Surface pressure measurements on an aerofoil

DEN 302 Applied Aerodynamics SURFACE PRESSURE MEASUREMENTS ON AN AEROFOIL IN TRANSONIC FLOW Abstract The objective of this exercise is to measure the pressure distribution across the surface on an aerofoil in a wind tunnel. The aerofoil is tested under several different Mach numbers from subsonic to supercritical. The purpose of measuring the pressure distributions is to assess the validity of the Prandtl-Glauert law and to discuss the changing chracteristics of the flow as the Mach number increases from subsonic to transonic.

As a result of the experiment and computation of data, the aerofoil was found to have a critical Mach number of M= 0. 732. Below this freestream Mach number the Prandtl-Glauert law predicted results very successfully. However, above this value, the law completely breaks down. This was found to be the result of local regions of supersonic flow and local shockwaves. Contents

Apparatus 1. Induction Wind Tunnel with Transonic Test Section The tunnel used in this experiment has a transonic test section with liners, which, after the contraction, remain nominally parallel bar a slight divergence to accommodate for boundary layer growth on the walls of the test section. The liners on the top and bottom are ventilated with longitudinal slots backed by plenum chambers to reduce interference and blockage as the Mach number increase to transonic speeds. The working section dimensions are 89mm(width)*178mm(height). The stagnation pressure , p0? is close to the atmospheric pressure of the lab and with only a small error , is taken to be equal to the settling chamber pressure. The reference staticpressure, p? , is measured via a pressure tapping in the floor of the working section, well upstream of the model so as to reduce the disturbance due to the model. The ‘ freestream’ Mach number, M? , can be calculated by the ratio of static to stagnation pressure. The tunnel airspeed is controlled by varying the pressure of the injected air, with the highest Mach number that can be achieved by the tunnel being 0. 88. 2. Aerofoil model

The model used is untapered and unswept, having the NACA 0012 symmetric section. The model chord length, c, is 90mm and the model has a maximum chord/thickness ratio of 12%. Non-dimensionalised co-ordinates of the aerofoil model are given in table 1 below. Pressure tappings, 1-8 , are placed along the upper surface of the model at the positions detailed in table 1. An additional tapping, 3a, is placed on the lower surface of the aerofoil at the same chordwise position as tapping 3. The reason for including the tapping on the lower surface is so that the model can be set at zero incidence by equalizing the pressures at 3 and 3a 3.

Mercury manometer A multitube mercury manometer is used to record the measurements from the tappings on the surface of the model. The manometer has a ‘ locking’ mechanism which allows the mercury levels to be ‘ frozen’ so that readings can be taken after the flow has stopped. This is useful as the wind tunnel is noisy. The slope of the manometer is 45 degrees. Procedure The atmospheric pressure is first recorded, pat, in inches of mercury. For a range of injected pressures, Pj, from 20 to 120Psi, the manometer readings are recorded for stagnation pressure (I0? , reference static pressure (I?), and surface pressure form tappings on the model (In, for n= 1-8 and 3a). Theory These equations are used in order to interpret and discuss the raw results achieved from the experiment. To convert a reading, I, from the mercury manometer into an absolute pressure, p, the following is used: p= pat±l-latsin? (1) For isentropic flow of a perfect gas with ? = 1. 4, the freestream Mach number, M? , is related to the ratio between the static and stagnation pressures by the equation: M? = 2? -1p? p0? -? -1? -1. 0(2) Pressure coefficient, Cp , is given by:

Cp= p-p? 12?? U? 2(3) For compressible flow this can be rewritten as: Cp= 2? M? 2pp? -1(4) The Prandtl-Glauert law states that the pressure coefficient, CPe, at a point on an aerofoil in compressible, sub-critical flow is related to the pressure coefficient, CPi, at the same point in in incompressible flow by the equation: CPe= CPi1-M? 2(5) Due to its basis in on thin aerofoil theory, this equation does not provide an exact solution. However it is deemed reasonably accurate for cases such as this in which thin aerofoils are tested at small incidence.

The law does not hold in super-critical flow when local regions of supersonic flow and shockwaves appear. The value of the critical pressure coefficient, Cp*, according to local sonic conditions is calculated by: Cp*= 10. 7M? 25+M? 263. 5-1for? = 7/5(6) The co-ordinates for the NACA 0012 section are as follows: Figure 1-Co-ordinates for aerofoil (Motallebi, 2012) Results Given atmospheric conditions of: Patm= 30. 65 in-Hg Tatm= 21°C The following results were achieved: Figure 2-Pressure coefficient vs x/c for M= 0. 83566 Figure 3-Pressure coefficient vs x/c for M= 0. 3119 Figure 4-Pressure coefficient vs x/c for M= 0. 79367 Figure 5-Pressure coefficient vs x/c for M= 0. 71798 Figure 6-Pressure coefficient vs x/c for M= 0. 59547 Figure 7-

Pressure coefficient vs x/c for M= 0. 44456 Figure 8-Cp* and Cpminvs Mach Number From figure 7 the critical Mach number is able to be determined. The critical Mach number (the maximum velocity than can be achieved before local shock conditions arise) occurs at the point where the curves for Cp* and Cpmin cross. From figure 7 we can see that this value is, M? = 0. 732. Discussion Transonic Flow

Transonic flow occurs when ‘ there is mixed sub and supersonic local flow in the same flow field. ’ (Mason, 2006) This generally occurs when free-stream Mach number is in the range of M= 0. 7-1. 2. The local region of supersonic flow is generally ‘ terminated’ by a normal shockwave resulting in the flow slowing down to subsonic speeds. Figure 8 below shows the typical progression of shockwaves as Mach number increases. At some critical Mach number (0. 72 in the case of Figure 8), the flow becomes sonic at a single point on the upper surface of the aerofoil.

