THICK SUPERCRITICAL AIRFOILS WITH LOW DRAG AND NLF CAPABILITY*

B. Eggleston, Manager Advanced Design, DHC
D.J. Jones, Senior Research Officer, NRC/NAE
R.J.D. Poole, Senior Research Engineer, DHC
M. Khalid, Associate Research Officer, NRC/NAE

ICAS-86-1.1.1

ABSTRACT

A family of NLF-capable, supercritical airfoils has been developed which was designed to have good characteristics with turbulent flow while being capable of supporting up to 70% chord natural laminar flow in favourable conditions. Tests are reported on airfoils of 16% and 21% chord maximum thicknesses in the NAE 0.38m x 1.52m two-dimensional wind tunnel. At Reynolds numbers up to 10 million, and over a narrow range of Mach numbers close to their design conditions, the airfoils supported extensive laminar flow and their drag was reduced more than 50% relative to values with turbulent boundary layers. At higher Reynolds numbers, the NLF capability diminished rapidly due to reduced boundary layer stability and other factors. The airfoil cruise drag characteristics at all Reynolds numbers tested (8 million to 20 million) were markedly superior to airfoils of similar thicknesses tested previously in the same facility.

INTRODUCTION

Future transport aircraft will benefit from improved airfoil designs that reduce wing section drag. Towards that objective the de Havilland Aircraft Company of Canada and the National Aeronautical Establishment of the National Research Council of Canada have an on-going program of R&D which is aimed at developing improved supercritical airfoils suitable for future commuter transport aircraft, possibly using some form of advanced propellers to cruise at high subsonic speeds. The purpose of this paper is to present some experimental results for new designs of thick supercritical airfoils that are capable of supporting extensive natural laminar flow (NLF) in suitably favourable conditions. This will provide substantial reductions in drag or alternatively allow substantial increases in wing thickness.

When designing these airfoils, one of the foremost objectives was to ensure that good aerodynamic characteristics were retained when the boundary layers were made turbulent from near the noses of the airfoils. This situation can be expected to prevail frequently, for premature transition will often be caused by some type of surface contamination or free stream turbulence. However, in circumstances where conditions are sufficiently "clean", the NLF-capable airfoils can support laminar flow back to 60%-70% chord on their upper surfaces and up to 50% chord on their lower surfaces. This extent of laminar flow will lead to drag reductions of at least 50% relative to the same airfoils with turbulent boundary layers.

Validating the performance of NLF-capable airfoils at high Reynolds numbers becomes difficult in wind tunnels, due to the need for very low turbulence levels and for extremely fine tolerances on small models. Some NASA data from (1), indicate that turbulence levels below 0.1% may be needed at the Reynolds numbers of 10 million to 15 million appropriate to commuter aircraft. Measurements made in 1978 by Elfstrom in the NAE 0.38m x 1.52m wind tunnel (2) showed turbulence levels were about 0.3% at Reynolds numbers of 14 million to 20 million, suggesting there was little prospect for extensive NLF. However, subsequent tests (3) showed evidence of NLF back to 40% chord on an airfoil with a flat pressure distribution at a Reynolds number of 14 million. As a consequence, at the outset of this investigation the prospects for achieving extensive NLF were uncertain. This points out the need for further measurements of tunnel turbulence and on the theoretical side the development of reliable criteria for predicting transition.

Two airfoils were designed and tested of 16% and 21% chord maximum thickness and their geometries are shown in Figure 1. The design conditions for the airfoils are summarised in Table 1. The table also includes estimates of the drag rise Mach numbers with turbulent boundary layers which were made using an extended version of the BGK code of (4). The design lift coefficient, C_{LDES}, selected for the airfoils was 0.6 to suit aircraft operating over long ranges and at high altitudes. It is noted that the C_{LDES} used is much higher than considered in (5) where a value of 0.35 was utilized and lower camber airfoils resulted.

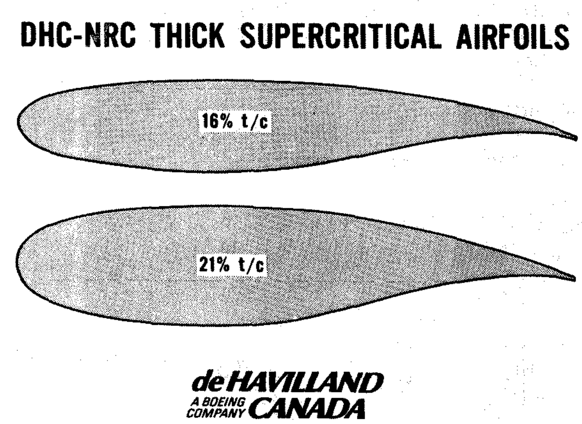


FIGURE 1

* This R&D was jointly supported by the de Havilland Aircraft Company of Canada (DHC), a Division of Boeing of Canada Ltd., Downsview, Ontario, Canada, and by the PILP Office of the National Research Council of Canada (NRC).

