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Low Reynolds Number Laminar Separation Bubble Control Using a Backward Facing Step

Dominique Rothan

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**LOW REYNOLDS NUMBER LAMINAR SEPARATION BUBBLE CONTROL USING
A BACKWARD FACING STEP**

By

Dominique Rothan

**A Thesis Submitted to the
Office of Graduate Programs
in Partial Fulfillment of the Requirements for the Degree of
Master of Science in Aerospace Engineering**

**Embry-Riddle Aeronautical University
Daytona Beach, Florida
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Dominique Rothan

This thesis was prepared under the direction of the candidate's thesis committee chairman, Dr. Jose Rodriguez, Department of Aerospace Engineering, and has been approved by the members of his thesis committee. It was submitted to the Office of Graduate Programs and was accepted in partial fulfillment of the requirements for the degree of Master of Science in Aerospace Engineering.

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ABSTRACT

Author: Dominique Rothan
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The problem of a laminar separation bubble on the upper surface of a 9.5 % thick airfoil operated at low Reynolds number ($R_N=380,000$) was investigated experimentally.

An unmodified LRN-010 airfoil and a modified LRN-010 airfoil with a backward facing step on its upper surface were built and tested in the ERAU wind tunnel in order to obtain pressure distributions and drag for different angles of attack.

Results were found to be in good agreement with the results obtained from the XFOIL CFD code. It was concluded that the step cutout modification improves the airfoil's lift-to-drag ratio at low angles of attack by tripping the boundary layer and decreasing the size of the laminar separation bubble. For higher angles of attack, the step cutout degraded the performance of the airfoil.

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I. INTRODUCTION

Low Reynolds number aerodynamics has been the subject of recent interest due to its application to both military and civilian systems (Ref.1). In the last two decades we have seen the development of human powered flight, encouraged by the series of Kremer prizes (Ref.2), remotely piloted vehicles (RPV's) for military use, high altitude (60,000 ft) manned or remotely piloted research aircraft, and ultra-light aircraft for sports. All of these systems require airfoils that operate at Reynolds numbers ranging from 50,000 to 1,000,000. At these Reynolds numbers, the boundary layer on the surface of the airfoil is usually laminar and the airfoils suffer of laminar separation followed by a bubble or stall.

There have been many studies done on low Reynolds number airfoils and especially on laminar separation bubbles (Ref.1,3,4,5,6,7,8,9,10,11,12,13,14,15,16,17,18,19,20,21, 22). The main thrust of these studies has been to obtain practical design data, seek understanding of the fluid mechanics of the flow or test possible solution to the problem of laminar separation. In recent articles, Gad-el-Hak (Ref.12,21) discusses methods by which low Reynolds number airfoils can be improved. Our research concentrates

on a new method for separation bubble control by means of a backward facing step. The airfoil chosen for this project is the LRN-010 designed by W. Pfenninger (Ref.5). The airfoil's separation bubble makes it ideal for this study. Models with and without the backward facing step were built and tested in the ERAU closed circuit subsonic wind tunnel at Reynolds numbers in the 380,000 range. Two step sizes, 0.3 and 0.6 percent chord, were used. The XFOIL design/analysis code was used to obtain comparable numerical data. Pressure distribution, lift, moment and drag were obtained numerically and experimentally for tripped and untripped airfoils at various angles of attack. The result indicate that the lift to drag ratio can be improved by use of the step for certain angles of attack ranges.

II. BACKGROUND

A. Separation Bubble Mechanism

For a detail account of the separation bubble mechanism, see Ref.17,19,22. In the 10^4 to 10^6 Reynolds number range - considered as low Reynolds numbers - many complex phenomena take place within the upper surface boundary layer of an airfoil. The flow remains laminar on the portion where the velocity U increases or pressure decreases. As soon as the velocity starts decreasing - corresponding to the minimum of the pressure distribution over the airfoil - the flow has a strong tendency to separate. The airfoil's performance, in terms of lift to drag ratio, is determined by the poor resistance to separation of the laminar flows (Ref.12). A region in which the pressure is approximately constant forms after separation. This region is bounded by a free shear layer which is highly unstable. The transition from laminar to turbulent flow occurs in the free shear layer close to the separation point. Provided that the resulting turbulence spreads and reaches the surface, a laminar separation bubble forms (Ref.18). The subsequent flow is fully turbulent and is less likely to separate since the velocity - or energy - of the fluid elements close to the surface is larger.

Figure 1 illustrates a laminar separation bubble of length L_b and height h_b in relation to the velocity gradients and the boundary layer thickness. The topology of the bubble depends on many factors including the radius of curvature and smoothness of the wall, the free stream velocity gradient just outside the boundary layer, and the thickness of the boundary layer. Some bubbles will have transition immediately after separation, and others right before reattachment. The later forms thicker bubbles.

Fig.1: Separation Bubble Mechanism.

Two conditions are necessary for a laminar separation

to occur: an adverse pressure gradient of sufficient magnitude to cause separation, and flow conditions over the surface such that the boundary layer is laminar at the separation point (Ref.17). Reattachment occurs when the flow entrained by the turbulent shear layer is able to overcome the adverse pressure gradient. Inherent conditions for reattachment and the formation of the separation bubble are factors such as the Reynolds number, the surface curvature, the surface roughness, the freestream turbulence, the acoustic level, etc.

As can be seen in Fig.2 (from Ref.17), a separation bubble has a profound effect on the pressure distribution over the airfoil. As soon as the flow separates a plateau appears on the pressure distribution. This plateau indicates that the velocity is fairly constant within the bubble. The modified pressure distribution produces a greater integrated force in the flow direction and therefore an increase in the pressure drag (Ref.23).

Fig.2: Effect of Separation Bubble (Ref.17).

The length of the separation bubble can be estimated from the size of the plateau and is usually between 2 and 10 percent of the chord (Ref.19).

Parameters such as the Reynolds number, the angle of attack and the leading edge radius influence the size and location of the separation bubble. The Reynolds number greatly affects the length of the separation bubble, as indicated schematically in Fig.3 (Ref.19). As the angle of attack increases, the adverse pressure gradient increases and the separation bubble moves forward (Ref.17). The leading edge radius also affects the formation of laminar separation bubbles. A small radius induces a small bubble close to the leading edge. This usually leads to abrupt leading edge stall. A large radius produces a separation bubble in the aft part of the airfoil that leads to a more gradual trailing edge stall.

The pressure drag associated with the laminar separation bubble can be decreased by proper use of flow separation control as discussed in the following section.

B. Methods for Flow Separation Control

Flow separation can be controlled by active or passive methods. Active methods require an external energy source while passive methods do not. For a complete review of these methods see Ref.21. We limit the discussion to the passive methods closely related to the one studied in this

1. High R_N : transition occurs before separation can occur; the velocity distribution approximates the inviscid distribution.
2. Medium R_N : separation takes place prior to transition leading to a short separation bubble; the velocity distribution approximates the inviscid distribution outside the bubble region.
3. Low R_N : long bubble the velocity distribution is significantly affected.
4. Very low R_N : no bubble, the airfoil is stalled.

Fig.3: Effect of Reynolds Number on Bubble Length.

research.

1. Conditions for Attached Flow

For the flow in a boundary layer to remain attached, the curvature of the velocity profile evaluated at the wall $[\frac{\partial^2 u_1}{\partial x_2^2}]_0$, must be negative (Ref.12). Therefore, to investigate the requirements for attached flow, we examine the averaged streamwise momentum equation for a steady, two dimensional, turbulent, incompressible flow, evaluated at the wall,

$$\rho v_o \times [\frac{\partial \bar{u}_1}{\partial x_2}]_0 + \frac{dP_0}{dx_1} - \frac{d\mu}{dT} \times [\frac{\partial \bar{T}}{\partial x_2} \times \frac{\partial \bar{u}_1}{\partial x_2}]_0 + \rho [\frac{\partial \bar{u}_1 \bar{u}_2}{\partial x_2}]_0 = [\mu \frac{\partial^2 \bar{u}_1}{\partial x_2^2}]_0$$

where,

$[]_0$: flow quantities computed at the wall.

ρ : constant density.

μ : variable viscosity.

v_o : normal velocity of fluid injected through the surface, $v_o > 0$ for injection, $v_o < 0$ for suction.

$\bar{u}_1$: time averaged velocity in the streamwise direction.

$\bar{P}$: mean pressure.

$\bar{T}$: mean temperature field.

$\rho \bar{u}_1 \bar{u}_2$: tangential Reynolds stress.

x_1 : streamwise coordinate.

x_2 : normal coordinate.

The factors that contribute to keep $[\frac{\partial^2 \bar{u}_1}{\partial x_2^2}]_0 < 0$ are:

wall suction, $v_0 < 0$, favorable pressure gradient, $dp_0/dx_1 < 0$, wall cooling in gases, $d\mu/dT > 0$ and $[\frac{\partial T}{\partial x_2}] > 0$, heating in liquids, $d\mu/dT < 0$ and $[\frac{\partial T}{\partial x_2}] < 0$ or turbulence intensification, $[\frac{\partial \overline{u_1 u_2}}{\partial x_2}] < 0$. Suction and cooling are widely used active methods, while geometric shaping and tripping devices to induce turbulence are the most common passive methods.

2. Flow Separation Control Using Passive Methods

Passive methods are those that seek a solution of the problem of flow separation without the use of external (or artificial) energy. These methods use alterations of the body topology in order to induce a transfer of energy from the freestream flow to the fluid close to the wall. The reenergized flow close to the wall, which usually becomes turbulent, tends to remain attached.

Among passive methods are the geometrical modifications to advance the transition point, boundary layer fences to prevent separation, vortex generators to raise the turbulence level and enhance the momentum and energy of the boundary layer, and screens to divert the flow and increase the velocity gradient on the surface.

a. Geometrical Modifications. Tape strips (Ref.4) and longitudinal and transverse grooves (Ref.24,25) are among the geometrical modifications that have proved successful in

reducing the drag of airfoils and bluff-bodies. Laminar separation bubbles can be reduced in size or canceled by means of vortex generators (Ref.17). In our research, we investigate the effect of a small backward facing step located so as to control the separation bubble.

Backward facing steps are a fairly new way of controlling the separation. Grooves or step cutouts have a pumping action on the flow that tends to decrease the size of the flow separation.

b. Backward Facing Steps. The idea of adapting a backward facing step on the upper surface of an airfoil at low Reynolds number is to capture the separation bubble and decrease its size. The size of the bubble depends on how far aft of the step the flow reattaches.

Many parameters influence the reattachment length (surface curvature after the step, boundary layer thickness etc.) but it is usually a linear function of the step height. Therefore, the step cutout tested in this project is small enough to enable a prompt reattachment and high enough to capture the separation bubble. Figure 4 (Ref.26) shows how the step can be used to trap the separation bubble.

Fig.4: Backward Facing Step with Separated Flow (Ref.26).

Backward facing steps have been studied mainly as a means of developing understanding of the separation-reattachment phenomenon. The studies (Ref.26,27,28) have concentrated on backward facing steps that cause a sudden expansion in a channel type flow. The flow is governed by the Reynolds number based on the step height, $R_N = \frac{\rho V h}{\mu}$, the boundary layer thickness ahead of the expansion and the expansion ratio. In this study, the Reynolds number based on the step height ranged between 1,000 and 3,000.

C. Low Reynolds Number Airfoils.

Two different design philosophies have been adopted in order to optimize the characteristics of low Reynolds number airfoils. Both intend to prevent laminar separation, but they use different ways of obtaining the transition (Ref.4). The following sections describe how Liebeck and ASM-LRN airfoils differ in their flow separation control.

