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NUMERICALLY ASSISTED DESIGN OF HELICOPTER TYPE AIRFOILS

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Abstract

New advanced airfoils for helicopter rotor blades have been numerically designed. Design objectives for the root and tip stations based on the mission requirements are described. The design procedure, which employs a subsonic and a transonic code for analysis/design, is explained. The characteristics of a designed profile with 7.2% thickness are further investigated by using an Euler code. Additional comments are issued based on the experience attained here by designing these airfoils.

1 - Introduction

An airfoil with good aerodynamic characteristics represents the basis of successful rotor design. Further improvement in performance for some helicopter configurations can be achieved by aerodynamic optimization of rotor blade profiles. This is a very complicated task since a large spectrum of conditions with significant effects on aircraft's stability has to be satisfied. The flow is extremely complex, the blade is exposed to rapidly changing angles of attack and yaw in combination with rapidly changing velocities. There are regions of high Mach numbers, stall, and reverse flow. In addition, there are crossflow and unsteady boundary layer effects.

Before an airfoil can be designed one has to know what requirements the new airfoil should meet, and what features are primarily desired. The design requirements are at first fixed by the characteristics of the rotor at high speed conditions including maneuvers. Selecting airfoils for rotorblades, so far simple qualitative comparisons between section shapes are normally made. The profile with the best Cl_{max}-value at moderate Mach numbers (0.40 ~ 0.50) and highest drag divergence Mach number at zero lift was always preferred[1] .

Specifications for the design of a helicopter blade profile concern generally the conditions which are encountered by the airfoil during a revolution of the rotor that is to say advancing blade condition and retreating blade condition. To find the best compromise between these two flight conditions it seems particularly interesting to use numerical techniques. Reneaux[2] coupled the CONMIN[3] optimization routine with an analysis code[4] and obtained 16% lower drag for the advancing blade conditions at the design point concerning a 7% thick airfoil[5] .

The task to obtain new helicopter type airfoils was carried out at issue by employing a full potential conservative analysis/inverse design code[6] , named PROJ2D, and a panel method for viscous/inviscid analysis and mixed-inverse design of subcritical airfoils, XFOIL[7] .

A lot of expertise has been gained during the design process of new airfoils and through comparison of theoretical results with wind tunnel data. The design capabilities from the code PROJ2D were also extensively tested.

2 - Requirements and Design Directives

Besides the fundamental requests of a lower drag coefficient at higher number of Mach and higher max.-lift coefficient at all near subsonic flow, further boundary conditions are important for the helicopter blade design. The assertive of a small moment coefficient at zero lift, Cm_0, at all flow conditions must be pursued. It is essential to have $|Cm_0| \leq 0.010$ in order to minimize the pitch control efforts and twist of the rotor blade

The following flight conditions occur at a typical mission of a light transport helicopter:

- Hovering
- max. cruise
- max. range
- maneuver with load factors > 1.

For this reason all kinds of velocity flow conditions appear which can be specified by rotor tip speed, inclination of the blade tip plane, twist of the rotor blades, flight speed, and other factors. The design requirements used in this work are more rigorous than the ones given in Ref. [8].

Outer airfoil (r/R = 0.95)

max. thickness	$\cong 9\ \%$
pitching moment coefficient Cm_0	≤ 0.010
drag coefficient at Mach 0.60 and Cl=0.60	≤ 0.00750
divergence Mach number ($Cl = \pm\ 0.20$, $\partial CD_w / \partial M = 0.10$)	≥ 0.90
$Cl_{\max}$ at Mach 0.40	≥ 1.40

Inner airfoil (r/R = 0.80)

max. thickness	$\cong 12\%$
pitching moment coefficient Cm_0	≤ 0.010
drag coefficient at Mach 0.60 and Cl=0.60	≤ 0.00800
divergence Mach number ($Cl = \pm 0.20$, $\partial CD_w / \partial M = 0.10$)	≥ 0.80
$Cl_{\max}$ at Mach 0.40	≥ 1.60

In Fig. 1 the shaded circular areas represent the main operational conditions of the blade section at 95% of the rotorspan. The design objectives are also illustrated.

