Full Day Discussion

DESIGN AND CONSTRUCTION OF ROTOR BLADES

Chairman A McCLEMENTS, A R T C , M I MECH E

An all-day meeting of The Helicopter Association of Great Britain was held on Saturday, October 6th, 1951, in the Library of the Royal Aeronautical Society, 4 Hamilton Place, London, W 1

The CHAIRMAN, opening the Proceedings, said To-day we are going to discuss a subject of very great importance Our rotating wings, as well as providing sustentation, must often provide propulsive thrust and control, and always operate with a reasonable degree of satisfaction under aerodynamic conditions which are far removed from the ideal Further, our wings are maintained in equilibrium by a series of couples quite foreign to the fixed wing , also, they generate disturbances which must be kept under control, otherwise the overall level of vibration can become wholly unacceptable Such considerations lead on quickly to the conclusion that we are concerned with a subject more complex in many respects than its fixed wing counterpart, and quite fundamental in the science of rotating wing aircraft With thoughts such as these in mind your Council decided that to-day would be devoted to a study of the design and construction of rotor blades We have accordingly arranged a programme which we believe will be of great interest

There will be three sessions The **morning session** will be devoted to theoretical considerations of various aspects of the problem We shall hear two papers read by well known authorities These papers will be followed by a short discussion The **afternoon session** will commence at 2 p m It will be concerned with the more practical aspects of blade problems and it will include five papers read by speakers well known in their own fields The **evening session** will extend from 4 30 to 5 45 p m , and it will be devoted to discussion only

Morning Session

(FIRST PAPER)

Calling on Mr H B SQUIRE to present his paper on " Some Aerodynamic Aspects of Rotor Blade Design," the Chairman said that Mr Squire was so well known that there was little need for him to be introduced We were aware of his contributions in the fields of performance evaluation methods, stability, and general theoretical work during his tenure of office at the Royal Aircraft Establishment , also of his work on the Helicopter Committee of the Aeronautical Research Council (of which he was Chairman from 1946-1951) The Chairman mentioned that Mr Squire was a member of the Staff of the National Physical Laboratory and it was gratifying to know that he still found time to devote to our problems

Some Aerodynamic Aspects of Rotor Blade Design*

By H B SQUIRE, M A

(National Physical Laboratory, Teddington, Middlesex)

A survey is made of the aerodynamic problems of rotor blade design, including the effects of taper and twist, stalling, compressibility, special aerofoil sections and surface condition

(1) Introduction

In studying the aerodynamics of rotor blades we are often uncertain how far we may apply quantitatively the data obtained from wind tunnel tests of aerofoils Of course, the rotor blade behaves qualitatively like an aerofoil , but I sometimes feel doubtful about calculations of the effect of detailed changes in blade design, say for example, the effect of taper on performance, and wish to stress the desirability of wind tunnel tests by which some at least of these questions could be settled more easily than by flight tests

I have found in preparing this lecture that covering the whole field would be impracticable I intend to concentrate mainly on the compressibility problem which, closely associated with stalling, seems to me to be the principal aerodynamic problem facing the blade designer today

(2) Blade Stalling

Consider what happens when the horizontal speed of a helicopter is increased The advancing blade now has a greater speed and the retreating blade has a lower speed relative to the air (Fig 1) The incidence of the advancing blade is reduced by flapping or cyclic pitch control and the incidence of the retreating blade is increased in the same way Thus, as the forward speed increases the incidence of the retreating blade increases and eventually stalling with loss of lift and increase of drag occurs It has been found by Gustafson and Myers[1] that stalling begins when the blade incidence reaches 12° and becomes prohibitively severe when the blade incidence reaches 16°

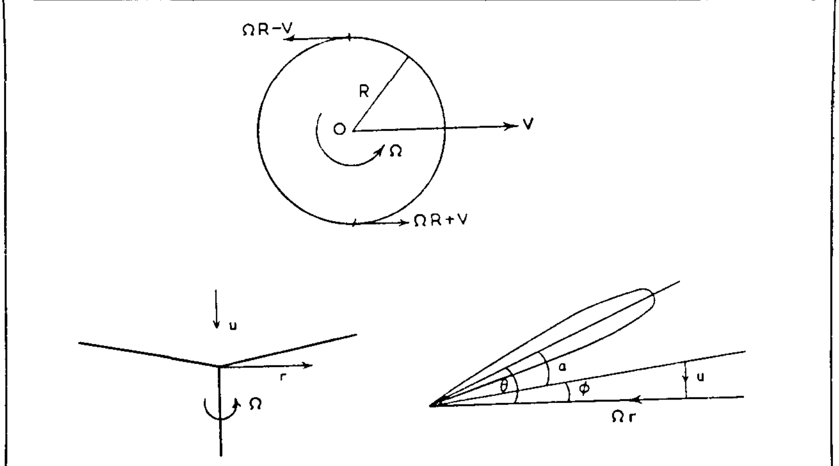

FIG 1

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(3) Taper and Twist

Before considering the effects of twist and taper on rotor performance I will indicate briefly why taper and twist may be desirable Consider a rotor with an axial velocity u which may be partly due to the rate of advance and partly due to the induced velocity Thus the blade element operates under the conditions shown in Fig 1 We have

$$\text{Thrust of blade element per unit span} = \tfrac{1}{2}\rho(\Omega r)^2 C_L\, c,$$

$$\text{and} \quad C_L = a\,\alpha = a(\theta - \varphi) = a(\theta - \frac{u}{\Omega r})$$