1. Liebeck Airfoils

The philosophy used by Liebeck is based on the Stratford pressure recovery of a turbulent boundary layer. The main feature of the airfoils is to maximize the lift, not to minimize the drag. Therefore, their maximum thickness tends to be forward of the 50% chord point. For the pressure to follow the Stratford recovery, transition must occur on a specific region, the transition ramp. If the transition is delayed, the pressure recovery fails and the flow separates, leading to stall or a separation bubble. These airfoils are very sensitive to off-design Reynolds numbers. Figure 5 from Ref.9 shows a typical pressure distribution over a Liebeck airfoil and the different flow regions.

Fig.5: High Lift Airfoil Pressure Distribution (Ref.9).

Notice the pronounced kink in the transition region and the step pressure recovery. The ramp is followed by a Stratford pressure recovery which avoids separation by a specified margin along its entire length (Ref.9). The pressure field is optimized to produce maximum lift under certain restrictions, i.e. no flow separation. An inverse calculation procedure is used to derive the airfoil shape.

2. ASM-LRN-010 airfoil

The ASM-LRN-010 airfoil - or more commonly LRN-010 - is one of the low Reynolds number airfoils designed by W. Pfenninger et al. (Ref.5). Figure 6 shows the LRN-010 airfoil. Coordinates are given in Appendix A.

Fig.6: LRN-010 Profile.

These airfoils minimize the size of the separation bubble by avoiding an excessively convex upper surface in the transition zone. This usually leads to thin airfoils with rounded pressure distributions, a small kink in the transition region and a mild pressure recovery (Ref.3).

However, the authors of the article presented in Ref.5 note: "unfortunately, the ... described measures to establish shallow and relatively harmless laminar separation bubbles and thereby minimize the associated airfoil pressure drag are usually paid by some loss in laminar flow and a correspondingly higher skin friction drag. This penalty may be avoided if it should prove possible to artificially enforce transition close to the start of laminar separation and thus avoid laminar separation bubble in the first place".

The present study is based on this idea, with a backward facing step on the upper surface as the main control mechanism.

The LRN-010 airfoil was chosen for this project. It is 9.5 percent thick and has a $(C_L/C_D)_{\max}$ of over a hundred at Reynolds numbers around 380,000 (Ref.8).

D. XFOIL Airfoil Design/Analysis Code

At low Reynolds numbers, the very strong and non linear coupling between the viscous, transition and inviscid formulations make the airfoil analysis extremely complex.

The XFOIL CFD code, developed by M. Drela of MIT (Ref.29), is capable of solving the viscous flow over airfoils with small separation bubbles.

The inviscid formulation uses a second order vortex and first order source panel method. Figure 7 shows the LRN-010

paneling (154 panels total).

Fig.7: LRN-010 Paneling.

For the viscous calculations the strength of the source elements are determined from the displacement thickness obtained from the solution of the integral boundary layer equations. The panel method and boundary layer equations are solved simultaneously with a global Newton integration method. To predict transition from laminar to turbulent flow, an estimate of the growth of Tollmien-Schlichting waves based on solutions of Orr-Sommerfeld equations and the Falkner-Skan velocity profiles is used. Once the amplitude of the amplified wave reaches a prescribed value, the flow is considered turbulent. The code is capable of solving flow fields with small separated regions, and predict lift, drag and pitching moment with excellent accuracy. One of the main features is the prediction of stall (Ref.30).

E. Pressure Coefficient Calculation

The pressure coefficient distribution over the airfoil was calculated from the manometer readings by means of:

$$Cp_i = \frac{P_i - P_\infty}{\frac{1}{2} \rho V_\infty^2} = \frac{P_i - P_\infty}{P_T - P_\infty}$$

with:

P_i : pressure at the i^{th} pressure tap

P_∞ : freestream pressure

V_∞ : freestream velocity

P_T : total pressure inside the test section

Also, the pressure is a linear function of the manometer reading:

$$p = \rho g h \Rightarrow Cp_i = \frac{h_i - h_\infty}{h_T - h_\infty}$$

with h_i , h_∞ , h_T , the i^{th} pressure tap, freestream and total manometer readings respectively.

F. Lift, Pitching Moment and Drag Calculation

This section gives the method used to calculate the lift, pitching moment and drag of the LRN-010 airfoil from the pressure distribution and momentum loss.

The force and moment coefficients are given as a function of the angle of attack α (Ref.23). Figure 8 illustrates the forces and moments acting on the airfoil:

Fig.8: Forces and Moments Acting on an Airfoil (Ref.23).

The lift and drag coefficients can be expressed in terms of the normal and axial force coefficients by,

$$C_l = C_n \cos \alpha - C_a \sin \alpha$$

$$C_d = C_n \sin \alpha + C_a \cos \alpha$$

where

$$C_n = \frac{1}{c} \left[\int_0^c (C_{p,l} - C_{p,u}) dx + \int_0^c \left(C_{f,u} \frac{dy_u}{dx} + C_{f,l} \frac{dy_l}{dx} \right) dx \right]$$

$$C_a = \frac{1}{c} \left[\int_0^c \left(C_{p,u} \frac{dy_u}{dx} - C_{p,l} \frac{dy_l}{dx} \right) dx + \int_0^c (C_{f,u} + C_{f,l}) dx \right]$$

$$C_{mLE} = \frac{1}{c^2} \left[\int_0^c (C_{p,u} - C_{p,l}) x dx - \int_0^c \left(C_{f,u} \frac{dy_u}{dx} + C_{f,l} \frac{dy_l}{dx} \right) x dx \right. \\ \left. + \int_0^c (C_{p,u} \frac{dy_u}{dx} + C_{f,u}) y_u dx + \int_0^c \left(-C_{p,l} \frac{dy_l}{dx} + C_{f,l} \right) y_l dx \right]$$

C_a : horizontal force coefficient

C_n : normal force coefficient

C_l : lift coefficient

C_d : drag coefficient

C_{mLE} : moment coefficient

c : chord length

C_p : pressure coefficient

C_f : friction coefficient

$$C_f = \frac{\tau}{1/2 \rho V_\infty^2}$$

Subscripts u and l designate the upper and lower surface respectively.

It is recognized that the skin friction has a small effect on the lift and moment coefficients. Therefore, the lift and moment coefficients can be written as:

$$C_l = \frac{1}{c} \left[\int_0^c (C_{p,l} - C_{p,u}) dx \right] \cos \alpha - \frac{1}{c} \left[\int_0^c \left(C_{p,u} \frac{dy_u}{dx} - C_{p,l} \frac{dy_l}{dx} \right) dx \right] \sin \alpha$$

$$C_{mLE} = \frac{1}{c^2} \left[\int_0^c (C_{p,u} - C_{p,l}) x dx + \int_0^c \left(C_{p,u} \frac{dy_u}{dx} - C_{p,l} \frac{dy_l}{dx} \right) y dx \right]$$

and hence, C_l and C_{mLE} can be computed from the pressure distribution obtained experimentally.

The total drag on an airfoil consists of skin friction drag and pressure drag. Since friction is an important factor in the computation of the total drag, the latter cannot be calculated from the pressure distribution. A practical way of obtaining this kind of measurement is to determine the profile drag from the velocity distribution in the wake (Ref.15). The total drag of an airfoil is given by the formula:

$$D = b \times \rho \times \int_{y=-\infty}^{y=+\infty} u (U_\infty - u) dy$$

where the integral is performed perpendicular to the freestream, downstream of the airfoil.

U_∞ : freestream velocity
 $u(y)$: velocity component in the wake
 y : direction normal to freestream
 b : airfoil span

This equation can be modified for the drag coefficient calculation of an airfoil:

$$C_d = \frac{1}{c \times q_{out}} \int_{wake} (q_{out} - q_{ins}) dy$$

or for a finite number of pitot tubes inside the wake:

$$C_d = \frac{\Delta y}{c \times q_{out}} \sum_{wake} (q_{out} - q_{ins})$$

with:

Δy : distance between two pitot tubes
 c : chord length
 q_{out} : dynamic pressure outside the wake
 q_{ins} : dynamic pressure inside the wake

G. The Wind Tunnel Turbulence Level

Turbulence level in a wind tunnel is related to the magnitude of the longitudinal and transverse fluctuations of velocity. It is a very important feature in wind tunnel experiments and, in particular, for low Reynolds number

testing. Turbulence in the freestream modifies the boundary layer (Ref.13), affects the stall histeresis of an airfoil (Ref.20) and advances the transition point. However, it has little effect on the linear part of the lift curve slope.

III. MODEL CONSTRUCTION

A. Generalities

The size of the models was based on the low Reynolds number and the wind tunnel setup requirements. The span length is 38.10 cm (15 in) and the chord length is 30 cm.

With a flow velocity of $V=18.5$ m/s inside the test section, the chord Reynolds number R_N is:

$$R_N = \left(\frac{\rho}{\mu} \right) \times V \times C$$

$$R_N = (6.85 \times 10^4) \times 18.5 \times 0.3$$

$$R_N = 380000$$

The models were made of styrofoam covered with fiberglass. This method allowed to obtain the desired profile with a good accuracy at relatively low effort.

B. Description of the Building Process

1. The Styrofoam Model

The process of cutting the LRN-010 airfoil into a blue styrofoam block using a hot wire is described in the following section.

Precise scaled plots were obtained from the coordinates in order to prepare 0.64 mm thick aluminum tin plates. The tin plates helped guiding the hot wire when cutting the

styrofoam. They had to be thick enough not to warp under the weight of the hot wire, and thin enough to not dissipate the heat of the hot wire. The precision of the contour was excellent. Scaled pictures of the tin plates is shown in Fig.9.

a: tin plate set for the upper surface cut.

b: tin plate set for the lower surface cut.

Fig.9 a,b: Scaled Pictures of the Tin Plates.

Two identical tin plates - defining the airfoil upper surface for example - were nailed on the sides of the styrofoam block. The upper surface was then obtained by

sliding the hot wire along the tin plates. Figure 10 gives an illustration of this process:

Fig.10: Cutting Process.

This process was repeated with the lower surface tin plates to finally obtain the complete airfoil made of styrofoam. Gravity was used to move the hot wire along the tin plates in order to obtain a constant cutting speed. Any speed variations lead to slight modifications of the airfoil's shape. This technique was mastered and gave excellent results in terms of accuracy with the exception of the leading edge. Due to the high curvature of the leading edge and the flexibility of the hot wire, a constant radius along the span could not be obtained. Therefore, the leading edge was removed and replaced by a balsa wood piece that was sanded to the right shape.

2. Fiberglass Cover

The purpose of the covering was to give the airfoil the required smoothness and stiffness.

Two 1.4mm thick Mylar sheets (38.10cm x 35.00cm each) were taped together. Wax was applied on them to prevent the epoxy to stick to them. A 5 Oz./ft² fiberglass cloth was placed to cover the two sheets and eight hours cure epoxy was evenly spread on the clothing. The process was repeated with a second layer of fiberglass and epoxy. The styrofoam airfoil was then placed on one of the Mylar sheet, trailing edge on the inside. The second Mylar sheets was then folded on the airfoil. Figure 11 illustrates the process. The amount of epoxy used was an important parameter that controlled the overall smoothness; too little epoxy lead to a rough surface and too much caused air bubbles to form during the drying step.

The Mylar sheet helped to give the desired smoothness. The covered airfoil was then placed in a vacuum bag, thus maintaining the Mylar sheets against the fiberglass with a 80kPa. pressure force. The Airfoil and Mylar sheets were removed after a period of about 12 hours. The edges of the model had to be sanded to remove the excess of dried epoxy.

Figure 11: Fiberglass Covering Process.

3. Pressure Taps

A total of 32 pressure taps were installed on the models: 23 on the upper surface and 9 on the lower surface. The upper surface was of more interest since the separation bubble occurs in this area.

The location of the pressure taps was determined with the help of the pressure distribution obtained from the ISES program (Ref.29) at $R_N=250,000$ and angle of attack $\alpha=5$ degrees. Pressure taps were concentrated where the pressure gradient was large and where the separation bubble formed. Therefore, they were not equidistant from each other. Also, taps were arranged at an angle of 20 degrees with respect to

the chord to minimize interference.