3 - Numerical Tools

An efficient technique for profile aerodynamic design[6] was employed for the design at transonic flows. This is an inverse method coupled to an analysis code, which solves the full potential equation in conservative form. Given a starting geometry and a desired Cp distribution a new geometry can be obtained typically in 20~30 design cycles even if moderate up to strong shock waves are part of the final or starting Cp distribution. We have performed with this code some design cases that were considered converged for engineering purposes after the 10^{th} cycle. The XFOIL code[7] was used for some high lift designs at subsonic or all near subsonic speeds, for viscous analysis of the designed geometry, and for estimation of $Cl_{\max}$. By a Karman-Tsien compressibility correction it is possible to calculate compressible subcritical flow around airfoils. Fig. 2 shows a comparison of calculated Cp distributions for the airfoil named here HX8015N13 at a free stream Mach number of 0.70 and zero degree angle of attack. The flow at this condition is slightly supersonic in a small region near the leading-edge and the rear portion of the airfoil generates no lift. The non-conservative TRANSEP code[9] calculated a different shape for the lower-side distribution near the leading edge. The XFOIL and PROJ2D codes are in very good agreement , except for some small differences around the lower side leading edge region. Both TRANSEP and XFOIL Cp distributions refer to no boundary layer calculation. In Fig. 3 two Cp distributions are portrayed for the same airfoil as in Fig. 2 at higher Mach number. We can not employ the XFOIL code because its compressibility correction does not enable supercritical flow calculations with shock waves. Again the upper-side Cp distributions agree very well but there are significant differences between both curves at the lower-side leading-edge region.

4 - Some Considerations about Cp Distributions for Helicopter Type Airfoils

The first generation of helicopter rotors were built with symmetric airfoils, which offer a null pitching moment at zero lift, like the NACA0012 profile. The use of cambered sections improved the rotor performance in hover and in forward flight with increased blade and control loads. A typical and largely employed airfoil in this class is the NACA23012. Fig. 4 shows the calculated Cp distribution for this airfoil at a transonic condition where shock waves are present on either side. A very strong suction peak can be realized at the leading-edge region of the lower-side. This suction peak is accepted in order to attain a favorable Cm_0. Indeed, it can be noted there are two strong shock waves on the airfoil's lower-side with consequent drag penalties. Although the NACA23012 has a relative good high lift behavior , its divergence Mach number is very low. Starting from a symmetric profile we made several attempts to obtain an airfoil with a higher divergence Mach number with an acceptable Cl_{max} Our approach was to get out a geometry from a prescribed Cp distribution without such strong suction peak and shock waves. This is a very hard task, since a geometry with lower wave drag can easily be obtained at the design point (see example in Fig. 5) whereas the off-design characteristics are normally undesirable. The design approach must also take in account the physical flow features. Some authors[10,11,12] have addressed this problematic. Some of them can be shortly summarized here:

• minimize the shock wave strength by

→ small contour curvature in the regions of supersonic flow in the cases of low lift and high number of Mach as well as in the case of high lift around Mach 0.40

→ avoiding high contour gradients in front of and at the beginning of supersonic flow regions in order to get out a low level of local number of Mach

• Higher Cl_{max} at Mach = 0.40 by reducing the max. velocity at the leading edge

• Low drag at Mach=0.60 and $Cl \cong 0.60$ by extending the laminar flow regions especially on lower side (other requirements make it not possible on upper-side)

• Lower side front loading and reflected meanline near the trailing edge to reduce Cm_0.

• Use Tab in order to shift the aerodynamic center backwards and to reduce the band-with of Cm_0 values.

5 - Results

A large family of helicopter type airfoils with thickness ≤ 9% has been obtained after several design attempts by employing the codes XFOIL and PROJ2D. In order to avoid supercritical flow regions, some designs were also carried out with XFOIL at a Mach number as high as 0.77. Unfortunately, when we submitted the designed geometries to flow analysis by using the TRANSEP and PROJ2D codes, the expected improvement of the divergence Mach number was not able to be attained. Some geometries obtained by this way presented very strong shock waves at higher Mach numbers. Otherwise, after designing some airfoils with the code PROJ2D, misalignment of the points employed to describe the geometries generated twin suction peaks on the leading edge region when employing XFOIL at high angle of attack. That was not realized in the Cp distributions calculated by the TRANSEP and PROJ2D at same flow conditions. To solve this problem, it was carried out interactive design with XFOIL attempting to preserve the good behavior at transonic flow attained by the input geometries. That poses no great problem, since a useful feature from XFOIL is the mixed-inverse formulation which permits the user to exercise an absolute control of the geometry.