DESIGN CONDITIONS

$t/c_{MAX}\%$	16	21
M_{DES}	0.72	0.68
C_{LDES}	0.60	0.60
$M_{DRAG\ RISE}$*	0.74	0.69

*FROM REF 4 WITH Re_c 20 MILLION

TABLE 1

Some features of the design velocity distribution are summarized in Figure 2. At the design condition the peak Mach numbers on the upper surface were kept to 1.12 - 1.15 to give low wave drag, and the velocity gradients were made slightly favourable to encourage NLF. A moderate amount of aft loading was used to enhance performance, subject to the constraint that pitching moment coefficients should not be less than -0.14 at the design condition. Boundary layer calculations for both airfoils with fully turbulent conditions showed good margins from separation, on both surfaces, at their respective design conditions.

The airfoils were tested in the NAE 0.38m x 1.52m facility over a range of Reynolds numbers from 8 million to 20 million per foot. The remainder of this paper will review selected results from tests of the 16% and 21% thick airfoils, concentrating mainly on the airfoil drag characteristics. Further information on the tests of the 21% thick airfoil are given in (6). Airfoils of 10% and 13% thicknesses have also been designed and tests are planned later this year to complete the investigations.

FEATURES OF DESIGN PRESSURE DISTRIBUTIONS
BASED ON BGK (EXTENDED) PREDICTIONS

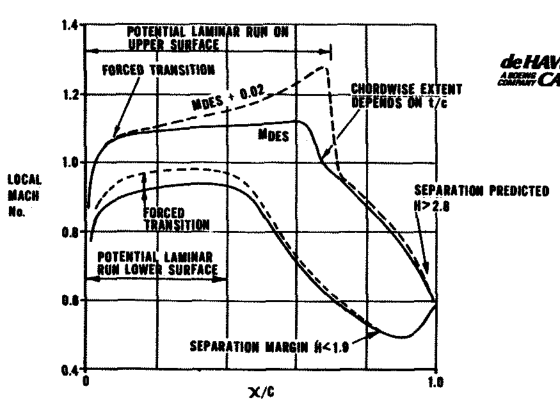


FIGURE 2

FLOW VISUALISATION

The flow visualisation was performed by spraying the airfoil with a thin film of oil containing a dye which fluoresced in ultra-violet light. The application required some care as a film that was too thick could cause premature transition, which altered the airfoil aerodynamic characteristics markedly at low Reynolds numbers.

Photographs of the flow visualisation results for the 16% thick airfoil at a lift coefficient of 0.6 and Mach numbers about 0.75, for several Reynolds numbers are shown in Figures 3, 4 and 5. At a Reynolds number of 8 million most of the upper surface showed stable laminar flow back to 70% chord which coincided with the shock location.

FLOW VISUALIZATION AT Re 8 MILLION
M 0.75 AND C_L = 0.6

1 TRANSITION WEDGE DUE TO LINE OF PRESSURE TAPS
2 TRANSITION WEDGES DUE TO TO SPECKS OF DIRT
3 LAMINAR RUN BACK TO 70% CHORD
4 TURBULENT FLOW IN PRESSURE RECOVERY REGION
5 SHOCK LOCATION IN TURBULENT FLOW
6 PRESSURE TAPPINGS OFFSET TO REDUCE EFFECTS OF TRANSITION ON PROBE No. 1