Tygon tubing of 2.4mm diameter was used to connect the pressure taps to the tubing outside the wind tunnel section. One of the main difficulties was to insert the Tygon tubes inside the model without damaging its shape. Two different methods were used. The first one consisted of opening the airfoil in the span-chord plane to access to the inside of the airfoil. The second one consisted of digging trenches to bury the tubing. These two methods are detailed in the following sections.

a. Airfoil Opening Method. The styrofoam was cut from the lower surface in order to keep the upper surface free of any defect that could affect the flow where the bubble forms. A drawing of the cut is shown in Fig.12:

Fig.12: Model Cut.

This allowed to easily place the tubes into the taps at the locations where the lower part had been removed, and also in the removed part itself which corresponds to the lower surface.

Copper tubes of 1.2 mm diameter were used to connect the Tygon tubing to the pressure taps drilled in the leading edge. The same copper tubes were also used at the trailing edge where the thickness became of the order of the tube diameter. In order to obtain pressure taps as far back as possible, the copper tubes were flattened.

A main metal spar was included inside the airfoil to give strength to the model and also to facilitate the mounting in the wind tunnel. Thus, the inside of the model had to be dug out for the main spar and also to allow the tubing to exit. This was done with a specially shaped hot wire and also by sanding the foam carefully. Figures 13 and 14 are pictures of the model with and without the bottom part. Figure 15 is a picture of the inside cutouts.

When cutting the airfoil in two parts, the foam melted along the plane on which the hot wire slit and therefore, a thin layer of foam was lost. In order to retain the original airfoil thickness, 1.6 mm thick aluminum plates were glued in the inside. This also gave further stiffness to the model.

Fig.13: Model with Removed Bottom Part.

Fig.14: Model with Replaced Bottom Part.

Fig.15: Details of the Inside of the Model.

Eight hours cure epoxy was used to fill the empty spaces inside the model, the 32 tubes were placed in the cutouts dug for that matter, the two main parts of the airfoil were assembled together and thin wood plates with the LRN-010 shape were glued on each tip of the model to prevent any leaking of epoxy.

The advantage of this method was that it left the upper surface - the surface of interest - untouched. However, a few problems made this method very delicate:

The fill-in epoxy had to be laid before assembling the two parts together. Therefore, the 32 tubes had to be placed at their assigned location while bringing the two

pieces together - each having tubes connected to them - and within the curing epoxy. This resulted in handling 32 tubes in an almost unreachable closed space inside a sticky environment, trying to avoid any leaking of epoxy.

Since this step was extremely delicate, the inside of the model could not be entirely filled with epoxy. Therefore, small deformations of the lower surface appeared when the airfoil was placed in the vacuum bag for fiberglass covering. The air inside the airfoil compressed, leading to undesirable cavities and lack of contour accuracy at some locations. Hence, the cavities were filled with Bondo after having very precisely located the pressure taps. Bondo was applied and sanded until the shape of the airfoil was restored. Then taps were drilled at the locations where the tubes were covered with Bondo. However, 3 pressure taps got definitely blocked - either by epoxy or by Bondo - and were not used.

b. Trenches Method. Before covering the airfoil with fiberglass two pieces of wood were inserted at each tip of the styrofoam model as shown by Fig.16. Also, two thin wood plates with the airfoil shape were glued at each end. The wood pieces were used to provide attachment areas for airfoil mounting. The model was then covered with fiberglass as described above.

Fig.16: Styrofoam Model with Wood Reinforcement

- Side and Upper Views.

The second step was to mark the location of the pressure taps on the surface and to dig cordwise trenches from each mark to the quarter chord line.

A main trench was dug in the middle along the span to centralize all the Tygon tubes. The paths followed by the trenches are shown in Fig.17. Digging was done with a Dremel tool.

Fig.17: Trenches Geometry - Lower Surface.

The fiberglass was cut and the underlying styrofoam smoothly sanded. Tygon tubes were glued in the trenches and the end left sticking out from the surface to prevent blockage from the Bondo during the fill-in phase.

The fill-in step consisted of covering the surface of the model around the trenches with transparent tape, thus avoiding alterations of the overall geometry. The trenches were filled with Bondo and the excess Bondo was sanded. The Tygon tubes were trimmed and the complete surface sanded with very thin paper smoothers.

The advantage of this method was to work on the clean fiberglass model which was not as fragile as the styrofoam model. Also, installing the tubing became very easy compared to the first method. However, this required the modification of the upper surface shape by digging and filling trenches. No matter how close the final result was, the shape lost some accuracy.

4. Remarks on the Cutout and Tubing

a. Step Cutout. The clean LRN-010 airfoil was made using the trenches method whereas the airfoil with the step used the cutting method. The step was made before covering the airfoil with fiberglass by sanding the styrofoam at $x/c = 0.528$ as shown by Fig.18.

Fig.18: Foam Model with Cutout.

After covering the step with fiberglass, its depth decreased to 2.0 mm. The size of the step was reduced to

1.0 mm by gluing a band of aluminum on the flat part of the step. Small holes were drilled through the plate to avoid blocking the pressure taps under it.

b. Tubing. Although the Tygon tubes had an outside diameter of 1.6 mm, it was difficult to bring the 32 tubes to the tip of the model: the model is 2.85cm thick and the tubing arranged in a circular shape has a diameter of 2.5cm. This does not include the space used by the main spar or the inserted wood pieces. Therefore, the tubes were placed in an elliptical shape around the spar or wood piece. The Tygon tubes were numbered depending on the corresponding pressure tap location on the airfoil.

IV. EXPERIMENTAL FACILITY AND MEASUREMENT TECHNIQUES

Tests were performed in the subsonic wind tunnel at Embry Riddle Aeronautical University (ERAU) in Daytona Beach, Florida. The general setup is shown in Fig.19:

Fig.19: Experimental Setup.

The experimental setup is detailed in the following sections:

A. The Wind Tunnel

Characteristics and a schematic drawing of the wind tunnel are given in Table 1 and Fig.20 respectively.

Table 1: Wind Tunnel Description.

Type	Closed circuit, rectangular (with corner fillets), closed test section, vertical single return.
Material	Plywood with wood and steel reinforcement.
Propeller	6-blade, fixed pitch, laminated wood, 1.42m dia. Approximate blade pitch angle at 75% radius=30°, spinner diameter=0.41m.
Drive	5000cm ³ , 8 cylinder internal combustion engine, 160 HP, automatic transmission.
Low speed test section	Rectangular, 0-50m/s, 0.914m high x 1.321m wide, contraction ratio=4.92. Cross sectional area=3.845m ² , wall diverge ¼°.
Diffuser	2.44m long, sidewall-centerline angle=9.6° Top/bottom wall-centerline angle=5.8° Vertical splitter plate along centerline.
Turning vanes	Rolled aluminum sheet, 0.40m chord with 0.076m camber.
Flow straightener	In entrance cone, plastic honeycomb, 0.025m x 0.025m cells, 0.13m deep.

Fig.20: Schematic Drawing of the Subsonic Wind Tunnel.

The wind tunnel was operated at a chord Reynolds number of 380,000 - or a corresponding velocity of 18.5m/s at standard conditions - during all the experiments.

The turbulence level of the wind tunnel was measured by finding the critical Reynolds number for a sphere (see Appendix B). Results gave a critical Reynolds number of 3.1×10^5 . Reference 31 gives a relation between the critical Reynolds number of a sphere and the turbulence intensity which is more commonly used to characterize is wind tunnel turbulence. The turbulence intensity is 0.23% at $V = 18.5$ m/s. As will be shown in the next chapter, the choice of the turbulence level did not have a dramatic effect on the XFOIL calculations.

B. Airfoil Mounting

The airfoil was mounted vertically in the test section between two end plates, as shown in Fig.21.

The airfoil was mounted on a rotating axis which was used to change the angle of attack via an electric motor. A 0.5mm gap existed between the tips of the model and the horizontal plates.

Fig.21: Airfoil Mounting.

The test section with the two horizontal plates is shown in Fig.22.

The momentum loss was measured with a rake mounted horizontally behind the airfoil at midspan. The distance between the trailing edge of the airfoil and the rake was $1\frac{1}{2}$ chord lengths, large enough so that the transverse velocity components were negligible (Ref.15). Figure 23 shows the airfoil between the two horizontal plates and the rake behind it.

Fig.22: Test Section.

Fig.23: Mounted Airfoil.

The rake was made of 18 1.6mm outside diameter tubes 5.08mm apart. A detail of the rake is shown in Fig.24:

Fig.24: Detail of the Rake.

C. Manometer

The tubes coming from the airfoil pressure taps and the rake were connected to a 50 tube manometer bank; 23 tubes were used for the upper surface pressure distribution, 9 for the lower surface, 16 for the drag calculation and 2 for the static and total pressures.

One advantage of using the somewhat archaic manometer system was the visualization in real time of the pressure distribution and hence the laminar separation bubble behavior. Figure 25 shows the manometer during an experiment.

Fig.25: Fifty Tube Manometer.

D. Experimental Errors

1. Disturbance of the Freestream

The acoustic and turbulence levels modify the boundary layer and hence must be kept as small as possible

Acoustic level was not measured. However, the only source of acoustical disturbance was the engine and the propeller and their acoustic level was kept low since all experiments were made at low speed.

The turbulence intensity of 0.23% at $V=18.5$ m/s was sufficiently low for the desired test although it can not be considered as a low degree of turbulence.

2. Gaps on the Tips of the Airfoil

A 0.5 mm gap between each tip of the model and the horizontal plates can result in a loss of lift and an increase of drag (Ref.20).

Experiment with and without sealed tips revealed that both pressure distributions were similar except at the trailing edge where the pressure taps were closer to the wall. Also, lift and drag coefficients were not influenced by the existence of the gap since they agreed within an experimental error of 2%.

3. Angle of Attack Setting

The precision in the reading of the angle of attack was 0.1 degree. However the rotating axis that supported the model had a ± 0.3 degree play due to its somewhat loose mechanism.

4. Manometer Reading

The reading accuracy was $\Delta h = 1\text{mm}$ of water which corresponds to an error in pressure of:

$$\Delta p = \rho_{\text{water}} g \Delta h = 10^3 \times 9.8 \times 10^{-3} = 9.8 \text{ Pa}$$

$$\Rightarrow \Delta C_p = \frac{\Delta p}{1/2 \rho_{\text{air}} V^2} = \frac{2 \rho_{\text{water}} g \Delta h}{\rho_{\text{air}} V^2} = \frac{2 \times 9.8}{(18.5)^2 \times 1.225} = 0.047$$

5. Wake Rake Method Errors

The momentum deficit method has inherent flaws at high angles of attack due to the rotational momentum losses (Ref.20). Therefore, at large angles of attack, the drag obtained with the wake rake may be smaller than the actual value.

6. Lift and Drag Hysteresis

At low Reynolds number the lift and drag obtained for increasing angles of attack can be significantly different from the ones obtained with decreasing angles of attack (Ref.1). This hysteresis phenomenon is known as stall hysteresis since it happens at large angles of attack. This effect was minimized during the wind tunnel experiments by obtaining the airfoil characteristics with increasing angles of attack only.

7. Model Accuracy

Particular attention was given to the smoothness of the models. The fiberglass gave a smooth surface. However, in order to remove the excess of epoxy which accumulated at the leading edge during the drying process, the leading edge had to be sanded. Although this was done with the finest sand paper, it caused a loss of smoothness at the leading edge.

The accuracy of the contour was excellent except at the trailing edge where the model was deformed by its own weight

during the manufacturing process, and the trailing edge became thicker when the foam model was covered with fiber glass. Figure 26 a,b compare the contour of the plain and step airfoils with the exact LRN-010 shape.

a: Plain Airfoil.

b: Step Airfoil.