The comparison of the new airfoils with the DMH4 and the two NACA profiles with reference to max. lift coefficient vers divergence Mach number at zero lift (DMN) is plotted in Fig. 6. The convergence process to obtain the airfoil which presents the higher divergence Mach number, the HX7215N16, is illustrated in Fig. 7, a severe design case which needed a large amount of relofting. The design process was interrupted after the 40^{th} cycle because there are only minor differences between the prescribed and calculated pressure distributions. Fig. 8 shows a comparison of the starting and designed geometries for this case. The obtained airfoil is 7.2 % thick and has an entirely new leading edge look. The spatial isobar pattern for this profile at the design point can be seen in Fig. 9, the dark bubble representing the supersonic flow zone. The calculated Cl_{max} at number of Mach of 0.40 and Reynolds number of 8.10^6 by the code XFOIL considering free transition for the airfoil HX7215N16 is 1.50, for the DMH4 type airfoil 1.82, and for the HX8116N15 profile 1.60, the highest from the designed geometries. However, experimental measurements conducted for the DMH4[13] at the same conditions as above mentioned revealed a max. lift coefficient of 1.53. This is 16% lower as expected from the calculations with XFOIL. Supposed the same deviation for the computed Cl_{max}, the HX7215N16 and HX8116N15 airfoils will have 1.26 and 1.34 max. lift coefficient , respectively. According Ref. [13] the measured DMN at zero lift for the DMH4 airfoil is 0.80, 0.040 higher than the computed value with PROJ2D. Ref. [13] gives for the NACA23012 airfoil a DMN of 0.77, again 0.035 higher than the calculated value. This can be explained partly by the fact that PROJ2D is a conservative code and the calculations were carried out considering no viscous flow or boundary layer influence on the results. This way, we can expect a DMN for the HX7215N16 airfoil as high as 0.875, compatible with most profiles of this thickness. In order to confirm these expectations, the wave drag for a DMH4 type airfoil and for the designed geometry HX7215N16 were calculated by employing the ISES Euler code[14] for several Mach numbers (Fig. 10). The derived divergence Mach numbers for both profiles are 0.795 and 0.875, respectively. The twist around Mach 0.83 in the curve of the new geometry, as can be seen in Fig. 10, reflects the successful design effort to achieve a higher divergence Mach number. Fig. 11 shows the drag evolution to Mach number for the HX7215N16 airfoil. This calculation was performed by using the ISES code at Reynolds number of 8.10^6. The viscous drag has practically remained unchanged for the Mach number range and Cl variation illustrated in Fig. 11. As can be observed in Fig. 12, the Cm_0 shows a good behavior to Mach number and the calculated values meet mainly the requirements.

6 - Conclusions

A family of helicopter type airfoils with thickness ≤ 9% was designed by using numerical tools. The profiles have an expected high divergence Mach number and good Cm_{le} behavior but the high lift characteristics must be yet investigated by wind tunnel tests. The combination of a subsonic and an efficient transonic analysis/design code has been proved useful and practical for rotor airfoil development. ISES but not PROJ2D can accurately predict the divergence Mach number of the airfoils considered in this work. The XFOIL calculations of the max. lift coefficient are up to 16% overestimated. The flat upper side of the HX7215N16 airfoil can lead to some instable behavior in unsteady flow, observed for some like-look geometries. Further work will be carried out to obtain a more curved surface while maintaining the good characteristics attained by the numerical design already performed.

