) for x (in this case xtransition) and substituting in the test section velocity and standard sea level values of r and m obtained from the standard atmosphere table: 3. 5 AIRFOIL CHARACTERISTICS Shape The differences in velocities and pressures which produce aerodynamic forces on an airfoil, and also its boundary layer profiles, transition, and separation characteristics are caused by the airfoil's shape and orientation. Aircraft designers spend a great deal of effort finding just the right shape for the airfoils they use on a particular design. Currently, many of these airfoil shapes are generated and optimized by computer programs. However, for many applications, airfoil shapes may be chosen from geometry and performance data published by airfoil designers. Airfoils of this sort are often grouped into families of similar shapes, distinguished from each other by gradual variation of one or more of the parameters which describe their shape. Figure 3. 20 illustrates a typical airfoil shape and the parameters which describe it. Figure 3. 20 Airfoil Shape Parameters The chord line shown in Figure 3. 20 is defined as a straight line drawn from the airfoil's leading edge to its trailing edge. The length of this line is called the chord or chord length and is given the symbol c. A curved line drawn from the leading edge to the trailing edge so as to be midway or equidistant between the upper and lower surfaces of the airfoil is called the mean camber line. The maximum distance between the airfoil's chord line and mean camber line is called the airfoil's maximum camber or just camber. An airfoil whose lower surface is a mirror image of its upper surface is said to be symmetrical or uncambered, and its mean camber line is coincident with its chord line. The airfoil is described by a thickness envelope wrapped around the mean chamber line. Thickness envelope is usually described by parameters which include the maximum thickness as a fraction of the chord length, the point where this maximum thickness occurs, and the leading edge radius. Lift and Drag Coefficients The lift and drag generated by an airfoil are usually measured in a wind tunnel and published as coefficients which are dimensionless. Lift and drag coefficients are defined as follows: (3. 24) (3. 25) where l and d are the lift and drag measured on the airfoil and S is the airfoil's planform area. Planform area is the area of a projection of the airfoil's shape onto a horizontal surface beneath it, similar to the airfoil's shadow when the sun is directly overhead. Now, we originally defined the airfoil as a slice of a wing, and as such it would have no planform area. When airfoils are tested in a wind tunnel, a section of wing is used which is frequently long enough to reach from one side of the test section to the other, as illustrated in Figure 3. 21. Figure 3. 21 Three-View Drawing of Rectangular Wing Section in a Wind Tunnel Test Section The length of the section of the wing, i. e. the distance which it must reach across the test section, is called its span. The wing has the same airfoil shape and size everywhere along its span, so that the same amount of lift and drag per unit span will be generated by any slice of the wing. A wing section such as this has a finite rectangular planform area which is used in defining the airfoil lift and drag coefficients. The flow around such a wing section is said to be two-dimensional, since flow properties vary in the streamwise (x) and vertical (y) directions, but not in the z or spanwise direction. Airfoil lift and drag coefficients are said to be two-dimensional coefficients. Angle of Attack Figure 3. 22 shows streamlines around an airfoil as its angle of attack is changed. In the first drawing, the airfoil is at zero angle of attack. Since the airfoil is symmetrical, the flowfield above it is a mirror image of the flowfield below it, so no net lift is produced. Note that as angle of attack increases the stream tubes above the airfoil become more constricted, so the velocities above the airfoil must increase. This will produce lower static pressure there, and consequently more lift. The lower static pressure above the middle of the airfoil will also produce a stronger adverse pressure gradient on the rear portion of the airfoil's upper surface. Note that the second drawing shows flow separation on the airfoil upper surface just ahead of the trailing edge. In the third drawing, the point of separation has moved upstream, due to the stronger adverse pressure gradient. (a) (b) (c) Figure 3. 22 A Symmetrical Airfoil at Three Angles of Attack Lift and Drag Coefficient Curves Figure 3. 