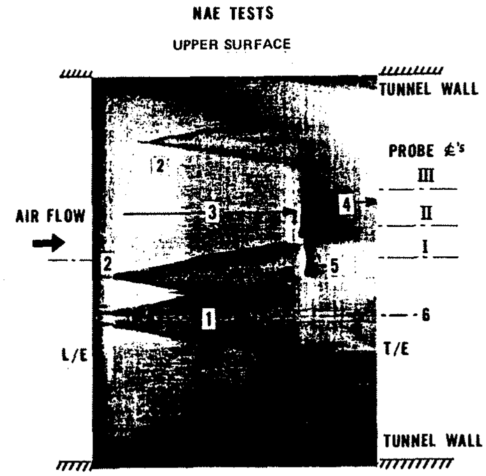


FIGURE 3

16% AIRFOIL FLOW VISUALIZATION
ON UPPER SURFACE TRANSITION FREE C_L = 0.6
Re_C = 14M M = 0.76

FIGURE 4

16% AIRFOIL FLOW VISUALIZATION ON UPPER SURFACE TRANSITION FREE C_L = 0.6

Re_c = 20M M = 0.74

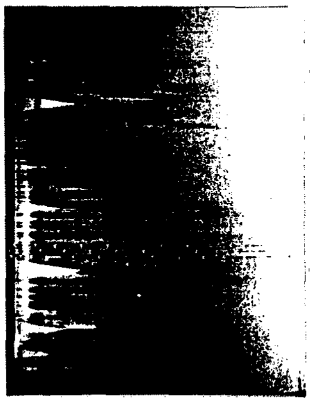
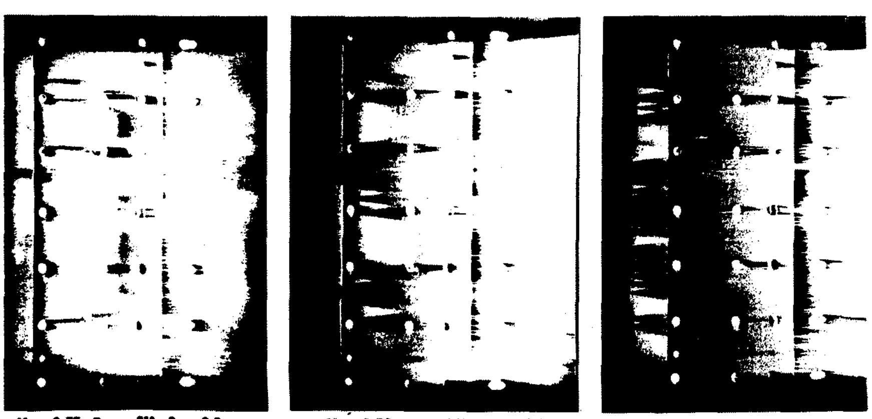


FIGURE 5

The disturbances due to the pressure taps caused a transition wedge of sufficient spanwise extent to influence the centre-line probe of the wake traverse apparatus. The flow visualisation showed a few regions of transition that were caused by minute particles and they were distributed haphazardly over the surface.

Increasing Reynolds number to 14 million reduced the extent of NLF to 45% chord, at the very most, on the upper surface. The laminar region became very patchy and transition even originated at the leading edge in several places. The further increase of Reynolds number to 20 million reduced the NLF to 20% chord at best, but the spanwise extent of the coverage was so sparse that the airfoil was essentially fully turbulent in behaviour.

FLOW VISUALIZATION WING LOWER SURFACE



M = 0.75, Re = 8M, C_L = 0.6 | M = 0.76, Re = 14M, C_L = 0.6 | M = 0.74, Re = 20M, C_L = 0.6

FIGURE 6

Flow visualisation results for the lower surface at a Reynolds number of 8 million are shown in Figure 6. The model featured a detachable bottom plate held in place by bolts, and although the bolt holes were filled carefully, they caused sufficient disturbance to trigger transition locally. The lower surface supported NLF to 50% chord between the holes but the reduced spanwise coverage would result in a significant drag increase.

The chordwise location of transition on the 16% airfoil is shown as a function of Reynolds number in Figure 7, as derived from the longest runs of NLF observed on each surface at a lift coefficient of 0.6 and Mach numbers about 0.75. Basing the local Reynolds number on the transition distance showed that Re remained roughly constant at about 5 million throughout. However, as chord Reynolds number increased, various adverse effects such as tunnel turbulence, the reduction in boundary layer stability and surface contamination led to increasingly patchy behaviour on both surfaces of the models.

