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AERODYNAMIC PROBLEMS OF HELICOPTER BLADE TIPS

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AERODYNAMIC PROBLEMS OF HELICOPTER BLADE TIPS *

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Abstract

ONERA endeavours to acquire an in-depth knowledge of the aerodynamics of helicopter blade tips, whose role is fundamental for both hovering flight and fast forward flight. The studies are carried out in wind tunnels on wall mounted half wings and on rotor models.

Results obtained on blades with straight or 30°-degree-swept tips are presented, in particular those concerning the three-dimensional and unsteady aspects of the flows over these blade tips in zero lift configuration at high speed. Information is given on computer programmes (either available or in course of development) for predicting these types of flows.

Some results of an experimental investigation on phenomena related to vortex interaction are also presented.

1 - INTRODUCTION

Whereas the aerodynamic performance of inboard parts of helicopter blades are now rather well known and can be reasonably well predicted by calculation, it is not the same for the blade sections near the tip, though these play an essential role :

- in hover flight, because of the tip vortex which perturbs the aerodynamic field of the following blade, and thus influences the rotor efficiency;

- in fast forward flight, for which the advancing blade tip functions in transonic regime, which can be penalizing as regards rotor drag and driving power;

- in the whole flight envelope, as regards noise.

Very important worldwide efforts are devoted for understanding, analyzing and, if possible, predicting the phenomena pertaining to the aerodynamic operation of blade tips. In this field, ONERA undertook basic studies on wall mounted half wings and on rotor models, with a view to obtain detailed data that might be used as a basis for comparisons with calculations of two-or three-dimensional, steady or unsteady flows. These studies, however, constitute only a part of those performed at ONERA on helicopter blades [1], in close cooperation with Aérospatiale Company and also within the framework of an agreement with USAAMRDL, of Ames, California.

* *Study sponsored by Service Technique Aéronautique*

2 - ROTOR AERODYNAMICS

In forward flight, the attack conditions of a helicopter blade section are widely varying with the azimuth of the blade, because of the local combination of the rotating velocity and the forward speed.

Figure 1 shows the Mach numbers and incidences encountered by the Gazelle SA341 helicopter blades during a slight dive at 330 km/h. If the incidences are small for the advancing blade, the Mach numbers are high, especially towards the tip, and there exists a crescent-shaped zone where intense transonic phenomena take place. We shall describe in detail, in Section 5, the effects of the unsteady attack conditions on such flows. For the retreating blades, on the other hand, the low attack velocities are associated with very high incidence angles.

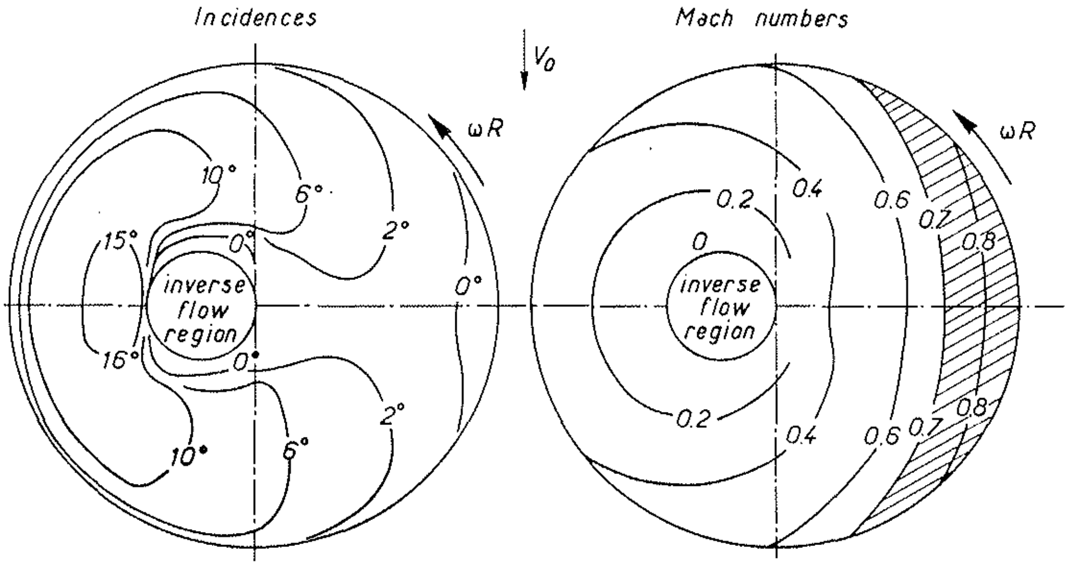


Fig.1 - Local incidences and Mach numbers on SA 341 helicopter
(V_o = 92.5 m/sec. ωR = 208 m/sec.)

Figure 2 shows the iso-sweep angle circles encountered in the same flight condition . This means that the various blade sections are attacked by a flow whose direction varies during the blade rotation (sweep effect).

The incidence angle chart has been traced thanks to a computer programme written at ONERA and based on the resolution of acceleration potential equations for three-dimensional, unsteady, compressible flows [2] .

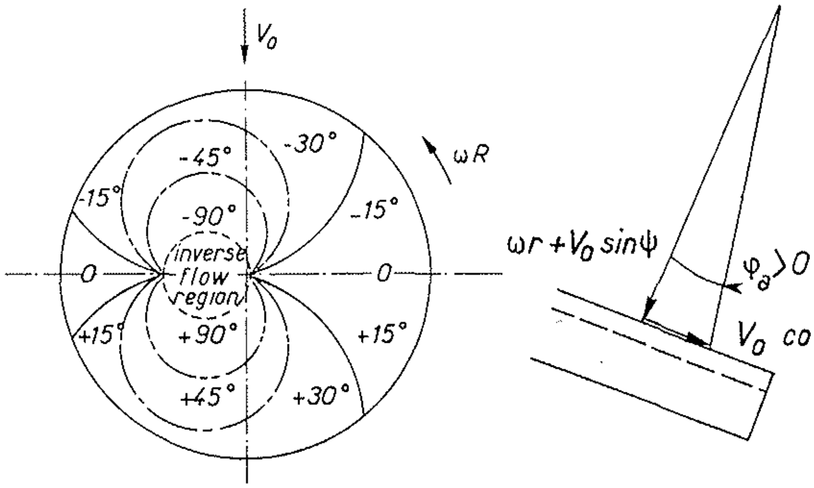


Fig.2 - Iso-sweep angle lines

3 - AERODYNAMIC PARAMETERS RELATED TO BLADE TIP OPTIMIZATION

Three main parameters can be distinguished :

- The blade profiles. It is endeavoured to find profiles having at the same time a high lift at low Mach numbers, in order to remain efficient on the retreating blade, a good L/D at Mach numbers around 0.6 for hover flight, a low drag at low lift at transonic Mach numbers for the advancing blade.

Rotorcraft constructors (Sikorsky, Boeing-Vertol, Aérospatiale) and public research centers (NASA, RAE, NLR, ONERA) defined families of profiles ensuring better compromises than the classical NACA profiles, sometimes still in use, and now try to define blades with profiles varying along the blade span. One such family, developed in close cooperation between ONERA and Aérospatiale, will be described in the next paper [3] of this forum.

- The blade twist. The tendency is at present to increase twist towards -12 to -18°, with a variation, most often linear, becoming sometimes non-linear towards the tip. It is the case, for instance, for the UTTAS of Sikorsky [4] . The highly non-linear twist at the tip may limit the adverse effects of the vortex emitted by the blade.

- The blade planform. Various planforms have been the object of studies all over the world for several years : blade tips with swept leading or trailing edge [5] , with a positive sweep angle followed by 2 or 3 negative sweep angles [6] or trapezoidal [7]. All these forms offer advantages (or sometimes disadvantages) for the flight domain chosen, as regards aerodynamic or acoustic performance of the rotor. More recently, ogee tips, studied by NASA and the US Army at Langley, appeared as very satisfactory even in flight [8] . The selection of a swept tip, either tapered or not, by about 20 to 30° rearwards, may also be beneficial, as proved on the UTTAS and S76 of Sikorsky [4,9] . Generally speaking, these are rather global studies, not allowing a detailed analysis of the actual local phenomena.

ONERA concentrates in gathering precise experimental data on some basic tip forms with a view to validate computing processes which, in turn, should allow an optimization of blade tip shapes.

4 - BASIC STUDIES ON HALF WINGS

These studies aim at ascertaining the influence of a blade tip planform in aerodynamic conditions much simpler than those encountered on a helicopter rotor. The purpose is first to know the total performance of half wings, and the pressure distributions along the span for Mach numbers between 0.3 and 0.9. These models also allow the study of the influence of a vortex near a blade.