23 shows plots of lift coefficient and drag coefficient as functions of angle of attack for the airfoil shown in Figure 3. 22. The letters in parentheses on the lift coefficient curve correspond to the letters in Figure 3. 22. Note that for smaller angles of attack, the lift coefficient increases linearly and drag changes very gradually with increasing angle of attack. The rate of change of lift coefficient with angle of attack on this part of the curve is called the lift curve slope: (3. 26) At higher angles of attack, the point of separation on the upper surface of the airfoil moves forward so far that it spoils some of the extra lift created by the additional constriction of the stream tubes. This causes the lift coefficient to increase more slowly with angle of attack and eventually reach a maximum. The earlier flow separation also produces more pressure drag. This causes the drag coefficient to increase much more rapidly at higher angles of attack. At the point on the lift curve where maximum lift coefficient is reached, further increases in angle of attack result in less lift. This phenomenon is called stall, and the angle of attack for maximum lift coefficient is called the stall angle of attack, or astall. Figure 3. 23 Symmetrical Airfoil Lift and Drag Coefficient Curves Cambered Airfoils Figure 3. 24 shows the flowfield around a cambered airfoil for an angle of attack of zero. Notice that even though the airfoil is not inclined relative to the free stream direction (a = 0), its shape causes the stream tubes above the airfoil to be more constricted than those below. This, of course, causes faster flow velocities and lower pressures above the airfoil. As a result, a cambered airfoil produces lift at zero angle of attack. As angle of attack increases, it has the same effect as for a symmetrical airfoil. However, since lift was already being generated at zero angle of attack, the cambered airfoil's lift curve remains above the symmetrical airfoil's curve. Adverse pressure gradients and flow separation also develop sooner for the cambered airfoil, so its stall angle of attack is lower. Figure 3. 24 Cambered Airfoil at Zero Angle of Attack Figure 3. 25 shows lift and drag coefficient curves for a cambered airfoil and a symmetrical one. Note that cla is approximately the same for both airfoils. Also note that clmax is higher for the cambered airfoil, even though it occurs at a lower angle of attack. The angle of attack for which the cambered airfoil generates zero lift is negative. It is called the zero-lift angle of attack and is given the symbol al= 0. The drag coefficient curves of Figure 3. 20 are plotted against lift coefficient instead of angle of attack in order to facilitate the comparison. Note that, unlike the symmetrical airfoil, the cambered airfoil has its minimum drag at a non-zero value of cl. Figure 3. 25 Lift and Drag Coefficient Curves for Cambered and Symmetrical Airfoils Moment Coefficient and Aerodynamic Center The distribution of pressure and shear stresses around an airfoil often produces net lift and drag forces, and it may also produce a net torque or moment. This is referred to as pitching moment and is given the symbol m. Pitching moment tends to rotate the nose or leading edge of the airfoil either up or down. A nose-up pitching moment is normally defined as positive. A pitching moment coefficient, cm, is defined as follows: (3. 27) where c is the airfoil chord length. Note that the equation defining cm differs from those for cl and cd in having an additional variable, the chord length, in the denominator. This extra quantity in the denominator is required to make cm dimensionless, since moment has dimensions of force times distance. The magnitude and sense of the moment generated by the airfoil will be different depending on what point is chosen as the moment reference center. In most cases, it is possible to choose a moment reference center for which the moment is zero. Such a point is called the center of pressure. The center of pressure is not very useful, however, because its location must shift with changes in angle of attack in order to keep the moment zero. A more useful moment reference center is the aerodynamic center. The aerodynamic center is a fixed moment reference center on the airfoil for which the moment does not vary with changes in angle of attack, at least for that range of angles of attack where the lift curve is linear. Figure 3. 26 shows the variation with cl of cm for a single airfoil using three different moment reference centers. Note that when summing moments about the aerodynamic center, the value of cm is not zero for cambered airfoils, but it remains constant for most of the range of lift coefficients. Figure 3. 