Fig.26 a,b: Actual Plain and Step Airfoils Contours

8. Cubic Spline Interpolation

Experimental lift and moment coefficients were calculated using XFOIL's subroutines (Ref.29). The XFOIL program first interpolated the experimental pressure coefficients with a cubic spline. The interpolation had a tendency to accentuate the experimental errors as shown in Fig.27.

This resulted in a small error in the experimental lift and moment computations, due to the nature of the interpolation.

Fig.27: Pressure Coefficients Interpolated by a Cubic Spline.

V. RESULTS

A. Generalities

Results were obtained from wind tunnel experiments and from the XFOIL design/analysis code. The Reynolds number was kept constant and equal to $R_N=380,000$. The angle of attack and the types of separation control devices were the only parameters varied.

The first model tested was the plain LRN-010 airfoil. It assured the quality of the experimental results and was used as a reference for comparing with the step airfoils. The same model was then tripped at different chord locations with duct tape of various thicknesses. The results helped determine the size and location of the step to be built on the second model. A 2mm step airfoil was then tested followed by a 1mm step airfoil. The three airfoils are shown in Fig.28:

Results are examined in detail in the following sections.

B. Pressure Distributions

Experimental pressure distributions were obtained from the manometer readings.

Fig.28: Plain and Step Airfoils.

1. Modification of the XFOIL Program

The XFOIL program was modified to process the experimental pressure distributions and calculate the airfoil characteristics from it.

Hence, a subroutine was added to read the experimental pressure data and plot it with the numerical data. The subroutine interpolated the data in order to compute the lift, moment and inviscid drag coefficients. The method of computation is described in chapter II, section F.

2. Typical Pressure Distribution

Figure 29 shows a typical pressure distribution with numerical and experimental data. The numerical pressure distribution is shown by a solid line. Numerical and experimental lift, moment and drag coefficients are displayed above the plot. The transition point on the upper surface is also given. The bubble location is shown on the pressure distribution.

Experimental pressure distributions are plotted with the '*' symbol for the upper surface and '#' symbol for the lower surface.

Lift and moment coefficients were computed from the experimental pressure distribution whereas the drag coefficient was obtained from the wake-rake data. C_D and C_{Dexp} both take the skin friction drag into account. The LRN-010 airfoil is displayed under the pressure distribution with the same chord-wise scale as the plot. Mach number, Reynolds number and N_{crit} parameters were kept constant throughout the project. N_{crit} describes the turbulence level of the freestream. It is usually chosen between 7 and 9 (Ref.16). Correspondence between N_{crit} and the turbulence intensity is given in table 2.

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

<u>ALFA</u>	<u>CL</u>	<u>CM</u>	<u>CD</u>	<u>Sta</u>	<u>CL_{EXP}</u>	<u>CM_{EXP}</u>	<u>CD_{EXP}</u>
0.000	0.7863	-0.170	0.00886	0.778	0.7392	-0.130	0.00770

Fig.29: Typical Pressure Distribution.

Table 2: Relation Between N_{crit} and the Degree of Turbulence.

N_{crit}	Turbulence intensity
7	0.161 %
8	0.106 %
9	0.070 %

Figure 30 shows that N_{crit} mainly modifies the transition location on the upper surface:

Fig.30: Comparisons of Pressure Distributions for $N_{crit}=7, 8$ and 9.

$N_{crit}=7$ corresponds to an amplification of the disturbance of e^7 or about 1.1×10^3 . It was found to fit the experimental data better than $N_{crit}=9$. The corresponding turbulence intensity of 0.161% was also the closest to the wind tunnel value. Lower N_{crit} values are rarely used and correspond to turbulence levels greater than 0.25%.

3. Plain Airfoil Analysis

Pressure distributions for various angles of attack from $\alpha=-9.2^\circ$ to $\alpha=13.8^\circ$ are included in Appendix C.

The general agreement between the experimental and numerical plots was good with the exception of the points at the trailing edge. This error was certainly due to the fact that the trailing edge got deformed during the manufacturing process, trailing edge up. The same reason also explained the difference between the numerical and experimental moment coefficients: the trailing edge up deformation increased the experimental moment coefficient. Numerical and experimental lift and drag coefficients agreed within 10%.

As expected, the laminar separation bubble moved forward with increasing angles of attack and disappeared at $\alpha=5.8^\circ$ where the natural transition occurred before 40% of the chord. At $\alpha=5.8^\circ$, the flow detached on the upper surface at 85% chord, thus stalling part of the airfoil. A partial stall of the upper surface is characterized by a constant pressure that extends down to the trailing edge.

An increasing part of the airfoil stalled when the angle of attack was further increased.

For $\alpha < -2^\circ$, the numerical calculations did not converge and hence, only the experimental data are available.

A small kink in the pressure distribution at $x/c=0.45$ was noticed for various angles of attack, in both numerical and experimental data. This was attributed to the very flat upper surface of the LRN-010 around 45% chord. However, results in Ref.5 using the ISES CFD code did not predict this kink.

4. Tripped LRN-010 Airfoil

Some experiments were made with a tripped LRN-010 airfoil in order to observe the effect of the tripping on the airfoil drag and pressure distribution. Several locations and several thicknesses were chosen for the tripping. The goal was not to optimize the airfoil performance but to obtain some useful information which could be used to determine the location and size of the step.

The tripping device chosen was a simple band of tape (duct tape or double-faced tape) placed along the span at various x/c locations as shown in Fig.31:

Fig.31: Tripping Device.

A typical pressure distribution is shown in Fig.32. In this case, the tripping was 1.6 mm thick, 2mm wide and located on the upper surface at $x/c=0.61$. The numerical pressure distribution in Fig.32 was obtained with the plain airfoil. The comparison between the two plots (numerical and experimental) show that the flow decelerates right before the tripping device - characterized by a pressure raise - and accelerates sharply when passing over it - characterized by a pressure drop. Another effect of the tripping on the pressure distribution is to raise the overall pressure over the first half of the upper surface. This naturally resulted in a loss of lift. Table 3 shows the drag of the tripped airfoil for different angles of

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

<u>ALFA</u>	<u>CL</u>	<u>CM</u>	<u>CD</u>	<u>SIN</u>	<u>CL_{EXP}</u>	<u>CM_{EXP}</u>	<u>CD_{EXP}</u>
3.800	1.1953	-0.167	0.00879	0.645	1.0656	-0.134	0.01710

**Fig.32: Experimental Data for the Tripped LRN-010 Airfoil
Compared with the Numerical Data for the Plain Airfoil.**

attack, and for different sizes and locations of the tripping device.

Table 3: Drag Coefficients for Tripped and Clean LRN-010.

Tripping →	x/c=0.63	x/c=0.61	x/c=0.4	x/c=0.4
	h=0.36 mm	h=1.6 mm	h=0.36 mm	h=0.7 mm
Alpha ↓	w=2.77 mm	w=2.0 mm	w=2.8 mm	w=2.8 mm
$\alpha = 3.8^\circ$	Cd=0.0103	Cd=0.0171	Cd=0.0108	Cd=0.0117
$\alpha = 1.8^\circ$			Cd=.0063*	Cd=0.0117

Tripping →	x/c=0.528	x/c=0.528	Plain
	h=0.36 mm	h=0.7 mm	LRN-010
Alpha ↓	w=2.8 mm	w=2.8 mm	airfoil
$\alpha = 3.8^\circ$	Cd=0.1120		Cd=0.0088
$\alpha = 1.8^\circ$	Cd=0.0092	Cd=0.0097	Cd=0.0084
$\alpha = -0.2^\circ$		Cd=.0063*	Cd=0.0089

* lower Cd than for the plain airfoil

The tripping proved to be efficient in two cases: at x/c=0.4, 0.36 mm thickness was sufficient to force the transition, and at x/c=0.528, a 0.7 mm thick tape band was needed to destabilize the boundary layer. As expected, the boundary layer thickness increased with the chord length and

required a thicker tripping device to be destabilized.

5. Location and Size of the Backward Facing Step

Previous results showed that an acceptable range for the step location was $0.4 \leq x/c \leq 0.528$. Since the LRN-010 upper surface is very flat around $x/c=0.4$, a step made at this location would extend over a large part of the airfoil, and hence, modify greatly the overall airfoil shape. Therefore, the $x/c=0.528$ location was selected.

The laminar separation bubble height is about 10% of its length (Ref.19) and its length is usually between 2% to 10% of the chord. Therefore, the bubble height h_b is such that:

$$\begin{aligned} h_B &\leq (10\%) \times (10\%) \\ \Rightarrow h_B &\leq \frac{10}{100} \times \frac{10}{100} \times 300 \\ \Rightarrow h_B &\leq 3 \text{ mm} \end{aligned}$$

Therefore, the step height (1 or 2 millimeters) was chosen to be greater than the height of the tripping device (0.7 mm) so that the bubble could be mostly hidden by the step. Ideally, the step should be high enough to cover the bubble and small enough to prevent the boundary layer to become too thick.

6. Two Millimeters Backward Facing Step

Coordinates of LRN-010 with a 2mm backward facing step

located at $x/c=0.528$ were computed and read by the XFOIL program. Special attention was given at the step location where many points were needed in order to prevent the cubic spline interpolation smoothing off the discontinuity created by the step. However, the XFOIL program computations did not converge for large discontinuities. Therefore, the actual backward facing step used in the computations was slightly inclined as shown by Fig.33:

Fig.33: Actual 2mm Backward Facing Step as Used in the XFOIL Program.

Airfoil coordinates are given in Appendix A. Experimental and numerical pressure distributions over the step airfoil were obtained for various angles of attack. A typical plot is shown in Fig.34.

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

<u>ALFA</u>	<u>CL</u>	<u>CM</u>	<u>CD</u>	<u>SIN</u>	<u>CL_{EXP}</u>	<u>CM_{EXP}</u>	<u>CD_{EXP}</u>
2.000	0.9622	-0.158	0.00892	0.582	0.8082	-0.085	0.01160

Fig.34: Viscous Pressure Distribution over the 2mm Step
Airfoil.

Numerical and experimental pressure distributions compared fairly well except at the step location; unlike the numerical data, experimental data showed a very strong rise of the pressure after the step. However, the inviscid calculation actually predicted this jump as shown in Fig.35.

The inviscid calculation showed that the pressure dropped sharply upstream of the step. It was then followed by a rapid pressure rise at the step location. The bubble was captured downstream of the step where the flow velocity became very small (leading to a stagnation point for $\alpha \geq 3^\circ$). The flow then reattached as the pressure dropped downstream of the step. This phenomenon appeared at $\alpha = -6^\circ$ and its intensity increased with increasing angles of attack.

Comparison of the general trend of the numerical and experimental pressure distributions was striking in the inviscid case. Unfortunately, there was no pressure tap slightly ahead of the step to capture the sharp drop of pressure, but the experimental data further ahead announced this rise.

Viscous calculations seemed to underestimate the effect of the steps on the pressure distribution, especially at the step location where the inviscid formulation was found to fit the data much better.

LRN-010
MACH = 0.100

$\frac{ALFA}{2.000}$	$\frac{CL}{1.0686}$	$\frac{CM}{-0.180}$	$\frac{CL_{exp}}{0.8284}$	$\frac{CM_{exp}}{-0.084}$	$\frac{CD_{exp}}{0.00000}$
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Fig.35: Inviscid Pressure Distribution over the 2mm Step
Airfoil.

For angles of attack greater than 5 degrees ($\alpha \geq 5^\circ$), the flow did not reattach before $x/c \approx 85\%$, leading to flow separation and higher drag.

As with the tripping, the step increased the overall pressure on the upper surface. The numerical pressure distribution on the upper surface was slightly above the experimental one: the step decelerated the overall flow a little more than what was predicted numerically.

The XFOIL code has difficulties with discontinuities such as backward facing steps. However, the program did converge for the small steps involved in this project but the discontinuity may have affected the accuracy of the numerical computations. The integral boundary layer method averages and damps sharp changes in the pressure distribution. This was clear at the step location where the jump of pressure was greatly reduced. Therefore, wind tunnel data were of great importance in this project.