4.1 - Straight tip and swept tip

Figures 3 and 4 show the C_L-C_D and C_L-C_m curves for wall mounted half wings having an aspect ratio of 3, one with a classical straight tip and the other with a 30° sweep angle over a span of 0.9 chord, the profile being the NACA 0012 for both models.

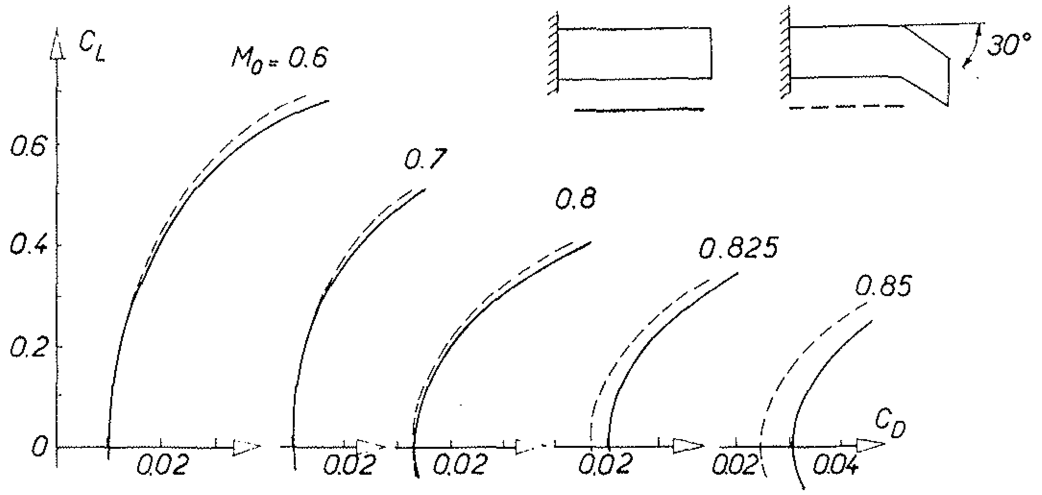


Fig.3 - Tests in S3Ch wind tunnel (C_L, C_D curves)

The influence of a swept tip appears as :

- drag reduction, for a given lift level, which becomes more important as the Mach number increases. We find again the beneficial influence of the sweep angle observed on aircraft wings in transonic flight (fig.3) ;

- pitching moments (relative to an axis at quarter chord) always nose-down for positive lift, whatever the attack Mach number is (fig.4).

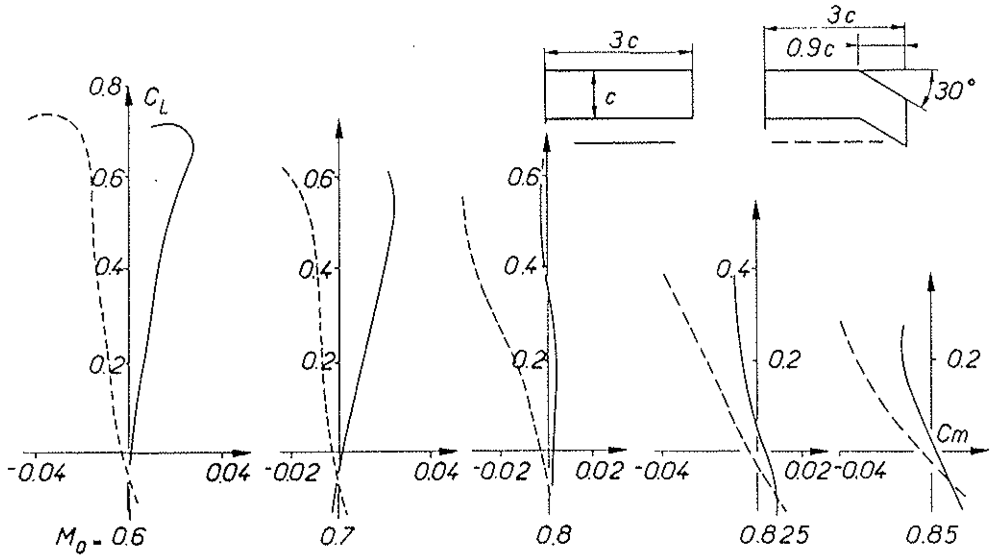


Fig.4 - Tests in S3Ch wind tunnel (C_L, C_m curves)

Twelve chordwise lines of pressure taps equip those models, in order to allow the definition of local flows on the whole half wings, especially near their tips. Figures 5 and 6 present the experimental isobaric lines for Mach 0.85 at zero incidence. The flow on the swept wing is slightly more complex than on the straight tip. The presence of the sweep neatly attenuates the underpressures around the kink of the planform, but there is a new zone of flow acceleration next to the tip.

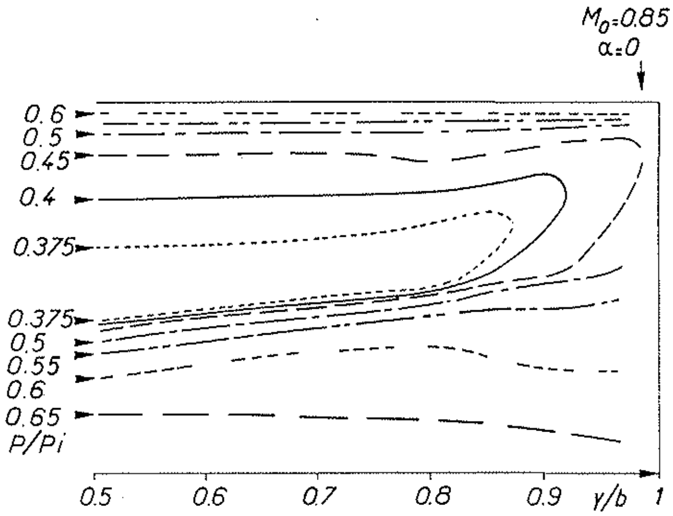


Fig.5 - Experimental isobaric lines on the rectangular tip

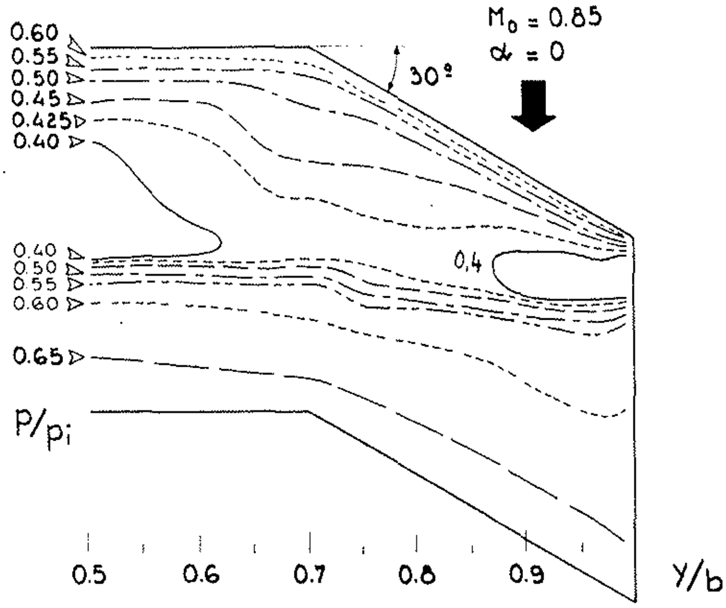


Fig.6 - Experimental isobaric lines on the 30° swept tip

A computer programme, solving the complete velocity potential equation for three-dimensional, transonic steady flows, has recently been assembled by J.J. Chattot and C. Coulombeix of ONERA. Figures 7 and 8 show a quite good agreement between calculated and experimental pressure distributions.