26 Variation of Cambered Airfoil Pitching Moment Coefficient with Lift Coefficient for Three Choices of Moment Reference Center Reynolds Number Effects Figure 3. 27 shows lift and drag coefficient curves for an airfoil at two different Reynolds numbers. As Reynolds number increases, transition from a laminar to a turbulent boundary layer occurs closer to the leading edge of the airfoil. This causes more skin friction drag, but delays separation and reduces pressure drag. At lower angles of attack this change in the relative magnitudes of skin friction and pressure drag may result in either higher or lower total drag at higher Reynolds numbers. At higher angles of attack, where separation and pressure drag dominate, the reduction in pressure drag due to delayed separation generally results in less total drag at higher Reynolds numbers. Figure 3. 27 shows an airfoil that for higher Reynolds numbers has almost the same drag at low angles of attack, but less drag at higher a's and a higher astall. Figure 3. 27 Airfoil Lift and Drag Coefficient Curves for Two different Reynolds Numbers Reading Airfoil Data Charts Figure 3. 28 shows a typical page of wind tunnel airfoil data charts. Data such as these are published in a variety of books2, 3 and technical papers4, 5, 6. Appendix B in this book contains several similar data pages. Reading one of these charts is easy, if you pay attention to the details. First, note the airfoil designation at the bottom of the chart. NACA is the acronym for the National Advisory Committee for Aeronautics, a US Government agency, forerunner of NASA, which performed many wind tunnel tests of airfoils and other shapes in the 1930’s and 40’s. The 4-digit code identifies the particular airfoil shape. NACA used a series of codes with 4, 5, and more digits to systematically classify the airfoils they tested. For instance, the first digit in the 4-digit series identifies the airfoil's maximum camber in per cent of the chord. The second digit indicates where on the airfoil the point of maximum chamber occurs in tenths of the chord length aft of the airfoil leading edge. The third and fourth digits indicate the airfoil's maximum thickness in percent of the chord length. Thus, a NACA 2412 airfoil has 2% camber, its point of maximum chamber is located at its 40% chord point, and its maximum thickness is 12% of its chord length. If an airfoil with a NACA 2412 section had a chord length of 4 m, its maximum thickness would be 48 cm. See Reference 2 for more details of the various NACA airfoil series and designations. A drawing of the airfoil is on the right half of Figure 3. 28. The airfoil section lift coefficient vs angle of attack curves are on the left half. Curves for drag coefficient and the coefficient of pitching moment about the aerodynamic center are plotted against lift coefficient on the right half of the figure. A legend on the right half identifies curves for three different Reynolds numbers. The curves for standard roughness are for airfoils which have a surface texture like sand paper on their leading edges. Generally, the data for smooth airfoils (not standard roughness) for an appropriate Reynolds number are used when designing an aircraft. NACA 2412 Figure 3. 28 Lift, Drag, and Moment Coefficient Data for a NACA 2412 Airfoil Example 3. 7 A NACA 2412 airfoil with a 2 m chord and 5 m span is being tested in a wind tunnel at standard sea level conditions and a test section velocity of 42 m/s at an angle of attack of 8 degrees. What is the airfoil's maximum thickness, maximum camber, location of maximum camber, and zero-lift angle of attack? Also, how much lift, drag, and pitching moment about its aerodynamic center is the airfoil generating? Solution: Airfoil maximum thickness, camber, and location of maximum camber depend only on the NACA 2412 airfoil shape and the length of the airfoil chord. The first digit of the 2412 designation specifies a maximum camber which is 2% of the 2 m chord = 0. 04 m. The second digit indicates that the location of the point of maximum chamber is 0. 4 c = 0. 8 m aft of the leading edge. The last two digits specify a 12% thick airfoil, so the maximum thickness is: tmax = 0. 12 . 2 m = 0. 24 m The aerodynamic properties of the airfoil may depend on the Reynolds number, which for standard sea level conditions and a test section velocity of 42 m/s is: so the airfoil data curves for Re = 5. 7 million (not standard roughness) will be used. The values of aL= 0 and the cl at a = 8o do not, in fact, vary with Reynolds number. Their values can be read from Figure 3. 28 as: aL= 0 = - 2o, at a = 8o, cl = 1. 05 Also from Figure 3. 28, for cl = 1. 05 and Re = 5. 7 million: cd = 0. 0098 and = -0. 