7. One Millimeter Backward Facing Step

The experiment with a 1mm step was conducted since it was noticed that the 2mm step produced an intense jump of pressure and an large drag as well. Figure 36 can be compared with Fig.34.

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

<u>ALFA</u>	<u>CL</u>	<u>CM</u>	<u>CD</u>	<u>STX</u>	<u>CL_{EXP}</u>	<u>CM_{EXP}</u>	<u>CD_{EXP}</u>
2.000	0.9995	-0.167	0.00795	0.700	0.8120	-0.093	0.00880

Fig.36: Viscous Pressure Distribution over the 1mm Step
Airfoil.

As expected, the smaller step induced a smaller jump of pressure. This jump first appeared for $\alpha=0^\circ$ and became bigger as the angle of attack increased. Eventually, for $\alpha \geq 6^\circ$, the flow did not reattach until 85% of the chord. This same phenomenon occurred with the 2mm step airfoil. Since the step discontinuity was smaller, numerical and experimental data compared better.

C. Lift Curve

Figure 37 presents the experimental lift curves for the plain and step airfoils.

Fig.37: Experimental Lift Curves.

Figure 37 shows that the three curves are close to each

other, especially for negative angles of attack. For positive alphas, the plain airfoil generates more lift than the step airfoils. This phenomenon is expected since the step raised the pressure on the upper surface of the airfoil. Table 4 shows how the tripping - using tape or a backward facing step - decreased the lift of the airfoil.

Table 4: Lift Comparison.

Airfoil→ Alpha ↓	Plain	Tape strip $x/c=0.528$ $h=0.7\text{mm}$	2mm step	1mm step
$\alpha = 1.8^\circ$	$C_l=0.94$	$C_l=0.89$	$C_l=0.79$	$C_l=0.79$
$\alpha = -0.2^\circ$	$C_l=0.76$	$C_l=0.65$	$C_l=0.58$	$C_l=0.62$

This comparison shows that the step device decreased the lift more than what the tape strip did. This difference could come from the nature of the device itself, but also from the difference of tripping height: the tape strip was 0.7mm high whereas the smallest step was 1mm high.

The plain airfoil practically stalled at $\alpha=6^\circ$ where the lift became a nonlinear function of the angle of attack. However, the lift still increased slightly after $\alpha=6^\circ$ up to $\alpha=15^\circ$, and numerical results confirmed this tendency. The 2mm and 1mm step airfoils stalled at $\alpha=5^\circ$ and $\alpha=6^\circ$ respectively. At higher angles of attack, the step airfoil lift was quite different from the plain airfoil lift, especially in the 2mm step case where the lift started to

decrease. This phenomenon can be explained by looking at the transition location versus alpha, presented in Fig.38.

Fig.38: Transition Location Versus α , Obtained from the XFOIL Program.

Five degrees was the critical angle above which the transition occurred before the step. The purpose of the step is to force the transition and capture the laminar separation bubble. However, for angles of attack greater than 5° , natural transition occurs before the step and hence, there is no need for the step modification. Moreover, the effect of the step at these angles of attack became detrimental to the airfoil performance since it caused the boundary layer to become thicker and hence, more

subject to separation. It was concluded that the step had a negative effect on the lift once the transition occurred ahead of the modification.

Figure 38 clearly shows the beneficial effect of the step for angles of attack less than 5 degrees: the steps advanced the transition and traded the natural separation bubble for the artificially formed bubble at the step. This lead to decreased drag in some instances. In others, the step simply acted as a tripping device. Also, the bigger the step height, the sooner the transition occurred. This result was expected since a bigger step produced a higher flow disturbance which destabilized the laminar flow more readily.

Figures 39 a,b,c compares the experimental and numerical lift curves for the three airfoils.

a: Plain Airfoil.

b: 2mm Step Airfoil.

c: 1mm Step Airfoil.

Fig.39 a,b,c: Comparison between Numerical and Experimental Lift Curves.

Agreements for the plain airfoil were excellent and gave good credit to the experimental data. Figure 39a shows that the LRN-010 stalled around $\alpha=6^\circ$ even though the lift coefficient slightly increased with higher angles of attack. Eventually, the lift coefficient drops as α increases.

Results for step airfoils from Fig. 39 b,c show that the experimental lift was consistently lower than the lift computed numerically, with the difference increasing with the angle of attack. One explanation is that XFOIL's viscous calculation has a tendency to underestimate the effect of the step on the pressure distribution and hence, coefficients computed from the pressure distribution were overestimated. Also, experimental errors could account for this difference, in particular the model roughness at the leading edge, which could have decelerated the flow, and hence lowered the airfoil's lift. The reduced camber and the slightly smaller chord of the models was also a source of error.

The numerical and experimental angles of attack at which the separation bubble burst were found to be identical for the three airfoils. Nevertheless, an even greater lift difference between computational and experimental values confirmed that the discontinuity affected the pressure distribution around the airfoil more than what computations predicted.

D. Drag Versus Angle of Attack

Figure 40 presents the vector pressure distribution over the 2mm airfoil for $\alpha=0^\circ$.

Fig.40: Vector Pressure Distribution.

The pressure vectors on the backward face pointing downward obviously created an additional drag on the airfoil. The question is to know if the drag reduction obtained from decreasing the bubble size would be larger than the drag associated with the tripping device itself. The answer is given in Fig.41.

A quick look at Fig.41 indicates a horizontal shift in the curve for the step airfoil relative to the curve for the plain airfoil. This shift is due to the presence of the step itself and it has to be considered as a change in drag

due to the modification ,rather than a shift in angle of attack. The angle of attack calibration was accurate within $\pm 0.5^\circ$ and its accuracy was confirmed by the comparison of experimental and numerical pressure distributions and lift curves.

Fig.41: Experimental Drag for the Plain and Step Airfoils.

Experimental results showed that the 2mm step airfoil had lower drag than the plain airfoil only for $\alpha < -1^\circ$. Also, the 1mm step airfoil lowered the drag of the airfoil for $\alpha < 2^\circ$. Drag reduction ranged between 25% and 50% for negative angles of attack.

The backward facing step - independently of its height - increased the drag for $\alpha \geq 3^\circ$. At these angles, the

drag reduction obtained by tripping the flow was insufficient to overcome the drag penalty brought by the backward facing step. The transition was also seen to occur before the step for $\alpha \geq 5^\circ$ and this had a direct repercussion on the step airfoil drag since it increased substantially at this point.

Fig.42 a,b,c compare numerical and experimental drag for the three airfoils.

a: Plain Airfoil.

b: 2mm Step Airfoil.

c: 1mm Step Airfoil.

Fig.42 a,b,c: Comparison between Numerical and Experimental Drag.

The comparison was excellent for the plain airfoil for $\alpha \leq 6^\circ$. For larger angles, the experimental drag became much less than the numerical drag. This experimental error is attributed to the failure of the wake rake method to take into account the rotational momentum loss which occurs at large angles of attack. Therefore, the experimental drag was accurate within the range of $-6^\circ \leq \alpha \leq 6^\circ$.

Figure 42 b,c show that numerical and experimental drag compared well although some differences appeared for $\alpha \geq 5^\circ$. Two kinds of errors arose: the experimental error due to the wake rake method, and the numerical error where computations underestimated the drag increase associated with the step induced pressure jump at high angles of attack. These two errors made the results difficult to analyze since neither the numerical nor the experimental drags were accurate for $\alpha \geq 6^\circ$.

However, the present study did not give too much importance to this range of angles of attack since the improvements brought by the step modification happened for small or negative alpha values. These angles are the ones of interest for cruise condition.

E. Airfoil Performance

The airfoil performance is measured in terms of lift to drag ratio. Figure 43 a,b presents the numerical results obtained with the XFOIL program.

The plain airfoil lift to drag ratios are plotted in both graphs a and b so the effect of the step airfoils results can be easily compared to it.

Computations for the 2mm step airfoil showed that the modification actually improved the airfoil performance for angles of attack less than 1 degree: the backward facing step increased the performance about 20% for $\alpha \leq 0^\circ$. However, the performance deteriorated for $\alpha \geq 2^\circ$, because the 2mm step produced a higher drag and a lower lift at these angles of attack.

Results shown in Fig.43b were even more optimistic since the 1mm backward facing step increased the maximum lift to drag ratio by 10 %, leading to a very high lift to drag ratio for a Reynolds number as low as 380,000. The 1mm step actually increased the airfoil performance 10% for $\alpha \leq 5^\circ$.

Figure 44 a,b presents the lift to drag ratio of the three airfoils as obtained experimentally.

Fig.a

Fig.b

Fig.43 a,b: Numerical Lift to Drag Ratio.

Fig.a

Fig.b

Fig.44 a,b: Experimental Lift to Drag Ratios.

Despite the experimental errors associated with the wind tunnel testings, these results should be considered with attention, especially because XFOIL was seen to handle discontinuities such as the backward facing step with some problems.

Experimental results in terms of lift to drag ratio were significantly different since the 2mm step did not show any improvement at any angle of attack and the 1mm step only improved the performance for $\alpha \leq 1^\circ$. The average improvement was about 10%.

Several factors explained the dissimilitude between experimental and numerical results: lift and drag were found to agree closely for negative angles of attack, and growing difference appeared for $\alpha \geq 0^\circ$ where the numerical lift and drag became too optimistic - i.e. too high a lift and too low a drag -. The actual pressure drag associated with the backward facing step was underestimated by XFOIL. Also, a decrease in lift due to the separation control device was also underestimated in the numerical case. Finally, experimental factors, such as the airfoil's smoothness certainly had a role in the differences observed. Consequently, these factors led to optimistic calculated lift to drag ratios and pessimistic experimental ratios.

Although the performance of the step airfoils was not improved for positive angles of attack, the 1mm step still showed a beneficial effects for $\alpha \leq 0^\circ$. For $\alpha \leq 0^\circ$, the step

worked as a tripping device, causing the boundary layer to become turbulent and reducing the trailing edge separation, thus improving the airfoil performance. At $\alpha=1^\circ$, the separation bubble formed at $x/c=0.7$, the 1mm step, by tripping the flow, effected a 12% reduction in the drag with a 0.5% reduction in lift. No cases where the separation bubble was actually captured by the step was observed. Further tests are needed in order to validate the hypothesis that the step can be used to control the separation bubble. The increase in performance by tripping the flow may be obtained easier by means of tape strips.

VI. CONCLUSION

The low Reynolds number airfoil LRN-010 has an extensive laminar flow on its upper surface at low angles of attack. Laminar separation occurs before transition and a separation bubble forms, which substantially increases the airfoil's drag. The backward facing step is a new passive bubble control device adapted to the upper surface of the LRN-010. The XFOIL design/analysis code procured the numerical results whereas experimental data were obtained from wind tunnel tests of models built for the purpose of the study.

Lift, drag and lift to drag ratio were computed from the pressure distributions and wake momentum loss at various angles of attack. Results showed a very good correlation between experimental and numerical data for the plain LRN-010 airfoil. The step airfoils showed a difference of results between XFOIL and wind tunnel tests for angles of attack $\alpha \geq 5^\circ$. Experimental errors as well as numerical errors explained these differences: a slightly rough region at the leading edge of the step airfoils slowed the flow over the surface, thicken the boundary layer and caused early transition. Numerical errors arose from the poor capability of XFOIL to handle abrupt changes in the

airfoil's shape. However, both wind tunnel tests and computational analyses gave encouraging results for small or negative angles of attack.

An angle of attack of 5 degrees was the upper limit for the step to force the transition since natural transition occurred before the step for $\alpha \geq 5^\circ$. For lower angles of attack, the step was seen to capture the bubble and force the transition. For lower alphas ($\alpha \leq 1^\circ$), the tripping device improved the airfoil performance.