16% THICK AIRFOIL CHORDWISE LOCATION OF START OF TRANSITION AT MACH 0.74-0.76 C_L = 0.6

NOTES

- BASED ON VISUALISATION USING FLUORESCENT OIL
- VERY STABLE LAMINAR FLOW AT 8M Re
- VERY SUSCEPTIBLE TO MINOR SPECKS OF DIRT AT 14M Re BUT LAMINAR REGIONS WELL DEFINED ON BOTH SURFACES
- AT 20M Re VERY FEW REGIONS OF SUSTAINED LAMINAR FLOW ON UPPER SURFACE. LOWER SURFACE EXTENT LIMITED BY MODEL SPLIT LINE

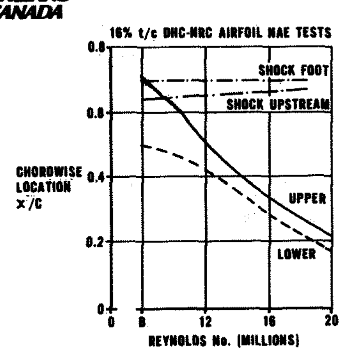


FIGURE 7

DRAG CHARACTERISTICS

The drag of the airfoil section was measured using a traversing wake rake with probes at four spanwise stations. The probe nearest the tunnel wall showed some wall effects so its output was not used. In tests of the airfoils with transition fixed, the other three probes showed very similar drag values, Figure 8, indicating good spanwise uniformity in the tunnel flow. In contrast, the tests with transition free showed spanwise variations in drag indicative of varying amounts of laminar flow ahead of each probe. In the case of the 16% airfoil, even though the pressure taps were displaced off the centre-line, probe #1 appeared to be influenced by the wake spreading from the pressure tapped region, Figure 8.

Accordingly the section drags were finally based on averages of the values from probes #2 and #3 only. In the case of the 21% airfoil the averaging was based on probes #1 and #3 because of a different installation.

16% THICK AIRFOIL SPANWISE DRAG VARIATIONS FOR M = 0.74; C_L = 0.6

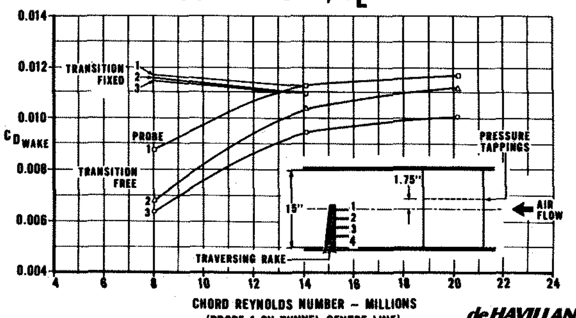


FIGURE 8

Typical drag polars for the two airfoils with NLF are shown in Figure 9, for a Reynolds number per foot of 8 million and at a Mach number of 0.75 for the 16% airfoil and M = 0.68 for the 21%. Both airfoils achieved minimum drag coefficients about 0.005, while the best lift/drag ratios were 109 for the 16% airfoil and 103 for the 21%. These values confirmed the extensive NLF observed in the flow visualisation work.

DRAG POLARS FOR 16 AND 21% THICK AIRFOILS WITH NLF CONDITIONS

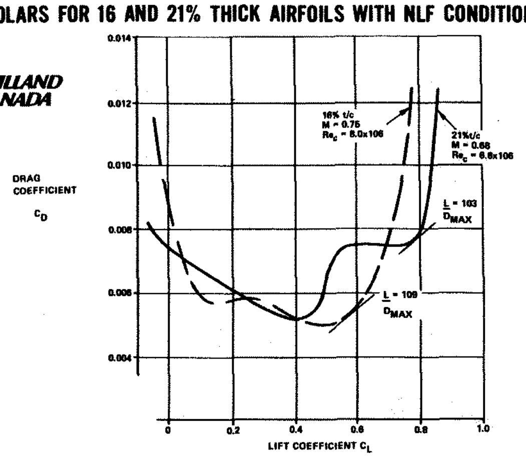


FIGURE 9

The effects of increasing Reynolds number on the drag of the airfoils are presented in Figure 10 for the design lift coefficient of 0.6, again at a Mach number of 0.75 for the 16% airfoil and at M = 0.68 for the 21%. In the case of the 16% thick airfoil the drag coefficient at a Reynolds number of 8 million was 0.0055 but NLF was rapidly lost above a Re_c of 11 million and drag increased progressively. The flow visualisation had shown that fully turbulent behaviour was established by a Re_c of 20 million and the corresponding drag value of 0.0127 was 2.3 times the NLF drag value obtained at a Re_c of 8 million. The behaviour of the 21% thick airfoil was similar in character but the drag increase was smaller and delayed to slightly higher Re_c's, due perhaps to more strongly favourable pressure gradients on the upper surface of this airfoil.