This point is where the flow reaches its highest local velocity. As seen in the figure, increasing the Mach number further, results in the development of an area of supersonic flow. Increasing the Mach number further again then moves the shockwave toward the trailing edge of the aerofoil and a normal shockwave will develop on the lower surface of the aerofoil. As seen in figure 8, approaching very close to Mach 1, the shockwaves move to the trailing edge of the aerofoil. For M> 1, the flow behaves as expected for supersonic flow with a shockwave forming at the leading edge of the aerofoil.

Figure 9-Progression of shockwaves with increasing Mach number (H. H. Hurt, 1965) In normal subsonic flow, the drag is composed of 3 components-skin friction drag, pressure drag and induced drag. The drag in transonic is

markedly increased due to changes to the pressure distribution. This increased drag encountered at transonic Mach numbers is known as wave drag. The wave drag is attributed to the formation of local shockwaves and the general instability of the flow. This drag increases at what is known as the drag divergence number (Mason, 2006).

Once the transonic range is passed and true supersonic flow is achieved the drag decreases. Analysis From figure 7, the conclusion was reached that the critical Mach number was 0. 732. This means ultimately that in the experiment local shockwaves should be experienced somewhere along the aerofoil for Mach numbers M= 0. 83566, 0. 83119 and 0. 79367. According to transonic theory, these shockwaves should be moving further along the length of the aerofoil as the freestream Mach number increases. To determine the approximate position of the shockwaves it is useful to look again at equation (4).

Cp= 2? M? 2pp? -1 Assuming constant p? , as static pressure in the test section is assumed to be constant and constant free stream Mach number as well, equation (4) may be written as: Cp= const. pconst. -1 Normal shockwaves usually present themselves as discontinuous data, particularly in stagnation pressure where there is a large drop. To detect the rough position of the shockwave on the aerofoil surface it is useful to look at the detected pressure by the different tappings and scrutinize the –Cpvs x/c graph to see where the drop in pressure occurs.

Investigating the graphs for the supercritical Mach numbers yields these approximate positions: M| x/c, %| 0. 835661| 40-60| 0. 831199| 35-55| 0. 793676| 25-45| Figure 10- Table showing approximate position of shockwave

According to the theory described earlier, these results are correct as it demonstrates the shockwave moving further along the aerofoil as the Mach number increases. As seen in figure 8, given a sufficiently high Mach number, a shock may also occur on the lower surface of the wing. This can be seen for M= 0. 835661, in figure 1, where there is a marked difference in pressure between tappings 3 and 3a.

The theoretical curves on each –Cpvs x/c graph were designed using the Prandtl-Glauert law. As mentioned earlier, this law is based on thin aerofoil theory, meaning it is not exact and there are sometimes large errors between the proposed theoretical values and the experimental values achieved. These large errors are seen most clearly in the higher Mach numbers. This is because in the transonic range, where there is a mixture of sub and supersonic flow, local shockwaves occur and the theoretical curves do not take shockwaves into account.

Hence, the theory breaks down when the freestream Mach number exceeds the critical Mach number for the aerofoil. At lower Mach numbers, the theoretical values line up reasonably well with those achieved through experiment. There only seems to be some error between the two, mainly arising in the 15-25% range. However, overall the Prandtl-Glauert law seems to be reasonably accurate as long as the Mach number remains sub-critical. The experiment itself was successful. The rough position of the shockwave and the critical Mach number were able to be identified.

There are however some sources of inaccuracy or error that can be addressed of the experiment is to be repeated for ‘ bettter’ results. Aside from the normal human errors made during experimentation the apparatus itself could be improved. Pressure tapping 1 (the closest to the leading edge) and pressure tapping 8 (the closest to the trailing edge) were placed at 6. 5% and 75% respectively. What this means is that they are not centralized relative to the leading and trailing edge effectively meaning it is not able to be determined whether or not the pressure is conserved.

At a zero angle of incidence, the pressure at the tip of the leading edge should be equal to the pressure at the tip of the trailing edge. To improve this pressure tappings should exist at the LE and TE and possibly more pressure tappings across the aerofoil surface to provide more points for recording. Another source of improvement could be using a larger test section so that there is absolutely no disturbance in measuring the static pressure. However, this may only produce a minute difference in the data and may not be worthwhile for such little gain. Conclusion

As desired, a symmetric aerofoil was tested in transonic flow and the experimental results were compared to the theoretical values predicted by the PrandtlGlauert law. In the cases where there was a large disparity between experimental and theoretical results, an explanation was given, relying on the theory behind transonic flow. Bibliography H. H. Hurt, J. (1965). Aerodynamics for Naval Aviators. Naval Air Systems Command. Mason. (2006). Transonic aerodynamics of airfoils and wings. Virginia Tech. Motallebi. (2012). Surface Pressure Measurements on an Aerofoil in Transonic Flow. London: Queen Mary University of London.