The height of the step clearly influenced the results. The smaller step size (1mm high or $h/c=0.003$) generally gave better improvements of the airfoil characteristics. Even the 1mm step seemed to thicken the boundary layer too much for angles $\alpha \geq 3^\circ$, and hence, the size of the step would have to be reduced for higher angles of attack. Ideally, the step size should be a decreasing function of alpha, and a simple mechanism could easily regulate the step height from 1mm for the most negative alphas down to no step at all once the natural transition occurs before the step, i.e. when alpha exceeds 5 degrees.

The step location was chosen at $x/c=0.528$ for this investigation but a possible range for the step was $0.4 \leq x/c \leq 1.0$, corresponding to the area where the airfoil thickness decreases with x/c . Since the natural transition moved forward when the angle of attack increased, a step located at $x/c=0.4$ could force the transition for alphas

greater than 5 degrees. In this case, the step height should be smaller than 1mm since the boundary layer would be thinner at this location. Further investigations could determine which location and step height would optimize the airfoil performance.

The Reynolds number is an interesting parameter that was kept constant in this project since the purpose of the study was to determine if the backward facing step could possibly have a beneficial effect on the LRN-010 characteristics. Since the location of the bubble is affected by the Reynolds number, its effect on the performance of the backward facing step is of interest.

The effects of the backward facing step could also be investigated on other airfoils such as Liebeck airfoils - LA2573 for example - on which a laminar separation bubble forms at low Reynolds numbers.

Obviously, only a very small range of the backward facing step capabilities were investigated and results obtained in this study encourage further developments of this laminar separation control method.

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APPENDIX A

Coordinates of the Plain, 2mm Step and 1mm Step Airfoils

Plain airfoil coordinates:

X	Y
1.00000	-7.17900E-03
0.998061	-6.70700E-03
0.996045	-6.24300E-03
0.993952	-5.77200E-03
0.991781	-5.27200E-03
0.989523	-4.74700E-03
0.987172	-4.19300E-03
0.984721	-3.60800E-03
0.982159	-2.98900E-03
0.979479	-2.32900E-03
0.976667	-1.62800E-03
0.973712	-8.84000E-04
0.970595	-9.20000E-05
0.967301	7.49000E-04
0.963809	1.65400E-03
0.960094	2.62200E-03
0.956126	3.66600E-03
0.951873	4.80600E-03
0.947289	6.03500E-03
0.942326	7.38700E-03
0.936915	8.87200E-03
0.930974	1.05110E-02
0.924405	1.23480E-02
0.917064	1.44100E-02
0.908782	1.67650E-02
0.899315	1.94780E-02
0.888339	2.26410E-02
0.875414	2.63760E-02
0.859959	3.08930E-02
0.841220	3.64470E-02
0.818279	4.31030E-02
0.790435	5.10100E-02
0.757552	5.98110E-02
0.720284	6.86730E-02
0.679913	7.69940E-02
0.637640	8.41090E-02
0.594219	8.98250E-02
0.550106	9.40890E-02
0.505594	9.69770E-02
0.460912	9.84850E-02
0.4162710	9.86930E-02
0.371996	9.86930E-02
0.328533	9.76110E-02
0.286698	9.52090E-02
0.247705	9.17060E-02
0.212840	8.72340E-02
0.182981	8.21170E-02
0.158195	7.68340E-02
0.137885	7.17360E-02
0.121209	6.70330E-02
0.107368	6.27350E-02
9.57130E-02	5.88170E-02
8.57490E-02	5.52360E-02
7.71170E-02	5.19510E-02
6.95430E-02	4.89200E-02
6.28320E-02	4.61030E-02
5.68280E-02	4.34630E-02
5.14140E-02	4.09760E-02
4.65010E-02	3.86180E-02
4.20140E-02	3.63700E-02

3.78950E-02	3.42200E-02
3.40980E-02	3.21560E-02
3.05870E-02	3.01650E-02
2.73290E-02	2.82370E-02
2.42990E-02	2.63620E-02
2.14770E-02	2.45320E-02
1.88420E-02	2.27410E-02
1.63810E-02	2.09870E-02
1.40920E-02	1.92580E-02
1.19600E-02	1.75390E-02
9.97800E-03	1.58310E-02
8.14400E-03	1.41300E-02
6.45400E-03	1.24300E-02
4.92100E-03	1.07250E-02
3.55800E-03	9.00300E-03
2.35900E-03	7.24900E-03
1.30200E-03	5.47600E-03
4.20000E-04	1.91000E-03
1.00000E-06	-4.80000E-05
1.43800E-03	-1.53900E-03
3.16000E-03	-2.81800E-03
5.29700E-03	-3.44300E-03
7.53300E-03	-4.04800E-03
9.90700E-03	-4.47500E-03
1.23950E-02	-4.84800E-03
1.50070E-02	-5.13300E-03
1.77490E-02	-5.37000E-03
2.06300E-02	-5.56000E-03
2.36610E-02	-5.71500E-03
2.68570E-02	-5.84000E-03
3.02360E-02	-5.94200E-03
3.38170E-02	-6.02400E-03
3.76240E-02	-6.09100E-03
4.16870E-02	-6.14100E-03
4.60380E-02	-6.17200E-03
5.07210E-02	-6.19400E-03
5.57860E-02	-6.20000E-03
6.12970E-02	-6.19000E-03
6.73310E-02	-6.16100E-03
7.39910E-02	-6.12400E-03
8.14010E-02	-6.06500E-03
8.97290E-02	-5.99300E-03
9.91960E-02	-5.89200E-03
0.110096	-5.75300E-03
0.122821	-5.56200E-03
0.137894	-5.29900E-03
0.155982	-4.93600E-03
0.177845	-4.43400E-03
0.204140	-3.75800E-03
0.235027	-2.82400E-03
0.270083	-1.44600E-03
0.308637	4.21000E-04
0.349594	2.78400E-03
0.391942	5.34100E-03
0.434968	7.93900E-03
0.478225	1.04550E-02
0.521468	1.27270E-02
0.564493	1.46280E-02
0.607122	1.61420E-02
0.649033	1.72540E-02
0.689663	1.78260E-02

0.728069	1.78810E-02
0.763104	1.74100E-02
0.793792	1.65360E-02
0.819809	1.53610E-02
0.841457	1.40730E-02
0.859422	1.26060E-02
0.874447	1.11760E-02
0.887159	9.84400E-03
0.898062	8.63000E-03
0.907536	7.51300E-03
0.915867	6.47000E-03
0.923277	5.49000E-03
0.929930	4.57200E-03
0.935955	3.71200E-03
0.941454	2.90500E-03
0.946503	2.14900E-03
0.951168	1.43600E-03
0.955498	7.59000E-04
0.959537	1.21000E-04
0.963319	-4.87000E-04
0.966873	-1.06700E-03
0.970226	-1.61900E-03
0.973395	-2.15000E-03
0.976400	-2.66800E-03
0.979254	-3.17200E-03
0.981974	-3.66100E-03
0.984568	-4.13900E-03
0.987050	-4.60300E-03
0.989426	-5.05900E-03
0.991706	-5.50500E-03
0.993898	-5.93900E-03
0.996003	-6.38000E-03
0.998034	-6.80200E-03
1.00000	-7.17900E-03

2mm step airfoil coordinates:

X	Y
1.00000	-7.17900E-03
0.998061	-6.70700E-03
0.996045	-6.24300E-03
0.993952	-5.77200E-03
0.991781	-5.27200E-03
0.989523	-4.74700E-03
0.987172	-4.19300E-03
0.984721	-3.60800E-03
0.982159	-2.98900E-03
0.979479	-2.32900E-03
0.976667	-1.62800E-03
0.973712	-8.84000E-04
0.970595	-9.20000E-05
0.967301	7.49000E-04
0.963809	1.65400E-03
0.960094	2.62200E-03
0.956126	3.66600E-03
0.951873	4.80600E-03
0.947289	6.03500E-03
0.942326	7.38700E-03
0.936915	8.87200E-03
0.930974	1.05110E-02
0.924405	1.23480E-02
0.917064	1.44100E-02
0.908782	1.67650E-02
0.899315	1.94780E-02
0.888339	2.26410E-02
0.875414	2.63760E-02
0.859959	3.08930E-02
0.841220	3.64470E-02
0.818279	4.31030E-02
0.790435	5.10100E-02
0.757552	5.98110E-02
0.720284	6.86730E-02
0.679913	7.69940E-02
0.637640	8.41090E-02
0.594219	8.98250E-02
0.569752	9.219E-02
0.56	9.219E-02
0.550106	9.219E-02
0.545	9.219E-02
0.540	9.219E-02
0.535	9.219E-02
0.530	9.219E-02
0.5296	9.219E-02
0.5292	9.219E-02
0.529	9.219E-02
0.528	9.2119E-02
0.528	9.38E-02
0.528	9.55233E-02
0.527	9.55881E-02
0.526	9.56530E-02
0.525	9.57179E-02
0.52	9.60423E-02
0.515	9.63667E-02
0.51	9.66911E-02
0.505594	9.69770E-02
0.460912	9.84850E-02
0.4162710	9.86930E-02
0.371996	9.86930E-02

0.328533	9.76110E-02
0.286698	9.52090E-02
0.247705	9.17060E-02
0.212840	8.72340E-02
0.182981	8.21170E-02
0.158195	7.68340E-02
0.137885	7.17360E-02
0.121209	6.70330E-02
0.107368	6.27350E-02
9.57130E-02	5.88170E-02
8.57490E-02	5.52360E-02
7.71170E-02	5.19510E-02
6.95430E-02	4.89200E-02
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5.14140E-02	4.09760E-02
4.65010E-02	3.86180E-02
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3.78950E-02	3.42200E-02
3.40980E-02	3.21560E-02
3.05870E-02	3.01650E-02
2.73290E-02	2.82370E-02
2.42990E-02	2.63620E-02
2.14770E-02	2.45320E-02
1.88420E-02	2.27410E-02
1.63810E-02	2.09870E-02
1.40920E-02	1.92580E-02
1.19600E-02	1.75390E-02
9.97800E-03	1.58310E-02
8.14400E-03	1.41300E-02
6.45400E-03	1.24300E-02
4.92100E-03	1.07250E-02
3.55800E-03	9.00300E-03
2.35900E-03	7.24900E-03
1.30200E-03	5.47600E-03
4.20000E-04	1.91000E-03
1.00000E-06	-4.80000E-05
1.43800E-03	-1.53900E-03
3.16000E-03	-2.81800E-03
5.29700E-03	-3.44300E-03
7.53300E-03	-4.04800E-03
9.90700E-03	-4.47500E-03
1.23950E-02	-4.84800E-03
1.50070E-02	-5.13300E-03
1.77490E-02	-5.37000E-03
2.06300E-02	-5.56000E-03
2.36610E-02	-5.71500E-03
2.68570E-02	-5.84000E-03
3.02360E-02	-5.94200E-03
3.38170E-02	-6.02400E-03
3.76240E-02	-6.09100E-03
4.16870E-02	-6.14100E-03
4.60380E-02	-6.17200E-03
5.07210E-02	-6.19400E-03
5.57860E-02	-6.20000E-03
6.12970E-02	-6.19000E-03
6.73310E-02	-6.16100E-03
7.39910E-02	-6.12400E-03
8.14010E-02	-6.06500E-03
8.97290E-02	-5.99300E-03
9.91960E-02	-5.89200E-03