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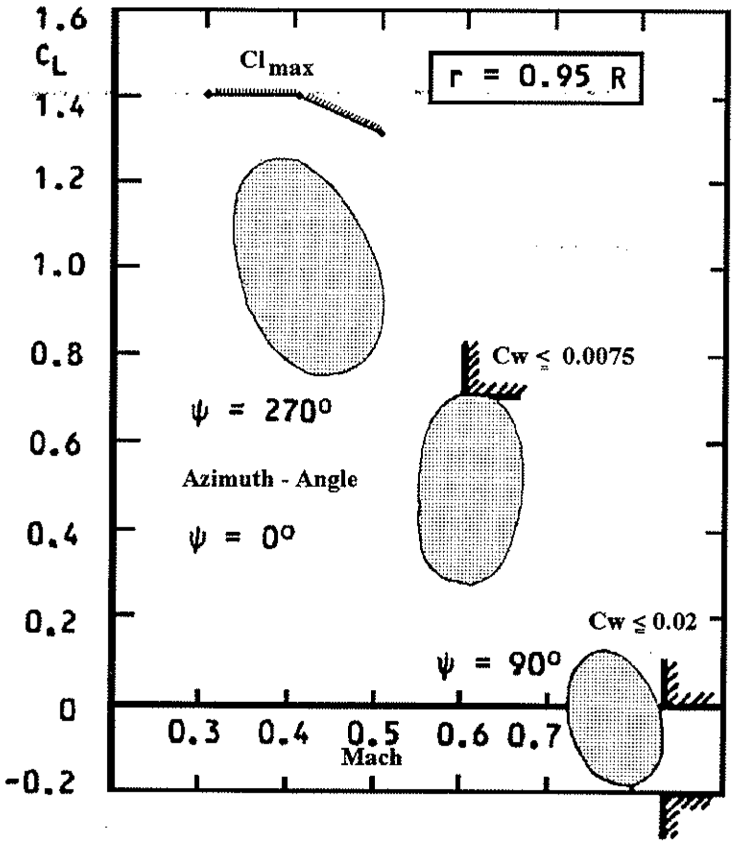

Fig. 1 - Objectives and Main Operational Conditions for the Tip Airfoil.

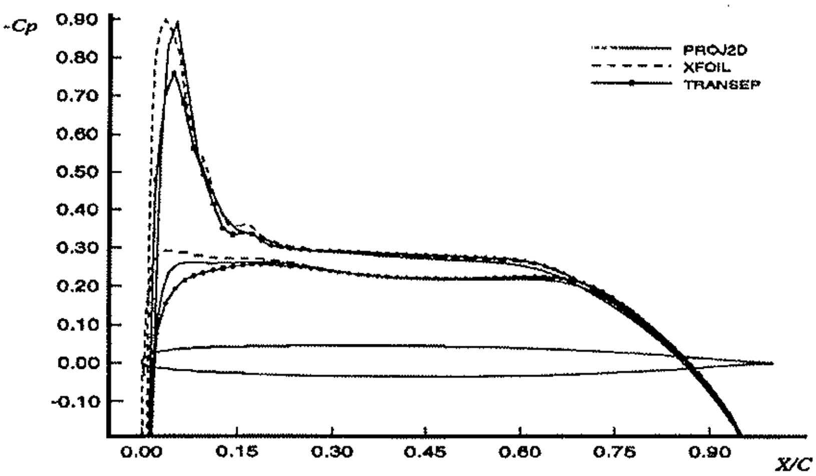

Fig. 2 - Comparison of Cp Distributions at Mach = 0.70, α = 0° for the HX7215N16 Profile.

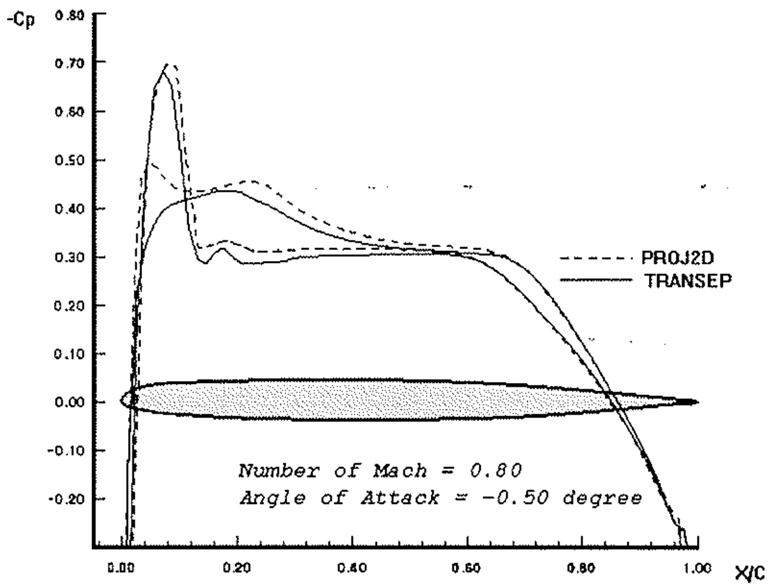

Fig. 3 - Calculated Cp Distributions for the HX8015N13 Airfoil. Mach Number of 0.80, $\alpha = -0.50°$.