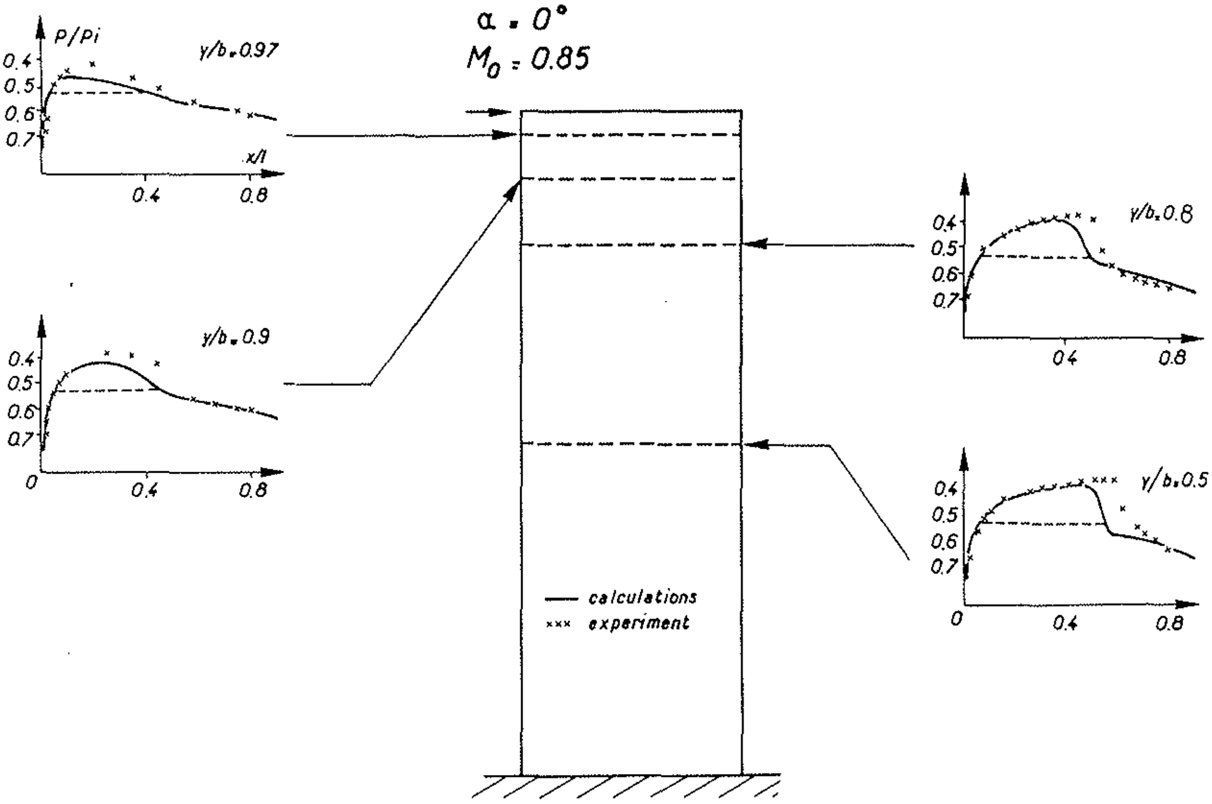


Fig.7 - Straight blade tip

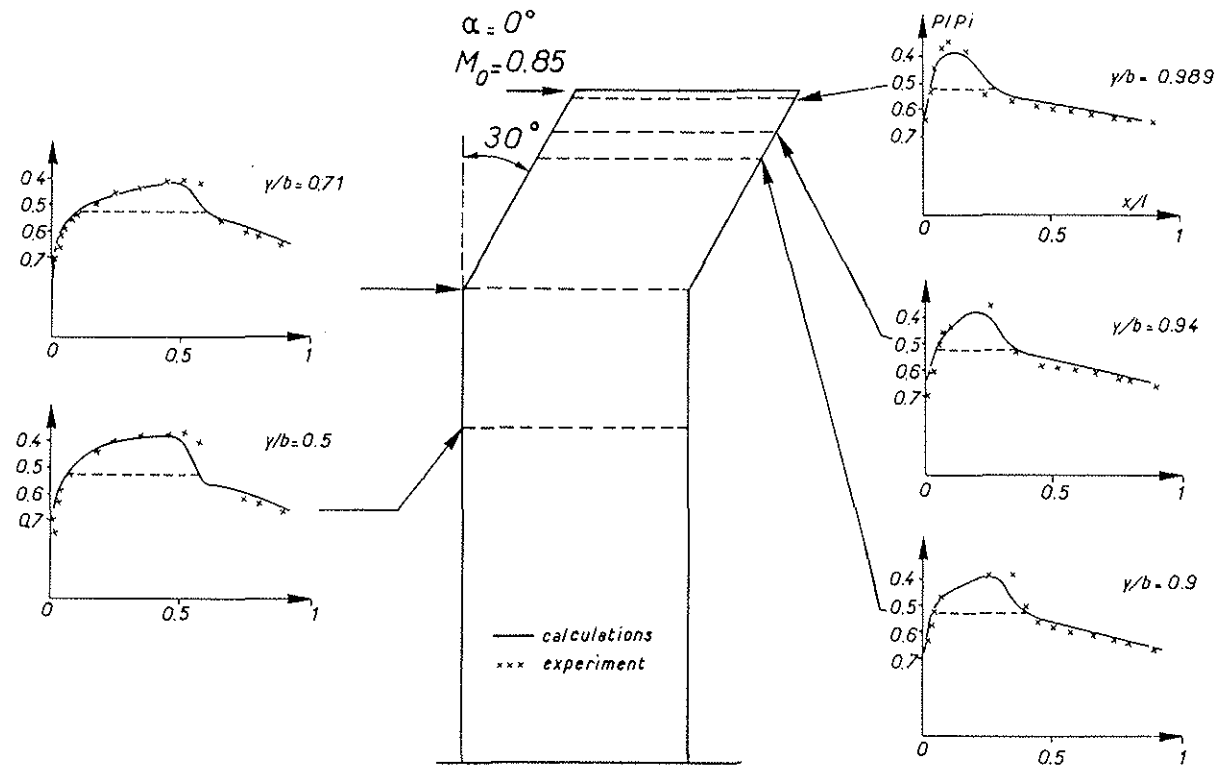


Fig.8 - 30° swept blade tip

As the calculations are done for an ideal fluid, we cannot expect a very precise positioning of the shock waves, but we remark that the main three-dimensional effects are well predicted.

4.2 - Vortex interaction

A wind tunnel simulation of the effect of the tip vortex effect emitted by a blade on the following one can be obtained by placing upstream of a half wing, called "receiving wing", another, so-called "emitting wing", as shown on figure 9.

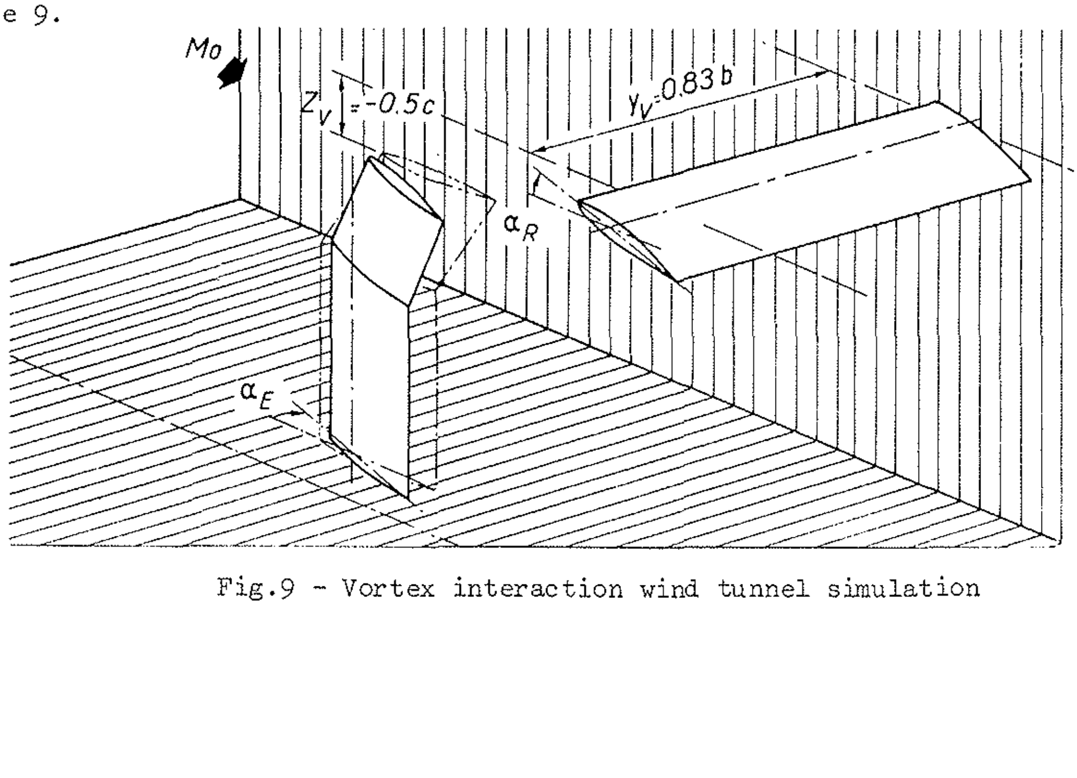
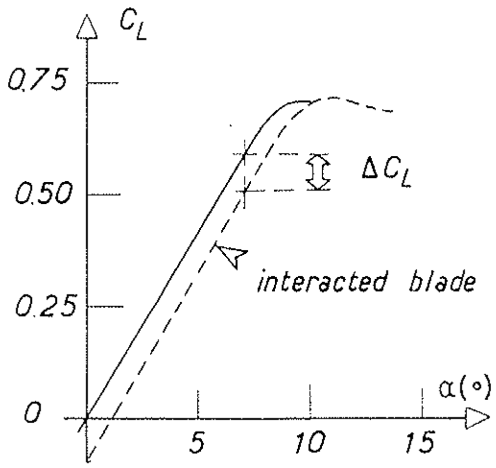