05 The dynamic pressure for the test is: The airfoil's planform area is it's chord multiplied by its span: S = b . c = 5 m . 2 m = 10 m2 The lift, drag, and moment about the aerodynamic center are then given by: l = cl q S = 1. 05 (1, 080 N/m2) (10 m2) = 11, 340 N d = cd q S = 0. 0098 (1, 080 N/m2) (10 m2) = 105. 8 N ma. c. = q S c = -0. 05 (1, 080 N/m2) (10 m2) (2 m) = -1, 080 N m Compressibility Effects The lift curve and drag data in charts like Figure 3. 28 are valid for relatively low speeds. At higher speeds, the large pressure changes which the air undergoes as it flows around an airfoil cause significant changes in the air density. These density changes in turn magnify the effects of the pressure differences which produce lift and pressure drag. These changes in the magnitudes of the lift and drag are called compressibility effects, since they result from the fact that the air's density is changing. Mach Number Understanding and predicting compressibility effects requires working with a flow parameter called Mach number, M. Mach number is named for the Austrian scientist and philosopher Earnst Mach, the first person to point out the significance of this parameter. It is defined as the ratio of the flow velocity to the speed of sound in the air. Free stream Mach number, , is the ratio of the aircraft's flight speed (and therefore the magnitude of the free stream velocity) to the speed of sound: (3. 28) The speed of sound is represented by the symbol a. Its value is given by the expression: (3. 29) where g = cp /cv is the ratio of specific heats (see Reference 7 for more details). For air, g = 1. 4. Understanding why the speed of sound should depend on temperature and no other flow properties is useful in understanding other Mach number effects. The explanation draws on the discussion in Chapter 1 of the origins of pressure and temperature in the random motions of molecules. The phenomenon called sound is actually fluctuations in air pressure which move through the air much like waves on the surface of a pond. As described in Chapter 1, air pressure has its origins in the collisions of air molecules which transfer momentum from the moving molecules to a body or to other air molecules. A sharp rise in pressure which moves as a wave through the air is really a surge in the momentum of the molecules which is transmitted from one part of the air mass to another through a series of collisions. The rate at which the momentum surge can move through the air (in other words, the speed of a sound wave) is limited primarily by the average speed of the molecules between collisions. But recall that temperature is a measure of average molecular kinetic energy, which depends on the average speed of the molecules. So temperature measures average molecule speed, and average molecule speed determines the speed at which sound can be transmitted. Prandtl-Glauert Correction

Corrections to airfoil lift coefficient data to account for compressibility effects are made using an expression known as the Prandtl-Glauert correction: = (3. 30) where is the lift coefficient read from the airfoil data chart (assuming airfoil data is from a low- speed test), is the airfoil lift coefficient corrected for compressibility, and is the flight Mach number for the conditions to which the airfoil data is being corrected. Note that (3. 30) is valid only for < 0. 7 or so. Also, the correction made by (3. 30) becomes trivial for < 0. 3. Also note that since the Prandtl-Glauert correction applies to all lift coefficients on the lift curve, the lift curve slope can also be corrected: (3. 31) Example 3. 8 A NACA 2412 airfoil with a 0. 5 m chord and 2 m span is being tested in a wind tunnel at standard sea level conditions and a test section velocity of 168 m/s at an angle of attack of 8 degrees. What is the airfoil's lift coefficient curve slope and how much lift is it generating? Solution: The aerodynamic properties of the airfoil may depend on the Reynolds number, which for standard sea level conditions and a test section velocity of 168 m/s is: so the airfoil data curves for Re = 5. 7 million (not standard roughness) will be used. As in Example 3. 7 , the values of aL= 0 and the cl at a = 8 degrees do not vary with Reynolds number. Their values can be read from Figure 3. 28 as: aL= 0 = - 2o, and cl = 1. 05 at a = 8o Since the lift coefficient curve appears linear between aL= 0 = - 2o and a = 8o, the lift curve slope may be estimated as the change in lift coefficient divided by the change in angle of attack: The test section velocity is greater than 100 m/s for this test, so compressibility corrections must be made. The Mach number for the test is calculated by substituting test section velocit