INFLUENCE OF REYNOLDS NUMBER ON SECTION DRAG AT C_L = 0·6 WITH FREE TRANSITION

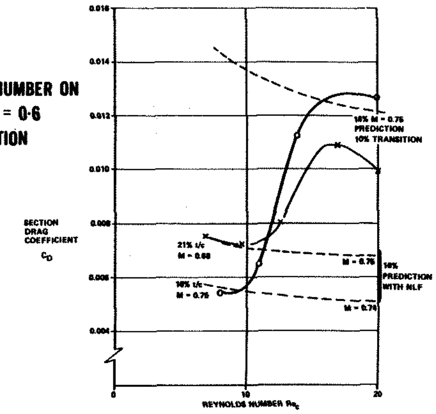


FIGURE 10

The drag estimated for the 16% airfoil at M=0.75 is also included in Figure 10, as predicted by an extended version of the BGK code. The turbulent airfoil values assumed that transition was forced at 10% chord on both surfaces and the predictions agreed well with the measured drag at a Re_c of 20 million. However, the drag predictions for the airfoil with NLF present were higher than measurements at M = 0.75 and the drag rise behaviour appeared to be incorrectly estimated. As a result the predictions made for M = 0.74 gave better agreement with the measurements made at M = 0.75. These discrepancies with NLF conditions suggest that further work is needed on modelling the shock wave interaction with a laminar boundary layer. Transition predictions were also made using the method of van Driest and Blumer (7). Even with zero turbulence levels it was found this method predicted transition at Re_c's lower than observed in practice, suggesting the method was unduly pessimistic.

The influence of increasing Mach number on the drag coefficients of the two airfoils at the design lift coefficient of 0.6 for conditions supporting NLF (Re/ft of 8 million, transition free) is shown in Figure 11. Both airfoils exhibited a deep bucket in their drag curves, centred about M = 0.74 in the case of the 16% air-

INFLUENCE OF MACH NUMBER ON DRAG OF 16 AND 21% t/c AIRFOILS WITH NLF Re/FT 8.1M

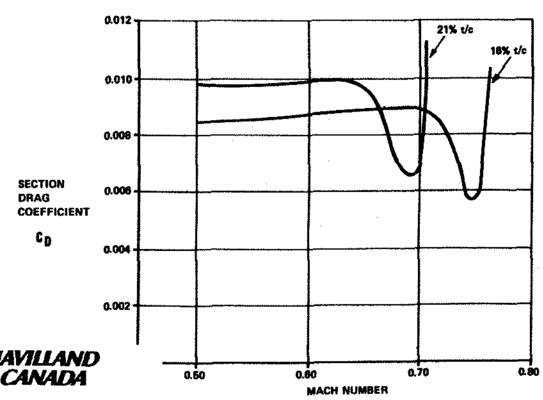


FIGURE 11

foil and about M = 0.69 for the 21% airfoil. There was a narrow spread of Mach number over which the drag reductions due to NLF were significant, suggesting that a variable airfoil geometry may be needed for successful exploitation of this effect over a wider range of Mach numbers.

The variation of drag coefficient with Mach number at high Reynolds numbers giving near fully turbulent behaviour, is included in Figure 12. Both of the new airfoils show very little drag creep and the 16% airfoil demonstrated a drag rise Mach number of 0.75 while the 21% airfoil achieved 0.69.

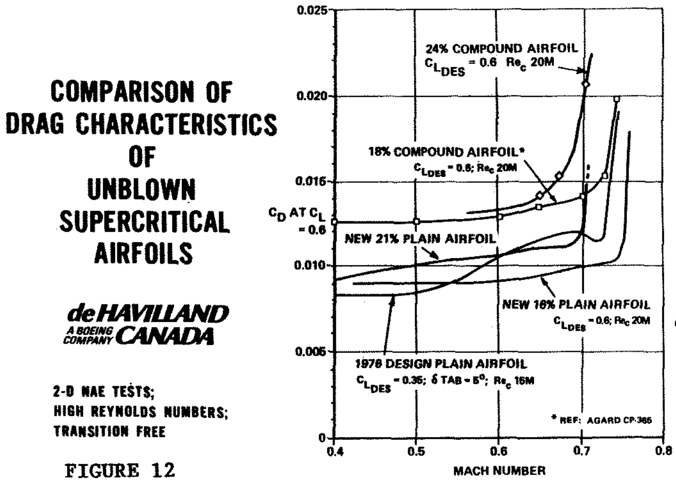


FIGURE 12

COMPARISONS WITH OTHER AIRFOILS

At the present time there is very little published data on NLF airfoils at supercritical conditions so most comparisons have to be made with low speed airfoil test data. NASA has recently developed two modern low speed NLF airfoils designated NLF(1)-0416 (8) and NLF(1)-0415(F) (9) which were tested in the low turbulence wind tunnel at Langley. There is also the earlier NACA work on the 64,65 and 66 series of NLF airfoils reported in (10). Recent NASA tests in the 0.3m cryogenic tunnel at transonic conditions provide some data on a 12% thick supercritical airfoil (11) and 10% airfoil (12) but these only displayed NLF behaviour at lower Re_c (about 4.4 million) than the NRC tests reported herein.