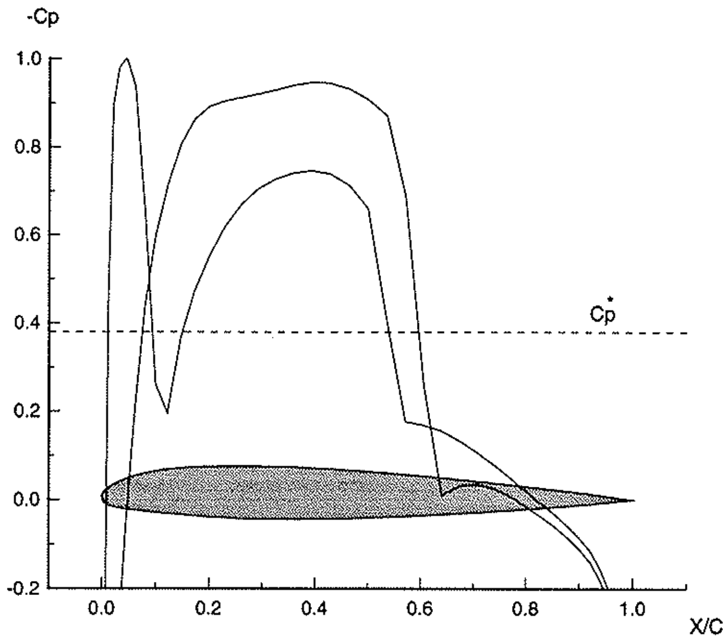

Fig. 4 - Cp Distribution for the NACA23012 Airfoil at Mach Number of 0.82, $\alpha = -1°$.

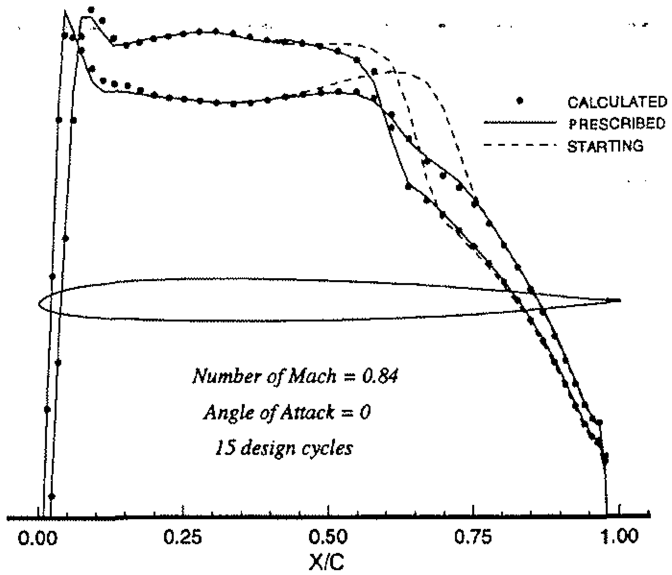

Fig. 5 - Attempt to Design an Airfoil with higher Divergence Mach Number.