Fig.9 - Vortex interaction wind tunnel simulation



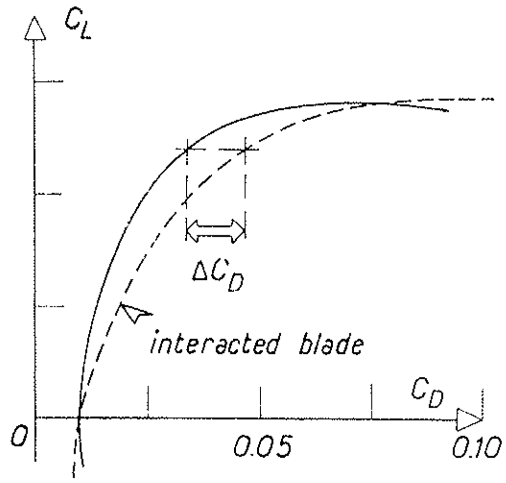


Fig.10 - C_L, C_D curves of a NACA0012 half wing with or without vortex interaction at M_o = 0.6

The presence of the perturbing vortex entails (fig.10):

- a lift decrease on the receiving wing, for a given angle of attack,
- a drag increase for a given level of lift.

Tests carried out by changing the angle of attack of the emitting wing and thus the intensity of the emitted vortex, at Mach number 0.6 characteristic of a hovering rotor, show that this drag increase is a nearly linear function of the emitting wing angle of attack (fig.11).

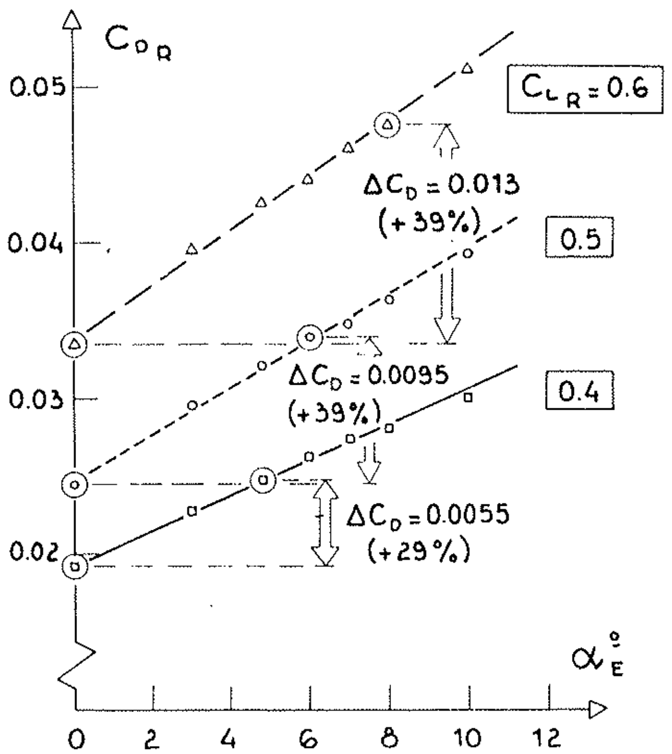


Fig.11 - Drag increase due to the vortex interaction at M_o = 0.6

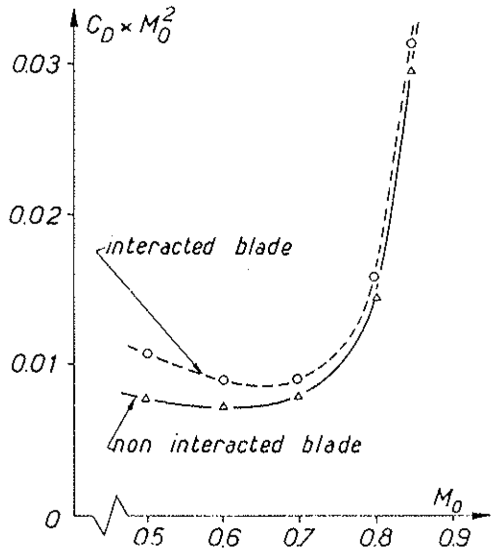


Fig.12 - Mach number effect on vortex interaction phenomena ($C_L \times M_o^2$ = 0.144 C_L = 0.4 M_o = 0.6)

For emitting wing lift levels equal to those of the receiving wing (circled points), there is a drag increase of nearly 40% due to the presence of a vortex centered at about 0.5 chord under the receiving wing.

These results have to be transposed in order to be utilized in real, wall interference free, conditions. However, one can see on figure 12 that the vortex interaction reduces always the L/D ratio of the blade, whatever the flow Mach number is. These studies complement the detailed ones that had been performed a few years ago in incompressible flow [10] .

So we can understand the difficulty of the problems raised by the optimization of a helicopter blade tip as, in hovering flight, it is always placed in the vortex shed by the preceding blade tip : the optimum blade tip should not be too sensitive to the presence of a perturbating vortex. It would be desirable to have a computing programme for the prediction of aerodynamic characteristics in these cases ; the US Army Ames Laboratory works on this project, and the above experiments will be useful to test this programme.

5 - STUDIES ON ROTOR MODELS

The ONERA Aerodynamics Department has equipped its S2 wind tunnel of Chalais-Meudon with a helicopter rotor test rig, and started the study of geometrically simple rotor models. As the laws of induced velocities through the disc of a lifting rotor are not yet well known and as, consequently, the local incidences cannot be precisely determined, it seemed interesting to study first the three-dimensional and unsteady effects in the case of straight and swept blade tips , on an untwisted rotor, tested at zero lift.

5.1 - The helicopter rotor test rig of the S2-Ch wind tunnel

Figure 13 shows a layout of this facility. The maximum flow velocity in the wind tunnel is 110 m/s (about 400 km/h). The rotors tested have a diameter of about 1.5 m, and can rotate at tip speeds exceeding 210 m/s. A six-component balance and a torquemeter provide the total forces acting on the rotor and the power necessary to drive it.

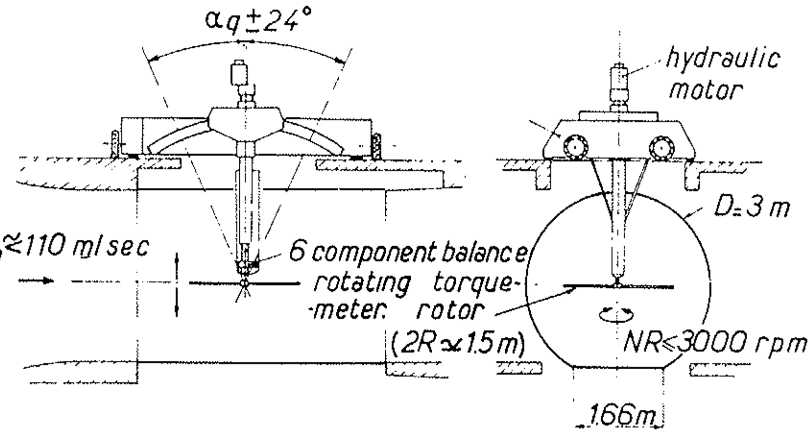


Fig.13 - ONERA S2 Chalais wind tunnel rotor test rig

The rotor chosen for these first blade tip studies is an untwisted two-blade rotor (fig.14), whose tip can be dismantled at 0.8 R. This tip is equipped with absolute pressure transducers Kulite LDQL, distributed along three sections at 0.85, 0.90 and 0.95 R (fig.15). Other details on the techniques used for the measurements data, acquisition and processing, are provided in ref. [1] and [11].