The minimum drag coefficients of the various airfoils with conditions giving NLF are given in Figure 13, while Figure 14 compares the drag data for a lift coefficient of 0.6. Bearing in mind that the drag of the DHC/NRC supercritical airfoils includes some wave drag, that the airfoils have blunt trailing edges and that the lower surfaces have incomplete laminar flow, the drag values achieved appear commendably low.

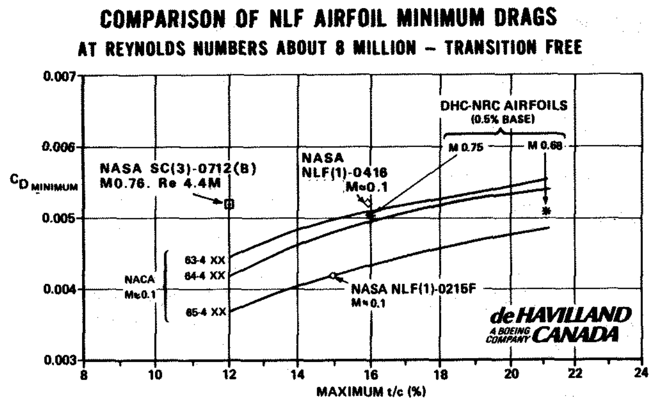


FIGURE 13

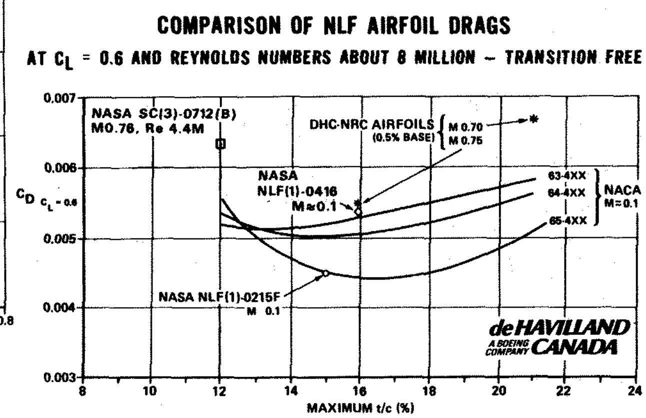


FIGURE 14

The variation of drag with Mach number at high Re_c with near fully turbulent flow, is compared with two other thick supercritical airfoils tested in the NAE facility in Figure 12. The 1976 airfoil was originally designed for a lift coefficient of 0.35 but it was recambered with 5 deg. of flap deflection to improve performance at higher lift. The second comparison is with an unblown compound airfoil of 18% thickness designed for a lift coefficient of 0.6 (13). The new airfoils show substantial improvements in their levels of drag and in their drag rise Mach numbers relative to the previously tested airfoils.

In addition to the previous comparisons, the drag data for the new 16% and 21% thick airfoils have been compared with data from other supercritical airfoils tested in the NAE facility. These data have been gathered over the past 15 years of testing and include for instance the BGK shockless No. 1 and No. 2 airfoils of 11.8% and 15% maximum thickness respectively. These data do not account for any varying Reynolds number effects but the "average" value was probably around 14 million, which was fairly standard for cruise testing in the mid 1970's. Also, during this period the tunnel has undergone several changes, in particular screening of the porous walls and new upstream screens and baffles which should have reduced turbulence and noise in the incoming stream. The work on screened walls reported in (14) shows little effect on the drag of BGK No. 1, but this airfoil might be less sensitive as the velocity gradients are adverse beyond 10% of chord. The drag data are compared at a variety of conditions in Figures 15 to 19, however it should be noted that conditions selected are not necessarily the design points, so the airfoil drags may be higher than designed values.