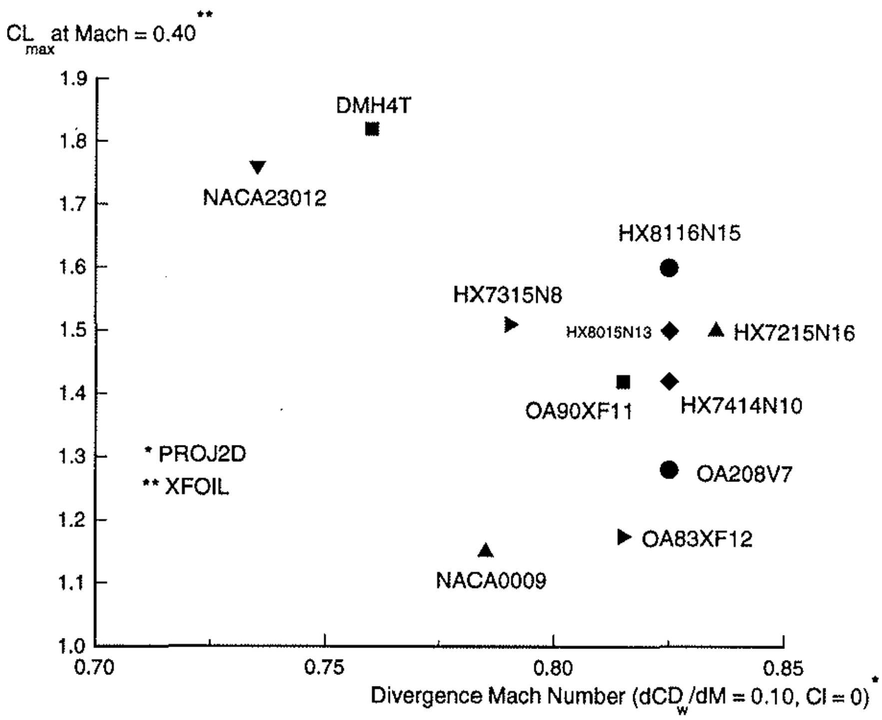

Fig. 6 - Calculated max. Lift Coefficient at Mach Number of 0.40 vs. Divergence Mach Number at Zero Lift for several Rotor Blade airfoils.

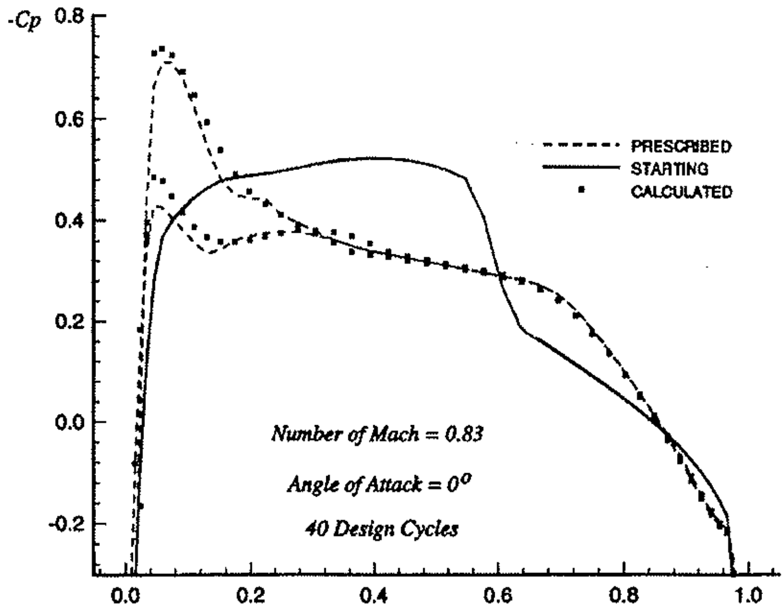

Fig. 7 - Cp Distributions concerning to HX7215N16 Design Case.