0.110096	-5.75300E-03
0.122821	-5.56200E-03
0.137894	-5.29900E-03
0.155982	-4.93600E-03
0.177845	-4.43400E-03
0.204140	-3.75800E-03
0.235027	-2.82400E-03
0.270083	-1.44600E-03
0.308637	4.21000E-04
0.349594	2.78400E-03
0.391942	5.34100E-03
0.434968	7.93900E-03
0.478225	1.04550E-02
0.521468	1.27270E-02
0.564493	1.46280E-02
0.607122	1.61420E-02
0.649033	1.72540E-02
0.689663	1.78260E-02
0.728069	1.78810E-02
0.763104	1.74100E-02
0.793792	1.65360E-02
0.819809	1.53610E-02
0.841457	1.40730E-02
0.859422	1.26060E-02
0.874447	1.11760E-02
0.887159	9.84400E-03
0.898062	8.63000E-03
0.907536	7.51300E-03
0.915867	6.47000E-03
0.923277	5.49000E-03
0.929930	4.57200E-03
0.935955	3.71200E-03
0.941454	2.90500E-03
0.946503	2.14900E-03
0.951168	1.43600E-03
0.955498	7.59000E-04
0.959537	1.21000E-04
0.963319	-4.87000E-04
0.966873	-1.06700E-03
0.970226	-1.61900E-03
0.973395	-2.15000E-03
0.976400	-2.66800E-03
0.979254	-3.17200E-03
0.981974	-3.66100E-03
0.984568	-4.13900E-03
0.987050	-4.60300E-03
0.989426	-5.05900E-03
0.991706	-5.50500E-03
0.993898	-5.93900E-03
0.996003	-6.38000E-03
0.998034	-6.80200E-03
1.00000	-7.17900E-03

1mm step airfoil coordinates:

X	Y
1.00000	-7.17900E-03
0.998061	-6.70700E-03
0.996045	-6.24300E-03
0.993952	-5.77200E-03
0.991781	-5.27200E-03
0.989523	-4.74700E-03
0.987172	-4.19300E-03
0.984721	-3.60800E-03
0.982159	-2.98900E-03
0.979479	-2.32900E-03
0.976667	-1.62800E-03
0.973712	-8.84000E-04
0.970595	-9.20000E-05
0.967301	7.49000E-04
0.963809	1.65400E-03
0.960094	2.62200E-03
0.956126	3.66600E-03
0.951873	4.80600E-03
0.947289	6.03500E-03
0.942326	7.38700E-03
0.936915	8.87200E-03
0.930974	1.05110E-02
0.924405	1.23480E-02
0.917064	1.44100E-02
0.908782	1.67650E-02
0.899315	1.94780E-02
0.888339	2.26410E-02
0.875414	2.63760E-02
0.859959	3.08930E-02
0.841220	3.64470E-02
0.818279	4.31030E-02
0.790435	5.10100E-02
0.757552	5.98110E-02
0.720284	6.86730E-02
0.679913	7.69940E-02
0.637640	8.41090E-02
0.601575	8.88566E-02
0.59	8.88566E-02
0.58	8.88566E-02
0.57	8.88566E-02
0.56	8.88566E-02
0.550106	8.88566E-02
0.545	8.88566E-02
0.540	8.88566E-02
0.535	8.88566E-02
0.530	8.88566E-02
0.5296	8.88566E-02
0.5292	8.88566E-02
0.529	8.88566E-02
0.5289	8.88566E-02
0.5288	8.90E-02
0.5286	9.04E-02
0.5284	9.21E-02
0.5282	9.38E-02
0.528	9.55233E-02
0.527	9.55881E-02
0.526	9.56530E-02
0.525	9.57179E-02
0.52	9.60423E-02
0.515	9.63667E-02

0.51	9.66911E-02
0.505594	9.69770E-02
0.460912	9.84850E-02
0.4162710	9.86930E-02
0.371996	9.86930E-02
0.328533	9.76110E-02
0.286698	9.52090E-02
0.247705	9.17060E-02
0.212840	8.72340E-02
0.182981	8.21170E-02
0.158195	7.68340E-02
0.137885	7.17360E-02
0.121209	6.70330E-02
0.107368	6.27350E-02
9.57130E-02	5.88170E-02
8.57490E-02	5.52360E-02
7.71170E-02	5.19510E-02
6.95430E-02	4.89200E-02
6.28320E-02	4.61030E-02
5.68280E-02	4.34630E-02
5.14140E-02	4.09760E-02
4.65010E-02	3.86180E-02
4.20140E-02	3.63700E-02
3.78950E-02	3.42200E-02
3.40980E-02	3.21560E-02
3.05870E-02	3.01650E-02
2.73290E-02	2.82370E-02
2.42990E-02	2.63620E-02
2.14770E-02	2.45320E-02
1.88420E-02	2.27410E-02
1.63810E-02	2.09870E-02
1.40920E-02	1.92580E-02
1.19600E-02	1.75390E-02
9.97800E-03	1.58310E-02
8.14400E-03	1.41300E-02
6.45400E-03	1.24300E-02
4.92100E-03	1.07250E-02
3.55800E-03	9.00300E-03
2.35900E-03	7.24900E-03
1.30200E-03	5.47600E-03
4.20000E-04	1.91000E-03
1.00000E-06	-4.80000E-05
1.43800E-03	-1.53900E-03
3.16000E-03	-2.81800E-03
5.29700E-03	-3.44300E-03
7.53300E-03	-4.04800E-03
9.90700E-03	-4.47500E-03
1.23950E-02	-4.84800E-03
1.50070E-02	-5.13300E-03
1.77490E-02	-5.37000E-03
2.06300E-02	-5.56000E-03
2.36610E-02	-5.71500E-03
2.68570E-02	-5.84000E-03
3.02360E-02	-5.94200E-03
3.38170E-02	-6.02400E-03
3.76240E-02	-6.09100E-03
4.16870E-02	-6.14100E-03
4.60380E-02	-6.17200E-03
5.07210E-02	-6.19400E-03
5.57860E-02	-6.20000E-03
6.12970E-02	-6.19000E-03

6.73310E-02	-6.16100E-03
7.39910E-02	-6.12400E-03
8.14010E-02	-6.06500E-03
8.97290E-02	-5.99300E-03
9.91960E-02	-5.89200E-03
0.110096	-5.75300E-03
0.122821	-5.56200E-03
0.137894	-5.29900E-03
0.155982	-4.93600E-03
0.177845	-4.43400E-03
0.204140	-3.75800E-03
0.235027	-2.82400E-03
0.270083	-1.44600E-03
0.308637	4.21000E-04
0.349594	2.78400E-03
0.391942	5.34100E-03
0.434968	7.93900E-03
0.478225	1.04550E-02
0.521468	1.27270E-02
0.564493	1.46280E-02
0.607122	1.61420E-02
0.649033	1.72540E-02
0.689663	1.78260E-02
0.728069	1.78810E-02
0.763104	1.74100E-02
0.793792	1.65360E-02
0.819809	1.53610E-02
0.841457	1.40730E-02
0.859422	1.26060E-02
0.874447	1.11760E-02
0.887159	9.84400E-03
0.898062	8.63000E-03
0.907536	7.51300E-03
0.915867	6.47000E-03
0.923277	5.49000E-03
0.929930	4.57200E-03
0.935955	3.71200E-03
0.941454	2.90500E-03
0.946503	2.14900E-03
0.951168	1.43600E-03
0.955498	7.59000E-04
0.959537	1.21000E-04
0.963319	-4.87000E-04
0.966873	-1.06700E-03
0.970226	-1.61900E-03
0.973395	-2.15000E-03
0.976400	-2.66800E-03
0.979254	-3.17200E-03
0.981974	-3.66100E-03
0.984568	-4.13900E-03
0.987050	-4.60300E-03
0.989426	-5.05900E-03
0.991706	-5.50500E-03
0.993898	-5.93900E-03
0.996003	-6.38000E-03
0.998034	-6.80200E-03
1.00000	-7.17900E-03

APPENDIX B

ERAU Wind Tunnel Turbulence Factor Determination

The level of freestream turbulence was found by measuring the critical Reynolds number over a sphere.

The critical Reynolds number over a sphere depends strongly on the degree of turbulence of the wind tunnel. Critical Reynolds number over a 0.127m diameter sphere was obtained by measuring the pressure difference between the forward stagnation point and a point at the rear of the sphere. The value of the critical Reynolds number measured on a sphere in free flight is $R_{crit}=3.85 \times 10^5$.

Figure 45 shows the experimental curve relating the pressure difference δC_p to the Reynolds number. $\delta C_p=1.22$ corresponded to R_{crit} over the sphere.

Fig.45: Critical Reynolds Number Determination.

Results gave $R_{crit}=310,000$. Figure 46 from Ref.31 gives

a relation between the critical Reynolds number of a sphere and the intensity of the wind tunnel turbulence level.

Fig.46: Relation between R_{crit} and the Degree of Turbulence (Ref.31).

Figure 46 gives, for a corresponding $R_{crit}=310,000$, a degree of turbulence for the ERAU wind tunnel of 0.23%.

APPENDIX C
Pressure Distributions

This appendix shows the experimental and numerical pressure distributions obtained for the plain, the tripped, the 2mm step and the 1mm step airfoils.

Plain airfoil pressure distributions:

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

<u>ALFA</u>	<u>CL</u>	<u>CM</u>	<u>CD</u>	<u>SIN</u>	<u>CL_{EXP}</u>	<u>CM_{EXP}</u>	<u>CD_{EXP}</u>
-9.200	0.0000	0.000	0.00000	0.000	-0.2425	-0.017	0.00000

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

<u>ALFA</u>	<u>CL</u>	<u>CM</u>	<u>CD</u>	<u>Sta</u>	<u>CL_{EXP}</u>	<u>CM_{EXP}</u>	<u>CD_{EXP}</u>
-7.200	0.0000	0.000	0.00000	0.000	-0.0727	-0.099	0.00000

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

<u>ALFA</u>	<u>CL</u>	<u>CM</u>	<u>CD</u>	<u>SIN</u>	<u>CL_{EXP}</u>	<u>CM_{EXP}</u>	<u>CD_{EXP}</u>
-5.200	0.0000	0.000	0.00000	0.000	0.1411	-0.107	0.00000

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

<u>ALFA</u>	<u>CL</u>	<u>CM</u>	<u>CD</u>	<u>Str</u>	<u>CL_{EXP}</u>	<u>CM_{EXP}</u>	<u>CD_{EXP}</u>
-3.200	0.0000	0.000	0.00000	0.000	0.4100	-0.139	0.01610

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

<u>ALFA</u>	<u>CL</u>	<u>CM</u>	<u>CD</u>	<u>STA</u>	<u>CL_{EXP}</u>	<u>CM_{EXP}</u>	<u>CD_{EXP}</u>
-2.000	0.5590	-0.169	0.01271	0.807	0.5467	-0.135	0.01260

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

<u>ALFA</u>	<u>CL</u>	<u>CM</u>	<u>CD</u>	<u>Str</u>	<u>CL_{EXP}</u>	<u>CM_{EXP}</u>	<u>CD_{EXP}</u>
-1.000	0.6796	-0.171	0.00982	0.793	0.6821	-0.125	0.00970

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

<u>ALFA</u>	<u>CL</u>	<u>CM</u>	<u>CD</u>	<u>Sta</u>	<u>CL_{EXP}</u>	<u>CM_{EXP}</u>	<u>CD_{EXP}</u>
-0.200	0.7655	-0.170	0.00894	0.781	0.7621	-0.137	0.00750

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

<u>ALFA</u>	<u>CL</u>	<u>CM</u>	<u>CD</u>	<u>STa</u>	<u>CL_{EXP}</u>	<u>CM_{EXP}</u>	<u>CD_{EXP}</u>
1.000	0.8932	-0.168	0.00837	0.759	0.8268	-0.132	0.00850

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

<u>ALFA</u>	<u>CL</u>	<u>CM</u>	<u>CD</u>	<u>Sta</u>	<u>CL_{EXP}</u>	<u>CM_{EXP}</u>	<u>CD_{EXP}</u>
1.800	0.9831	-0.169	0.00840	0.743	0.9435	-0.144	0.00880