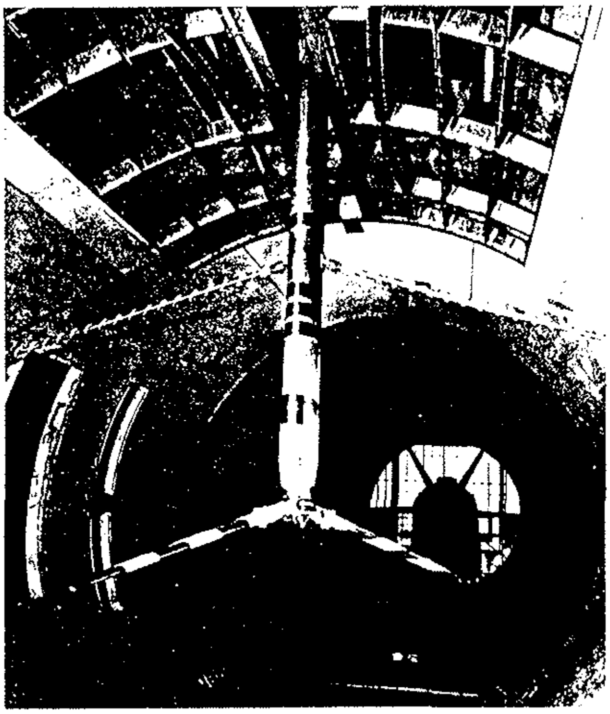


Fig.14 - The 1.5 m-dia.-research rotor in the S2Ch wind tunnel

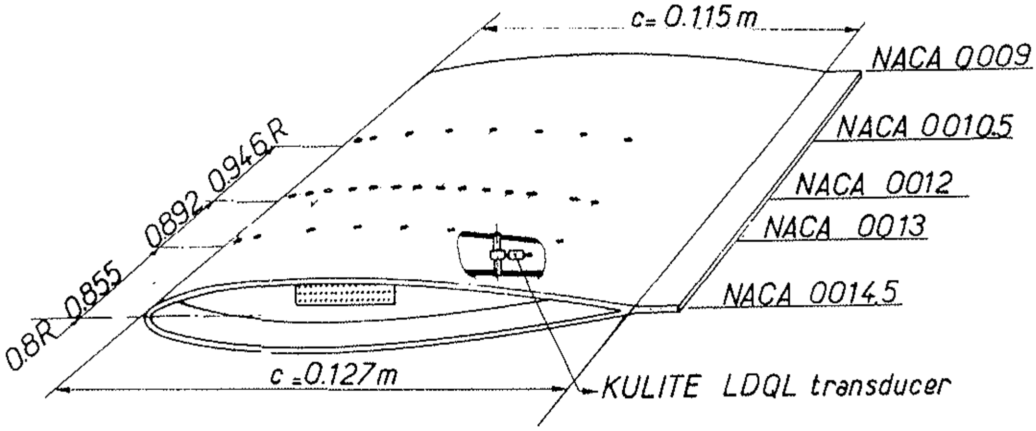


Fig.15 - Instrumented rotor blade tip layout

5.2 - Effects of the unsteady incident velocity

These effects have been brought to light during tests of rotors with straight tips. They appear as a dissymmetry in pressure distributions for two symmetrical azimuths ψ relative to azimuth 90°, for which the incident velocities are the same ; these pressure distributions would then be the same in the absence of unsteadiness and sweep effects. Figure 16 shows that, for all the sections near the blade tip, the supersonic zones are less extended and the shocks (when they exist) less strong at $\Psi = 60°$ than at $\Psi = 120°$, the pressure coefficient C_p being relative to the local instantaneous dynamic pressure :

$$C_p = (p - p_0) / \tfrac{1}{2} \gamma p_0 M^2$$

with : $$M = (\omega r + V_0 \sin \psi) / a_0$$

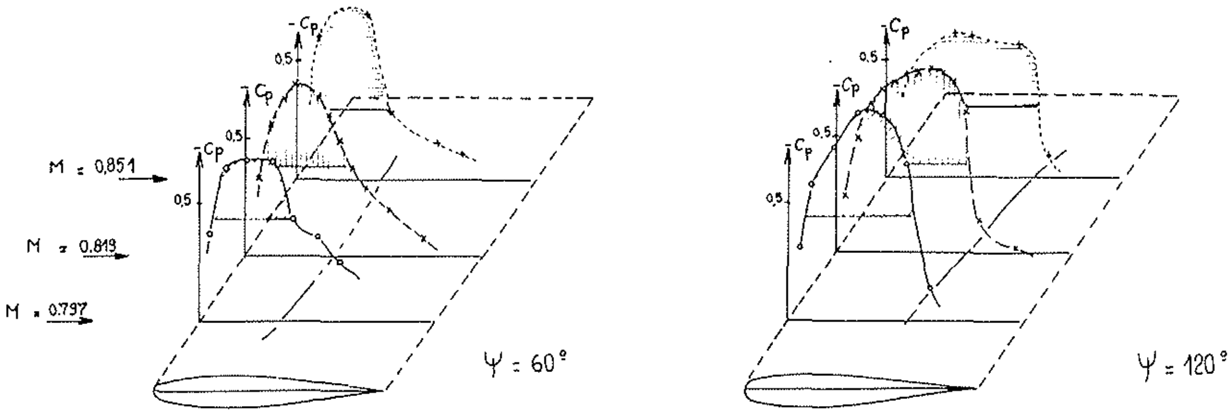


Fig.16 -Three-dimensional pressure distributions on straight blade tip. Non lifting case - Sections at 0.85, 0.9, 0.95 R. V_0 = 105 m/sec. ωR = 210 m/sec.

The supersonic zones are hachured and the shock lines are brought to light on the figure.

These phenomena are well predicted by calculations of two-dimensional, unsteady, transonic flows (see figures 17,18 and 19, (from ref. 1 and 11) for a section at 0.892 R), which shows that they are essentially related to unsteady effects.

Figure 17 shows comparisons between experimental and calculated pressure distributions, from a computer programme developed by F.X Caradonna of the US Army Ames Laboratory. The method consists in solving the small perturbation equation of the velocity potential for transonic flows, and is described in detail in ref. [11]. Calculation and experiment agree on the fact that, for azimuth 60° there is a recompression without shock, while there is a strong shock for azimuth 120°.

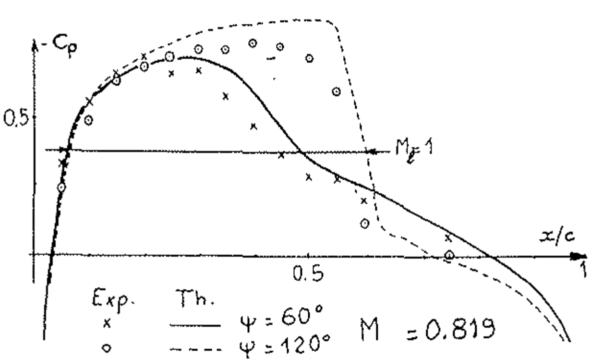


Fig.17 - Computed and measured pressure distributions, non lifting case, NACA 0012 r/R = 0.892 V_0 = 110 m/sec. ωR = 200 m/sec.

Figure 18 shows the experimental evolution of the pressures at mid-chord on section 0.892 R. It is compared with the evolution calculated :

. on the one hand by the small perturbation programme of F.X Caradonna

. on the other hand by a programme solving the complete Euler equation for two-dimensional transonic flows on a profile in arbitrary motion. (This work has been carried out by A. Lerat and J. Sidès [12] by imposing, in the present case, a sinusoïdal in-plane motion).

Both calculations and experiment show that the shock crosses over the mid-chord while travelling backwards, towards the trailing edge, when the incident Mach number increases, at an azimuth close to 70°, while it recrosses over the mid-chord, travelling upwards towards the leading edge, at an azimuth around 130°.

Figure 19 shows the importance of taking viscosity into account to obtain a precise definition of the phenomena : the pressure evolution with azimuth is correctly predicted for x/c = 0.5 only if we take into account the profile shape after it has been fattened by the boundary layer displacement thickness, calculated by solving the integral equations of unsteady boundary layers.