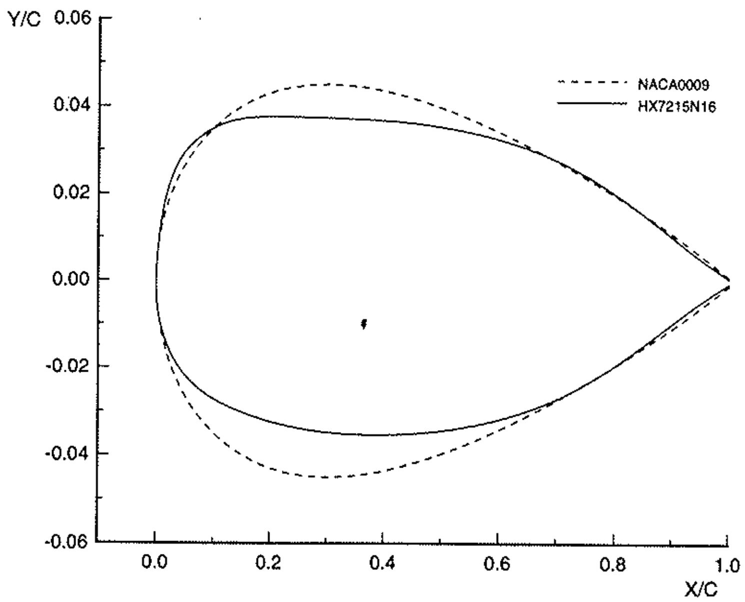

Fig. 8 - HX7215N16 Airfoil compared with the NACA0009 Profile.

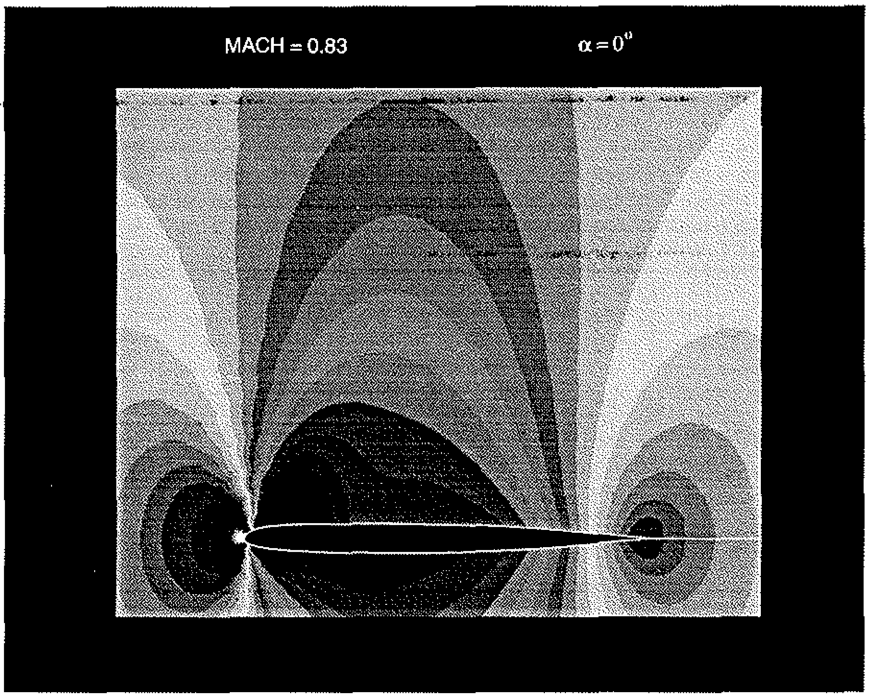

Fig. 9 - Isobar Pattern at the Design Point for the HX7215N16 Airfoil.

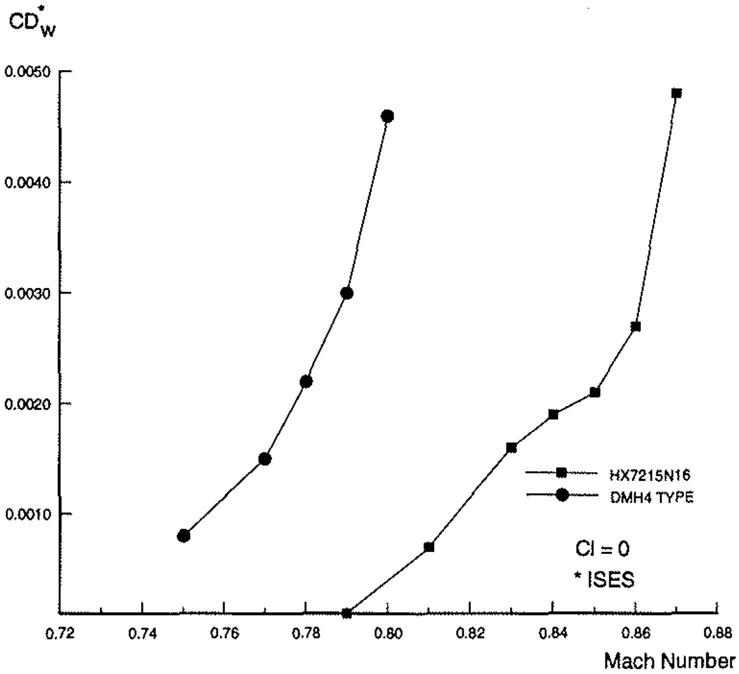

Fig. 10 - ISES Wave Drag calculations.