COMPARISON OF AIRFOILS TESTED AT THE NAE 2-D FACILITY

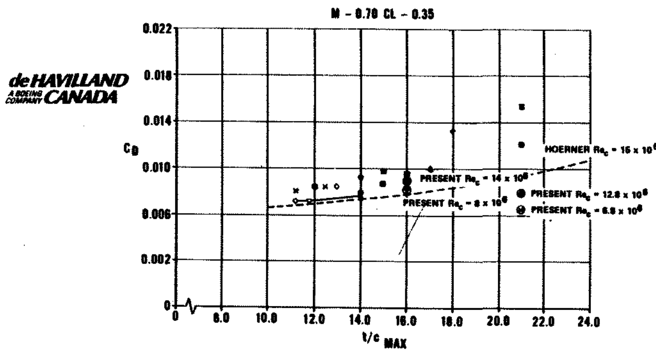


FIGURE 15

COMPARISON OF AIRFOILS TESTED AT THE NAE 2-D FACILITY

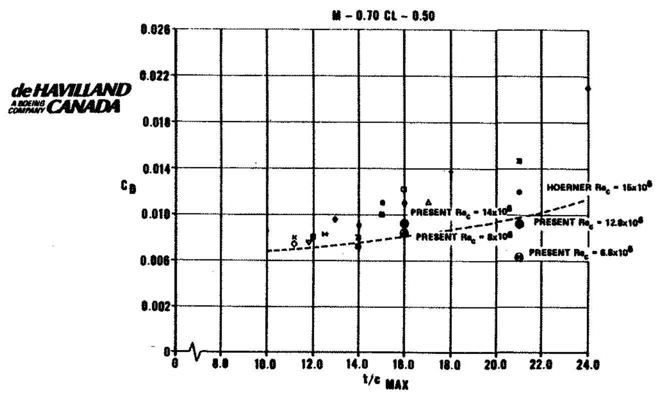


FIGURE 16

COMPARISON OF AIRFOILS TESTED AT THE NAE 2-D FACILITY

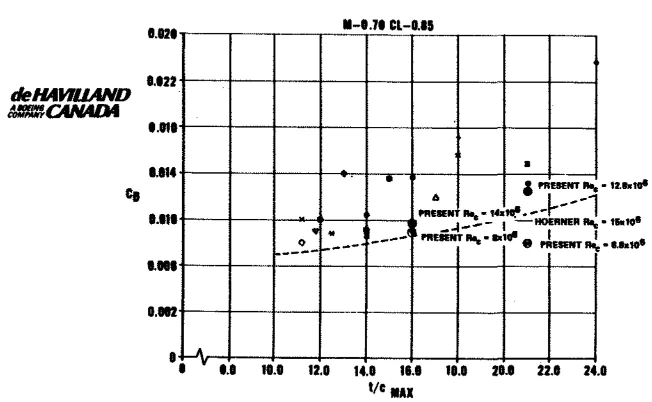


FIGURE 17

COMPARISON OF AIRFOILS TESTED AT THE NAE 2-D FACILITY

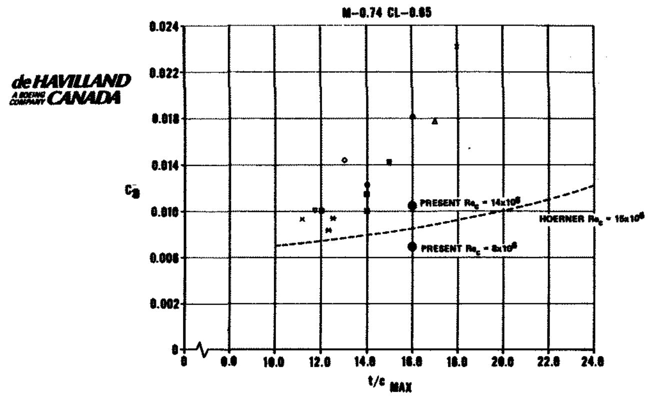


FIGURE 18

COMPARISON OF AIRFOILS TESTED AT THE NAE 2-D FACILITY

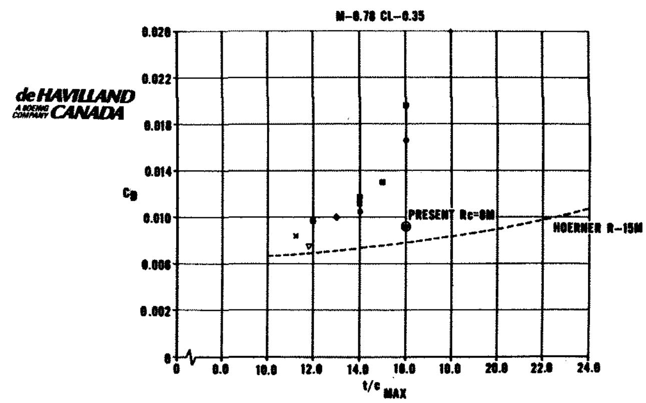


FIGURE 19

Examining the data in Figures 15 to 19, it can be seen that the new airfoils either compare very favourably or are superior to the best airfoils in previous tests. Furthermore, in many instances the new airfoils are better than the reference curve for turbulent flow airfoils derived from Hoerner (15). In previous comparisons this curve has usually represented a lower bound on drag, as it does not include any allowance for wave drag.