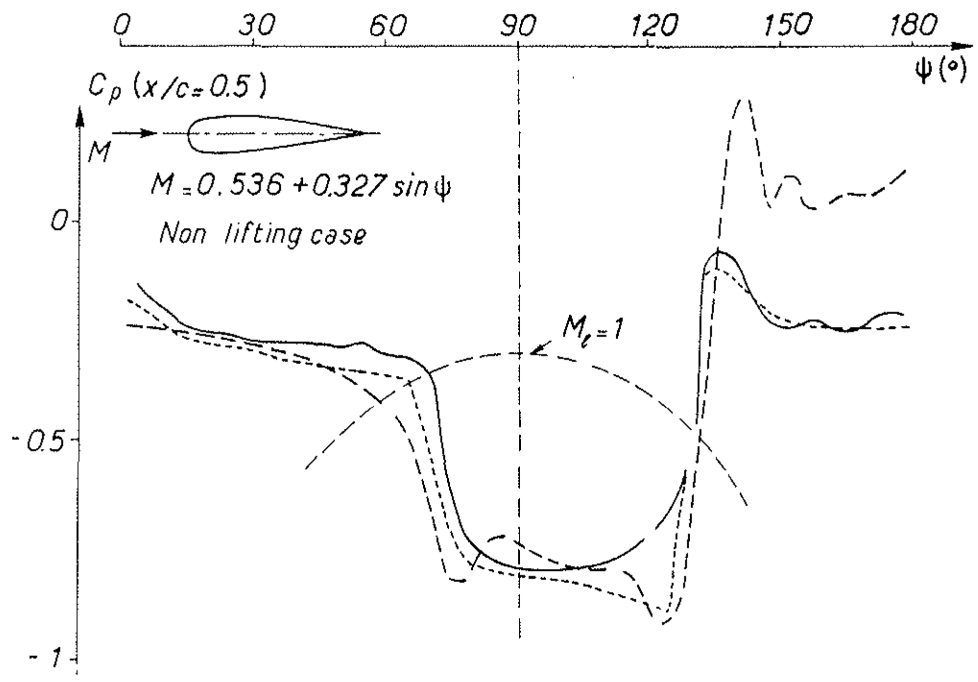


Fig.18 - Comparison between calculations and experiment

—— rotor experiment

— — Euler equations (In-plane equation)

---- small disturbance potential equation

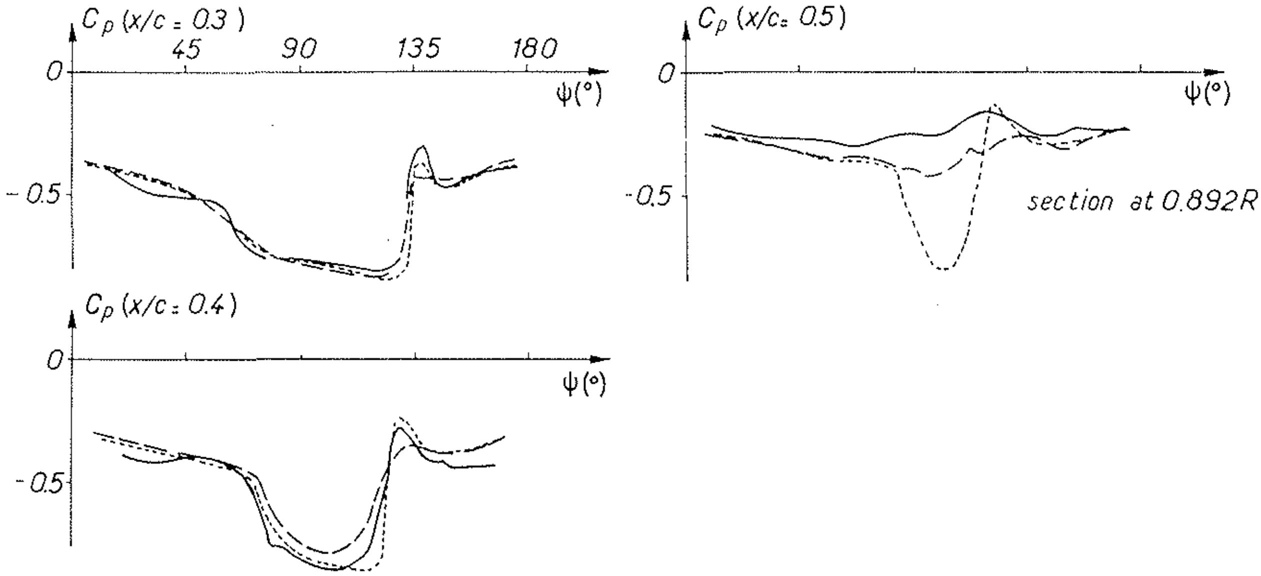


Fig.19 - Two-dimensional unsteady transonic calculations
— — with boundary layer coupling
- - - - or without boundary layer coupling
——— Rotor experiment in S2Ch wind tunnel

The calculation, that requires several iterations before convergence, has been carried out by J.J. Thibert of ONERA, and the example of figure 19 shows that, then, the maximum backward travel of the shock on the profile is much better predicted.

Two-dimensional flow calculations cannot however predict what happens in sections very close to the tip, where three-dimensional effects are no more negligible. Figure 20 shows the necessity of three-simensional, unsteady transonic flow calculations (on which the US Army Ames Laboratory is working at present), since two-dimensional calculations give a bad prediction for a section at 0.945 R.

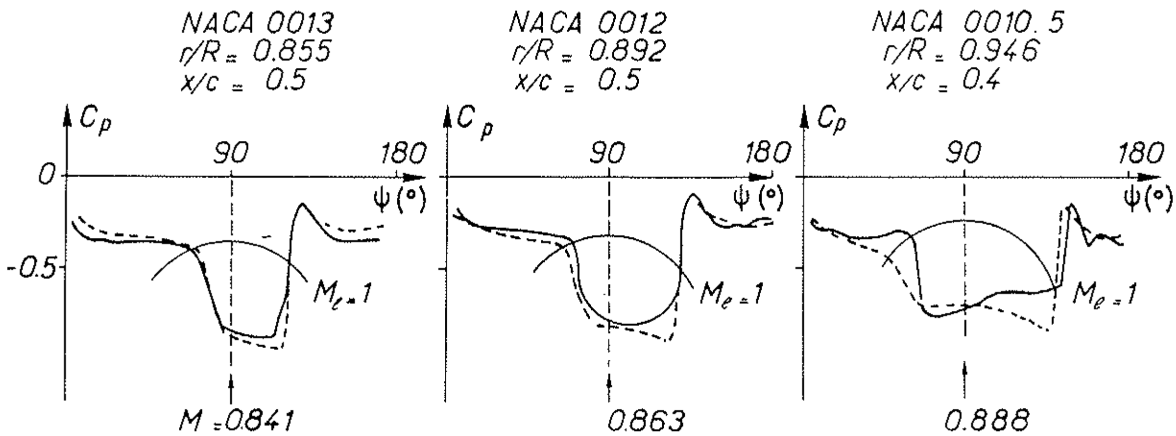


Fig.20 - Evolution of absolute pressures on a non lifting rotor blade tip at V_o = 110 m/sec. and ωR = 200 m/sec.
——— Tests at S2Ch - - - - Calculations

5.3 - Case of 30°-degree-sweep blade tip

As soon as 1972, W.F Ballhaus and F.X Caradonna had performed three-dimensional, steady, transonic flow calculations, showing the interest of swept blade tips [13] . This section will present the main results obtained on a rotor model whose straight tip, previously studied, has been replaced by a 30° swept tip over the last 15% of the blade span, while retaining the same laws of relative thickness ; we thus studied, the total and local effects of sweeping a blade tip.

5.3.1 - Compared total performance

As shown in section 2, the tip sweep postpones the appearance of important transonic troubles, and entails a lower aerodynamic drag (see fig.3). In as much as these qualities are maintained on a rotor, the swept tip rotor, at zero lift, should have a lesser drag and require less power than the same rotor with straight tips.

This is confirmed by experiments (fig.21), but we should remark that we must have a high advancing tip Mach number (Mach number at the tip for an azimuth ψ of 90°), above 0.87 for the gain to be appreciable. When a rotor is running in the transonic regime, the advancing tip Mach number is anyway the main parameter for the evolution of the rotor drag and driving power. This is the reason why figure 22 provides the same curve for a large number of test configurations where the tip speed varies from 190 to 220 m/sec and the wind tunnel velocity from 75 to 110 m/sec.