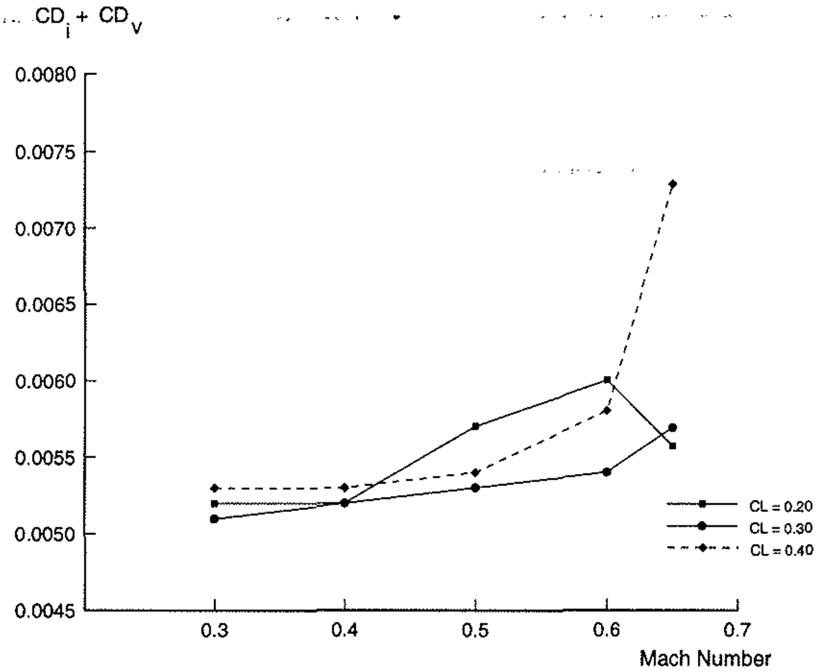

Fig 11 - Drag Evolution for the HX7215N16 Airfoil.
Mach = 0.40, α = 0°.

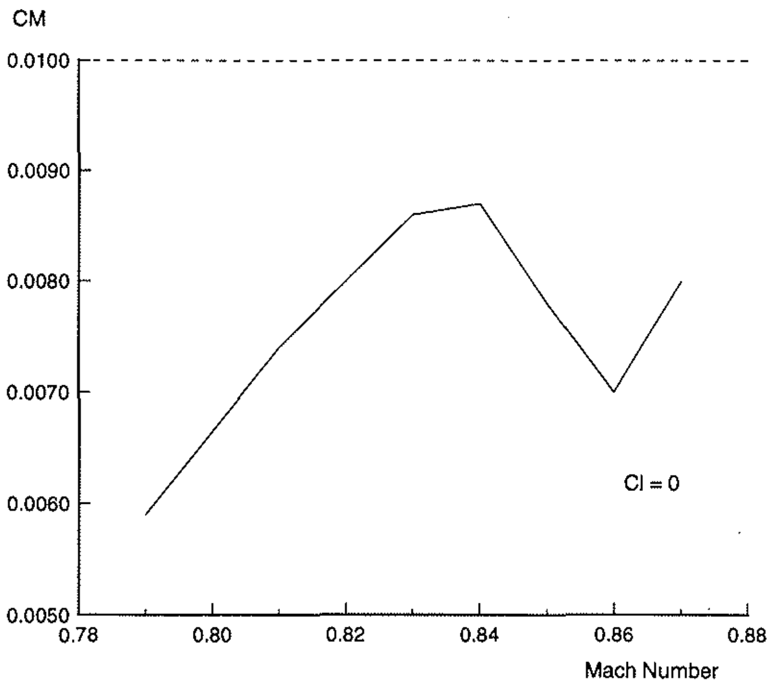

Fig 12 - Cm Evolution for the HX7215N16 Airfoil.
ISES code.