CONCLUSIONS

Based on the work to date on NLF-capable airfoils the following can be concluded:

1. It was found possible to demonstrate extensive NLF on airfoils at chord Reynolds numbers of at least 11 million in the NAE facility used for these tests.
2. The design features needed for airfoils to have good characteristics with turbulent boundary layers can be reconciled with good NLF capabilities for the range of thicknesses investigated.
3. The discrepancies observed between drag predictions and test values for cases with NLF show the need to improve methods, possibly in the areas of shock wave/laminar boundary layer interaction and also for transition prediction in the presence of turbulence, roughness and noise.

SYMBOLS

c	airfoil chord
C_L	lift coefficient
C_{LDES}	design lift coefficient
C_D	drag coefficient
C_M	pitching moment coefficient
H	boundary layer shape factor
M	Mach number
M_{DES}	design Mach number
M_{DR}	drag rise Mach number
Re	Reynolds number
Re_c	chord based Reynolds number
t/c	maximum thickness/chord ratio
x	chordwise location from leading edge, positive aft.

ACKNOWLEDGEMENTS

The authors wish to acknowledge the valuable contributions made by F.C. Tang (NRC/NAE) and I.G. Fejtek (DHC) in the wind tunnel testing, data reduction and reporting for these investigations.

REFERENCES

1. Harvey D. - "NASA Supercritical Laminar Flow Control Experiment" Unpublished paper, CASI AGM, Toronto, May 1982

2. Elfstrom, G.M. - "Skin Friction Measurements On Two Relatively Thick Airfoil Sections At High Reynolds Numbers", NRC/NAE Aero Note NAE AN23, November 1984

3. Fancher, M.F. - "Aspects of Cryogenic Wind Tunnel Testing at Douglas", AIAA Paper 82-0606

4. Bauer, F., Garabedian, P. and Korn, D. - "Supercritical Wing Sections III", Lecture Notes in Economics and Mathematical Systems, Springer-Verlag, 1977

5. Eggleston, B., Jones, D.J. and Elfstrom, G.M. - "Development of Modern Airfoil Sections for High Subsonic Cruise Speeds", AIAA Paper 79-0687, Atlantic Conference, March 1979

6. Jones, D.J. and Khalid, M. - "Analysis of Experimental Data for a 21% Thick Natural Laminar Flow Airfoil, NAE 68-060-21:1", Aero Note NAE-AN-34, NRC No. 25076, October 1985

7. Van Driest, E.R. and Blumer, C.B. - "Boundary Layer Transition: Freestream Turbulence and Pressure Gradient Effects", AIAA Journal Vol. 1, No. 6, June 1963

8. Somers, D.M. - "Design and Experimental Results for a Natural-Laminar-Flow Airfoil for General Aviation Applications", NASA TP 1861

9. Somers, D.M. - "Design and Experimental Results for a Flapped Natural Laminar Flow Airfoil for General Aviation Applications", NASA TP1865, June 1981

10. Abbott, I.H. and von Doenhoff, A.E. - "Theory of Wing Sections", Dover Publications, 1959

11. Johnson, W.G., Hill, A.S., Eichmannnn, O. - "High Reynolds Number Tests of a NASA SC(3)-0712(8) in the Langley 0.3 Meter Transonic Cryogenic Tunnel", NASA TM 86371, June 1985

12. Johnson, W.G., Hill, A.S. - "Pressure Distributions From High Reynolds Number Tests of a Boeing BAC 1 Airfoil in the NASA Langley 0.3 Meter Transonic Cryogenic Wind Tunnel", Nasa TM 87600

13. Whittley, D.C. - "An Update on the Canada/USA Augmentor-Wing Project", Paper 10, AGARD CPP365, May 1984

14. Ohman, L.H., Plosenski, M.J. and Tang, F.C. - "Investigation of the Effect of Edgetone Noise on 2-D Test Data for the BGK No. 1 Airfoil", NRC Report LTR-HA-64, 1982

15. Hoerner, S.F. - "Fluid Dynamic Drag", 1965