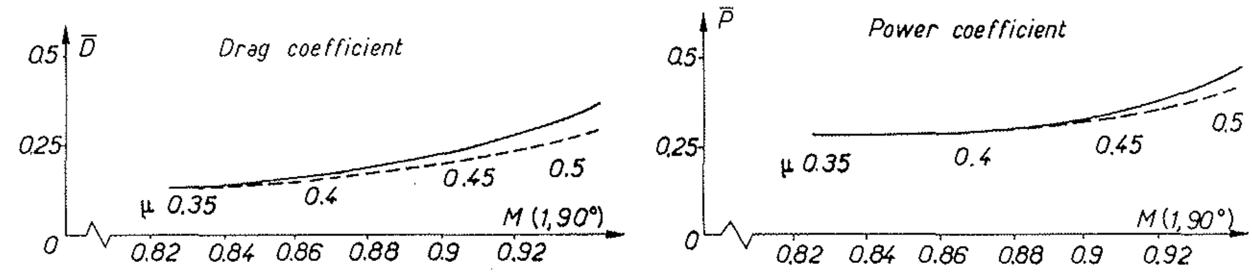
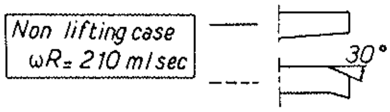
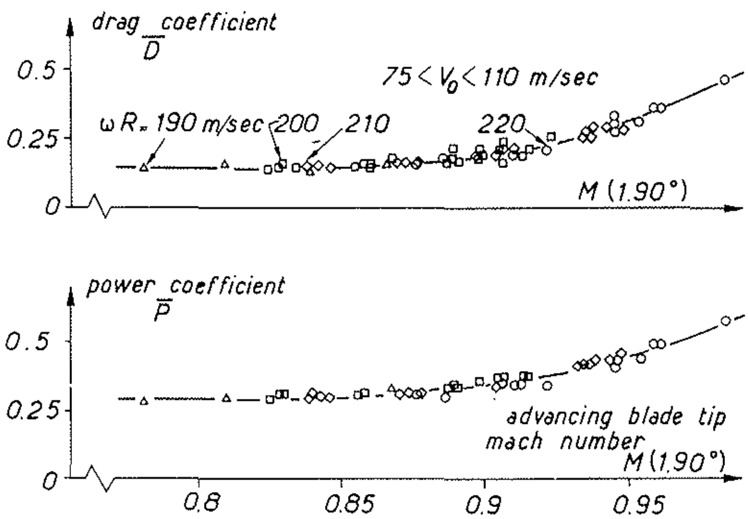


Fig.21 - Influence of blade tip sweep on rotor drag and required power





$$\bar{D} = 100\,D / \left(\tfrac{1}{2}\rho(\omega R)^2 S\sigma\right)$$

$$\bar{P} = 100\,P / \left(\tfrac{1}{2}\rho(\omega R)^3 S\sigma\right)$$

Fig.22 - Drag and power required for the 30° swept blade tip. Non lifting case

5.3.2 - Local flows on swept blade tips

Pressure distributions measured in three sections of the tip allow a better understanding of the advantages of the swept over the straight tip.

Figure 23 presents one of the characteristic results obtained. For test conditions very close to those of a straight tip rotor (see fig.16), we compare the pressure distributions measured at ψ = 60° and 120°. At 60°, the supersonic zones are very little extended and without shock, and the local Mach numbers are much smaller than with straight tips. At 120° we have very important shocks, and the pressure distributions have a much more marked three-dimensional character than with the straight tip. Apart from the 0.85 R section, the shocks at that azimuth are much stronger on the swept tip than on the straight tip.

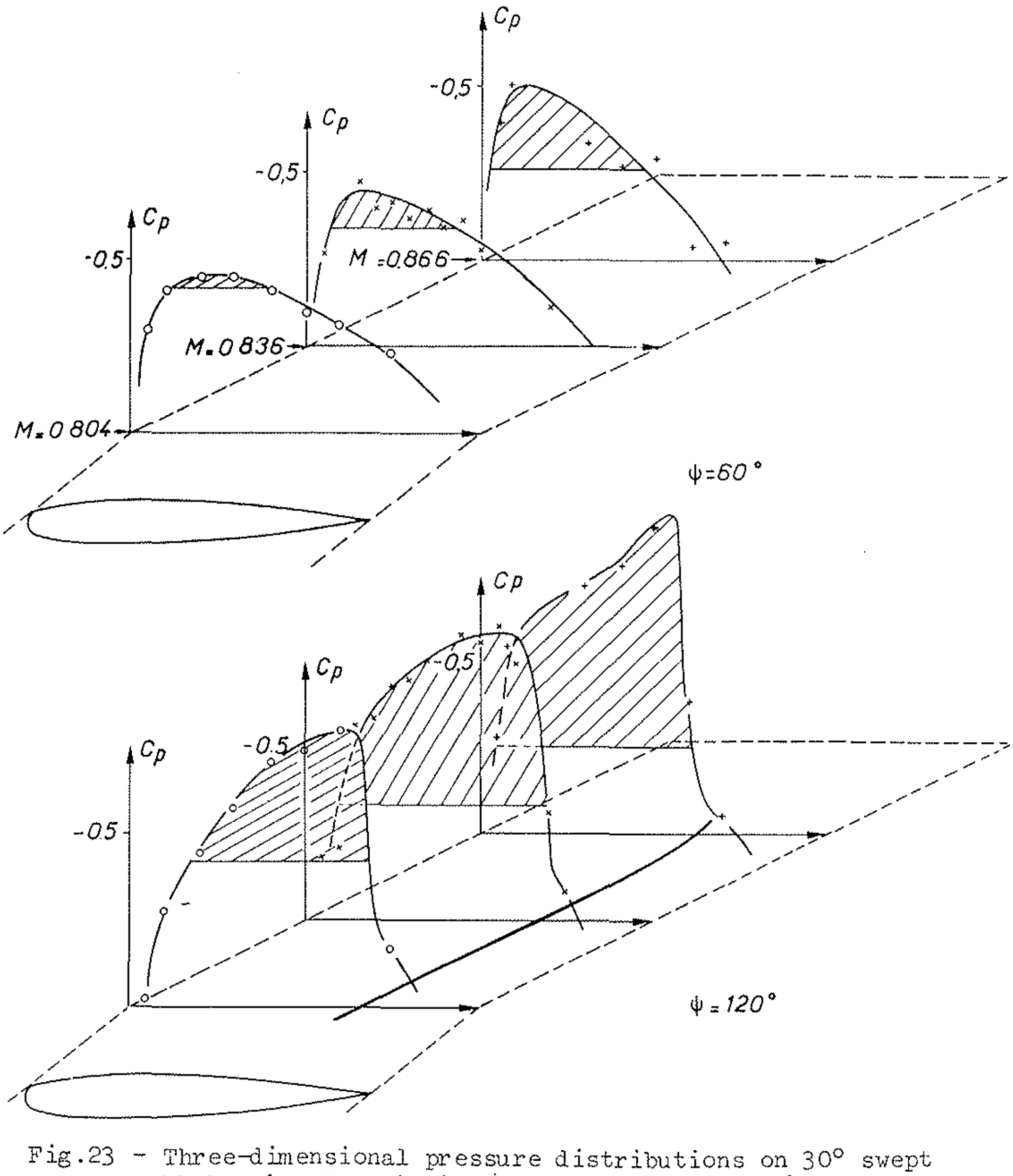


Fig.23 - Three-dimensional pressure distributions on 30° swept blade-tip. Non lifting case. Sections at 0.85, 0.9, 0.95 R. V_o = 105 m/sec. ωR = 210 m/sec.

Figure 24 shows that, at 0.9 R, the swept tip presents local Mach numbers lower than the straight tip for the advancing blade from azimuths 0 to about 115°, while later it is the contrary : the maximum local Mach numbers are then higher and the shocks stronger for the swept blade.

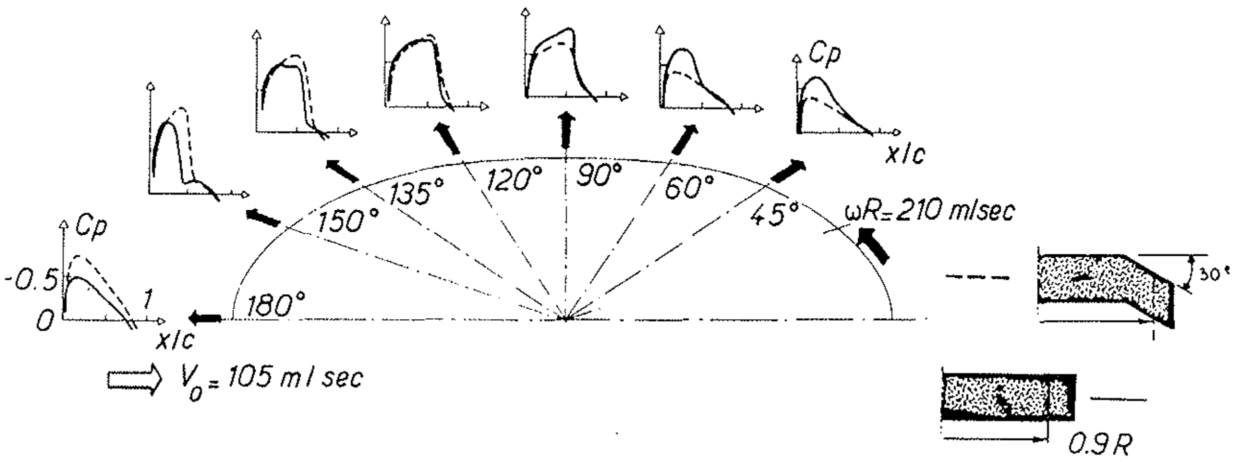


Fig.24 - Evolution of pressure distributions on rotor blade tips
Non lifting case - 0.9 R NACA 0012 section

The azimuth at which this phenomenon occurs depends on the section considered, and figure 25 shows that the swept effect is less favorable for sections nearer the tip.