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

<u>ALFA</u>	<u>CL</u>	<u>CM</u>	<u>CD</u>	<u>Sta</u>	<u>CL_{EXP}</u>	<u>CM_{EXP}</u>	<u>CD_{EXP}</u>
3.000	1.1137	-0.168	0.00852	0.704	1.0482	-0.145	0.00850

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

<u>ALFA</u>	<u>CL</u>	<u>CM</u>	<u>CD</u>	<u>STX</u>	<u>CL_{EXP}</u>	<u>CM_{EXP}</u>	<u>CD_{EXP}</u>
3.800	1.1953	-0.167	0.00879	0.645	1.1935	-0.160	0.00810

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

<u>ALFA</u>	<u>CL</u>	<u>CM</u>	<u>CD</u>	<u>Sta</u>	<u>CL_{EXP}</u>	<u>CM_{EXP}</u>	<u>CD_{EXP}</u>
3.800	1.1953	-0.167	0.00879	0.645	1.1751	-0.143	0.00830

Sealed tips

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

<u>ALFA</u>	<u>CL</u>	<u>CM</u>	<u>CD</u>	<u>Str</u>	<u>CL_{EXP}</u>	<u>CM_{EXP}</u>	<u>CD_{EXP}</u>
5.800	1.3331	-0.152	0.01343	0.388	1.2935	-0.140	0.01180

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

α	C_L	C_M	C_D	S_{Ta}	$C_{L_{EXP}}$	$C_{M_{EXP}}$	$C_{D_{EXP}}$
7.800	1.3802	-0.129	0.02343	0.041	1.4119	-0.141	0.01500

Tripped airfoil pressure distributions

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

<u>ALFA</u>	<u>CL</u>	<u>CM</u>	<u>CD</u>	<u>Str</u>	<u>CL_{EXP}</u>	<u>CM_{EXP}</u>	<u>CD_{EXP}</u>
3.800	1.1953	-0.167	0.00878	0.645	0.9961	-0.124	0.01030

$x/c=0.631$ $h=0.36\text{mm}$ $w=2.77\text{mm}$

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

<u>ALFA</u>	<u>CL</u>	<u>CM</u>	<u>CD</u>	<u>Sta</u>	<u>CL_{EXP}</u>	<u>CM_{EXP}</u>	<u>CD_{EXP}</u>
3.800	1.1953	-0.167	0.00879	0.645	1.1339	-0.136	0.01080

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

<u>ALFA</u>	<u>CL</u>	<u>CM</u>	<u>CD</u>	<u>Stn</u>	<u>CL_{EXP}</u>	<u>CM_{EXP}</u>	<u>CD_{EXP}</u>
1.800	0.9831	-0.169	0.00840	0.743	0.9238	-0.139	0.00630

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

<u>ALFA</u>	<u>CL</u>	<u>CM</u>	<u>CD</u>	<u>Sta</u>	<u>CL_{EXP}</u>	<u>CM_{EXP}</u>	<u>CD_{EXP}</u>
3.800	1.1953	-0.167	0.00879	0.645	1.0865	-0.133	0.01170

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

<u>ALFA</u>	<u>CL</u>	<u>CM</u>	<u>CD</u>	<u>SrA</u>	<u>CL_{EXP}</u>	<u>CM_{EXP}</u>	<u>CD_{EXP}</u>
1.800	0.9831	-0.169	0.00840	0.743	0.8420	-0.128	0.01170

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

<u>ALFA</u>	<u>CL</u>	<u>CM</u>	<u>CD</u>	<u>Sta</u>	<u>CL_{EXP}</u>	<u>CM_{EXP}</u>	<u>CD_{EXP}</u>
3.800	1.1953	-0.167	0.00879	0.645	1.0006	-0.123	0.01120

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

<u>ALFA</u>	<u>CL</u>	<u>CM</u>	<u>CD</u>	<u>STa</u>	<u>CL_{EXP}</u>	<u>CM_{EXP}</u>	<u>CD_{EXP}</u>
1.800	0.9831	-0.169	0.00840	0.743	0.9522	-0.155	0.00920

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

<u>ALFA</u>	<u>CL</u>	<u>CM</u>	<u>CD</u>	<u>Sta</u>	<u>CL_{EXP}</u>	<u>CM_{EXP}</u>	<u>CD_{EXP}</u>
1.800	0.9831	-0.169	0.00840	0.743	0.8252	-0.130	0.00970

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

$\overline{ALFA}$	$\overline{CL}$	$\overline{CM}$	$\overline{CD}$	$\overline{S_{Ta}}$	$\overline{CL_{Exp}}$	$\overline{CM_{Exp}}$	$\overline{CD_{Exp}}$
-0.200	0.7655	-0.170	0.00894	0.781	0.6596	-0.128	0.00630

2mm step airfoil pressure distribution:

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

<u>ALFA</u>	<u>CL</u>	<u>CM</u>	<u>CD</u>	<u>Str</u>	<u>CL_{EXP}</u>	<u>CM_{EXP}</u>	<u>CD_{EXP}</u>
-8.000	0.0000	0.000	0.00000	0.000	-0.1261	-0.087	0.04640

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

<u>ALFA</u>	<u>CL</u>	<u>CM</u>	<u>CD</u>	<u>Sta</u>	<u>CL_{EXP}</u>	<u>CM_{EXP}</u>	<u>CD_{EXP}</u>
-6.000	0.0000	0.000	0.00000	0.000	0.1218	-0.081	0.02040

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

<u>ALFA</u>	<u>CL</u>	<u>CM</u>	<u>CD</u>	<u>Str</u>	<u>CL_{EXP}</u>	<u>CM_{EXP}</u>	<u>CD_{EXP}</u>
-4.000	0.0000	0.000	0.00000	0.000	0.1906	-0.058	0.01620

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

<u>ALFA</u>	<u>CL</u>	<u>CM</u>	<u>CD</u>	<u>Sta</u>	<u>CL_{EXP}</u>	<u>CM_{EXP}</u>	<u>CD_{EXP}</u>
-2.000	0.5438	-0.162	0.01028	0.755	0.4555	-0.093	0.01150

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

<u>ALFA</u>	<u>CL</u>	<u>CM</u>	<u>CD</u>	<u>Str</u>	<u>CL_{EXP}</u>	<u>CM_{EXP}</u>	<u>CD_{EXP}</u>
-1.000	0.6598	-0.165	0.00800	0.734	0.5221	-0.056	0.10500

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

<u>ALFA</u>	<u>CL</u>	<u>CM</u>	<u>CD</u>	<u>Sta</u>	<u>CL_{EXP}</u>	<u>CM_{EXP}</u>	<u>CD_{EXP}</u>
-8.000	0.0000	0.000	0.00000	0.000	-0.2577	-0.030	0.03310

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

<u>ALFA</u>	<u>CL</u>	<u>CM</u>	<u>CD</u>	<u>Sta</u>	<u>CL_{EXP}</u>	<u>CM_{EXP}</u>	<u>CD_{EXP}</u>
0.000	0.7645	-0.163	0.00761	0.693	0.6048	-0.076	0.00980

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

<u>ALFA</u>	<u>CL</u>	<u>CM</u>	<u>CD</u>	<u>Str</u>	<u>CL_{EXP}</u>	<u>CM_{EXP}</u>	<u>CD_{EXP}</u>
1.000	0.8617	-0.160	0.00782	0.602	0.7281	-0.082	0.00980

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

<u>ALFA</u>	<u>CL</u>	<u>CM</u>	<u>CD</u>	<u>Str</u>	<u>CL_{EXP}</u>	<u>CM_{EXP}</u>	<u>CD_{EXP}</u>
3.000	1.0727	-0.159	0.01026	0.563	0.9398	-0.067	0.01380

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

<u>ALFA</u>	<u>CL</u>	<u>CM</u>	<u>CD</u>	<u>Sin</u>	<u>CL_{EXP}</u>	<u>CM_{EXP}</u>	<u>CD_{EXP}</u>
5.000	1.2565	-0.152	0.01182	0.446	1.1554	-0.072	0.01750

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

<u>ALFA</u>	<u>CL</u>	<u>CM</u>	<u>CD</u>	<u>Str</u>	<u>CL_{EXP}</u>	<u>CM_{EXP}</u>	<u>CD_{EXP}</u>
7.000	1.2931	-0.125	0.02203	0.011	1.0713	-0.081	0.02230

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

<u>ALFA</u>	<u>CL</u>	<u>CM</u>	<u>CD</u>	<u>Sta</u>	<u>CL_{EXP}</u>	<u>CM_{EXP}</u>	<u>CD_{EXP}</u>
9.000	1.3204	-0.102	0.03656	0.006	0.9023	-0.094	0.00000

1mm step airfoil pressure distribution:

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

<u>ALFA</u>	<u>CL</u>	<u>CM</u>	<u>CD</u>	<u>Stn</u>	<u>CL_{EXP}</u>	<u>CM_{EXP}</u>	<u>CD_{EXP}</u>
-10.000	0.0000	0.000	0.00000	0.000	-0.2546	-0.024	0.03310

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

<u>ALFA</u>	<u>CL</u>	<u>CM</u>	<u>CD</u>	<u>STR</u>	<u>CL_{EXP}</u>	<u>CM_{EXP}</u>	<u>CD_{EXP}</u>
-6.000	0.0000	0.000	0.00000	0.000	0.0454	-0.101	0.01650

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

<u>ALFA</u>	<u>CL</u>	<u>CM</u>	<u>CD</u>	<u>Str</u>	<u>CL_{EXP}</u>	<u>CM_{EXP}</u>	<u>CD_{EXP}</u>
-4.000	0.0000	0.000	0.00000	0.000	0.3183	-0.104	0.01230

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

<u>ALFA</u>	<u>CL</u>	<u>CM</u>	<u>CD</u>	<u>Sta</u>	<u>CL_{EXP}</u>	<u>CM_{EXP}</u>	<u>CD_{EXP}</u>
-3.000	0.0000	0.000	0.00000	0.000	0.3981	-0.102	0.01130

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

<u>ALFA</u>	<u>CL</u>	<u>CM</u>	<u>CD</u>	<u>Stn</u>	<u>CL_{EXP}</u>	<u>CM_{EXP}</u>	<u>CD_{EXP}</u>
-2.000	0.5542	-0.166	0.01143	0.784	0.5264	-0.104	0.00870

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

<u>ALFA</u>	<u>CL</u>	<u>CM</u>	<u>CD</u>	<u>Sta</u>	<u>CL_{EXP}</u>	<u>CM_{EXP}</u>	<u>CD_{EXP}</u>
-1.000	0.6757	-0.169	0.00881	0.770	0.5826	-0.118	0.00780

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

<u>ALFA</u>	<u>CL</u>	<u>CM</u>	<u>CD</u>	<u>Sta</u>	<u>CL_{EXP}</u>	<u>CM_{EXP}</u>	<u>CD_{EXP}</u>
0.000	0.7826	-0.168	0.00805	0.754	0.6310	-0.106	0.00780

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

<u>ALFA</u>	<u>CL</u>	<u>CM</u>	<u>CD</u>	<u>STN</u>	<u>CLEXP</u>	<u>CMEXP</u>	<u>CDEXP</u>
1.000	0.8888	-0.166	0.00775	0.732	0.7212	-0.099	0.00750

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

<u>ALFA</u>	<u>CL</u>	<u>CM</u>	<u>CD</u>	<u>Sta</u>	<u>CL_{EXP}</u>	<u>CM_{EXP}</u>	<u>CD_{EXP}</u>
4.000	1.1904	-0.161	0.01024	0.549	0.9783	-0.080	0.01810

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000

<u>ALFA.</u>	<u>CL</u>	<u>CM</u>	<u>CD</u>	<u>Stn</u>	<u>CL_{EXP}</u>	<u>CM_{EXP}</u>	<u>CD_{EXP}</u>
7.000	1.3828	-0.140	0.01740	0.232	1.0851	-0.097	0.02380

LRN-010

MACH = 0.100

RE = 0.380×10^6

NCRIT = 7.000