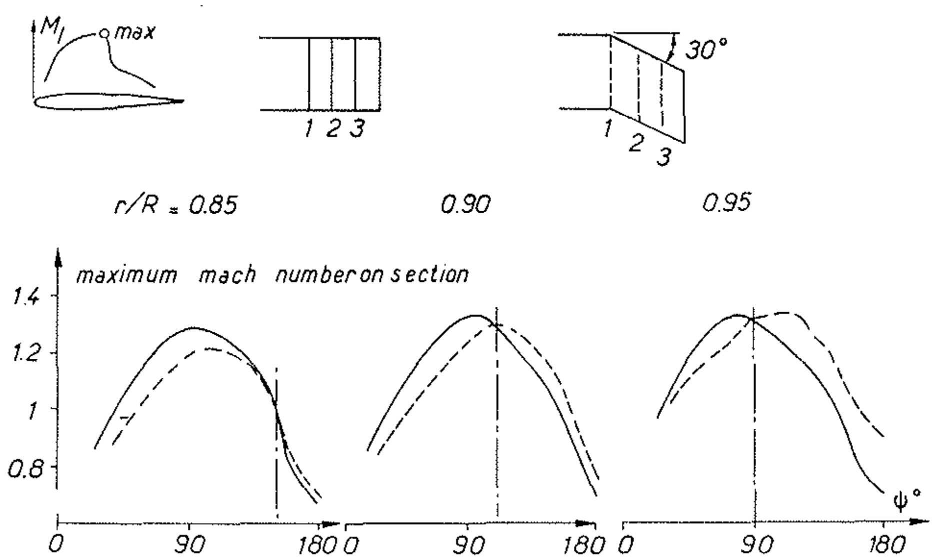


Fig.25 - Maximum Mach number on various rotor blade tip sections
Non lifting case V_o = 105 m/sec. ωR = 210 m/sec.

Figure 26 also shows that if the appearance of shock waves is postponed by the tip sweep, the shock intensity, when they appear, is higher than with the straight tip. In fact, the blade tip sections are, in the second quadrant, placed in less favourable attack conditions than those of the straight tip, as the actual sweep angle (algebraic sum of geometric and aerodynamic sweep angles) may become, in absolute value, smaller with the swept tip than with the straight tip (the aerodynamic sweep angle is then negative).

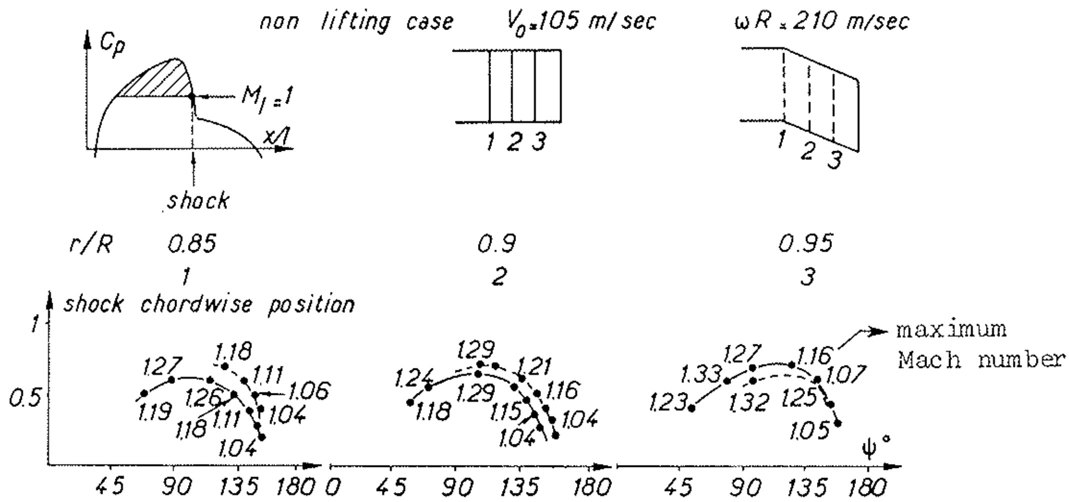


Fig.26 - Azimuthal evolution of shock position on rotor blade tip sections

For a 30°-degree-geometric sweep, the azimuthal limit for which the swept tip is working with a sweep angle smaller, in absolute value, than the straight tip is the azimuth at which the aerodynamic sweep is equal to -15°. Figure 27 obviously shows that this explanation is not sufficient, though it determines an order of magnitude of the azimuthal limit up to which the swept tip presents local Mach numbers lower than those of the straight tip.

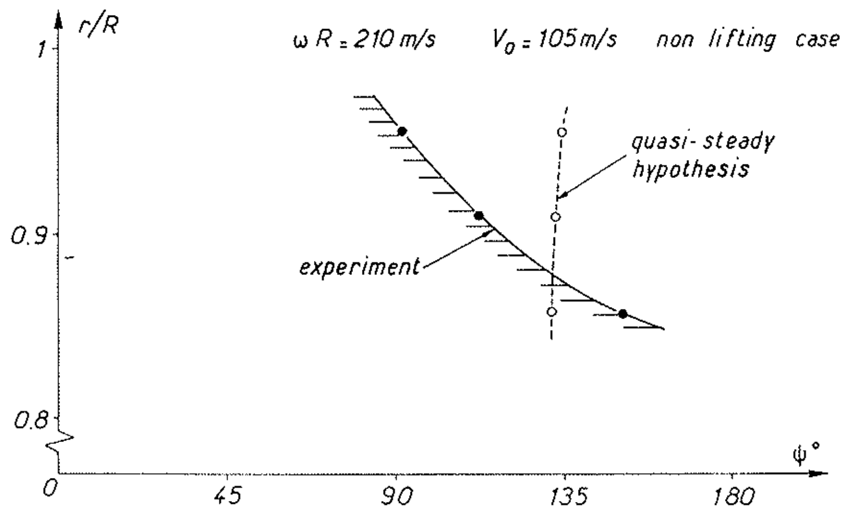


Fig.27 - Advancing blade azimuthal limit advantageous to the 30° swept tip

In spite of the existence of a zone of the rotor disc where the swept tip has a lower performance than the straight tip, the overall balance remains favourable to the swept tip, as the latter is working in better conditions over a larger part of the disc, and especially around azimuth 90°, where the dynamic pressures are the highest.

The results presented here give a good idea of the complexity of the phenomena occurring on a blade tip with relatively simple geometric shape. Steady flow calculations are a mean of optimization as a first approximation, as we find a good similarity in the aerodynamic behaviours of a half wing at the wall and a helicopter blade tip . However, unsteady flow calculations will become essential to make sure that the unsteadiness of the velocity and sweep attack conditions do not withdraw any interest to the new blade tip planform, especially in transonic flow.

ONERA intends to pursue its effort for understanding the flows over helicopter blade tip in the lifting case by testing in its S2-Ch wind tunnel a twisted, three-blade rotor equipped with absolute pressure transducers.

6 - CONCLUSIONS

Only the aerodynamics problems related to the helicopter blades tips have been approached in this paper. Even in the case of rigid blades, the knowledge of three-dimensional, unsteady flows is still imperfect. The work carried out at ONERA on simple blade tip forms, straight or swept at 30°, contributes however to the understanding of the difficult problems to be solved in order to optimize blade tips. Measurements performed on half wings at the wall and on rotor models in simple configurations will help to validate the more and more complex computer programmes which it will be mandatory to develop to predict with reasonable accuracy the aerodynamic behaviour of helicopter blade tips.

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