In these equations the following notation is used

ρ	=	air density
Ω	=	angular velocity of blade
c	=	blade chord at radius r
θ	=	blade angle at radius r
C_L	=	lift coefficient of blade element
a	=	lift curve slope of blade section
α	=	incidence of blade element
u	=	axial velocity of air relative to blade element

Now in order to obtain a uniform loading over the disc, which is the ideal condition, we should have the thrust increasing linearly along the blade Thus C_L cr is constant along the blade for ideal conditions If further we require the lift coefficient to be constant along the blade to keep the profile drag losses down to a minimum then we shall need to have α and cr both constant along the blade so that $c \propto \frac{1}{r}$ In addition $\theta = \alpha + \frac{u}{\Omega r}$ so that the blade angle consists of a constant term and a term which varies inversely with the radius If the term $\frac{u}{\Omega r}$ which depends on flow through the disc is appreciable compared with α then twist of the blade is necessary As a rough indication it may be said that twist is desirable if the blade angle at 0 75R is more than 10° but is hardly worth while if the angle is much less than 10°

Gessow[2] has made calculations of the effect of taper and twist on performance in hovering and some of his results are given in the following tables

EFFECT OF TWIST UNTAPERED BLADES

$C_T = 0\ 005, \quad C_Q = 0\ 00035, \quad \sigma = 0\ 06$

Blade twist (deg)	*Increase of thrust*
0	—
— 8	2½%
—12	3½%

EFFECT OF TAPER UNTWISTED BLADES

$C_T = 0\ 005, \quad C_Q = 0\ 00035, \quad \sigma = 0\ 06$

Blade taper (ratio of tip chord to root chord)	*Increase in thrust*
1	—
3	2½%

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These tables are calculated for the following representative values

$$\text{Thrust coefficient } C_T = \frac{T}{\rho\pi R^2(\Omega R)^2} = 0\ 005$$

$$\text{Torque coefficient } C_Q = \frac{Q}{\rho\pi R^3(\Omega R)^2} = 0\ 00035$$

$$\text{Solidity } \sigma = \frac{Nc}{\pi R} = 0\ 06$$

where T = thrust
Q = torque
R = radius of disc
N = number of blades

Calculation of the effect of combined twist and taper showed that the effects are nearly but not quite additive

These increases in thrust during hovering are significant but the actual gains may not be quite so large because the theory used may exaggerate the effects somewhat However, it has been shown in another report by Gessow[3] that twist is advantageous in postponing the onset of the stall at the tip of the retreating blade as the forward speed increases Tests were made on a Sikorsky Hoverfly helicopter with untwisted blades and with blades having a twist of —8° Fig 2 shows the rotor profile-drag power plotted against forward speed and it will be seen that the blades with 8° twist suffered from the onset of blade stalling, as indicated by an increase in the profile-drag power, at a speed of 7 m p h higher than for the untwisted blades

(4) Effect of Aerofoil Thickness and Shape on Drag at High Speeds

Wind tunnel tests

Up to the present it has been usual in dealing with the effects of compressibility on rotors to propose a limiting Mach number for the tip of the advancing blade This is, of course, too simple as the Mach number for the onset of serious compressibility effects depends on the aerofoil thickness, shape and incidence

Fig 3 shows the results of German tests[4] on symmetrical aerofoils of different thickness at zero incidence The line marked M_D represents the drag divergence Mach number, which corresponds to the Mach number at which the drag coefficient has risen by 0 005 above its low speed value We may deduce from this diagram that it is necessary to keep the thickness down to 12% if the tip of the advancing blade is to operate at speeds approaching M = 0 8

The effect of the position of maximum thickness on this drag divergence Mach number is shown in Fig 4, which gives the results of R A E tests[5] on three aerofoils of thickness 10% It will be seen that a rearward position of maximum thickness is better at moderate lift coefficients, but gives little advantage at low lift coefficients The low speed maximum lift coefficient may be higher with forward positions of the maximum thickness, owing to the associated blunter noses, so there does not seem to be a good case for recommending the use of sections with maximum thickness aft of the 30% chord position, at any rate for the present

Full scale rotor tests

Full scale rotor tests have been made on the N A C A helicopter test tower[6] to study the effects of compressibility on the hovering performance of two rotors with blade section N A C A 23015, one with untwisted blades and one with blades having 8° of twist (wash-out) Increasing the tip speed produced a falling off in performance before any limitation due to vibration or loss of control occurred The results are presented in the form of curves of Measured Torque/Theoretical Torque plotted against calculated blade angle at the tip The departure of this ratio from unity is an indication of stalling or compressibility losses The tip speed at which this occurs

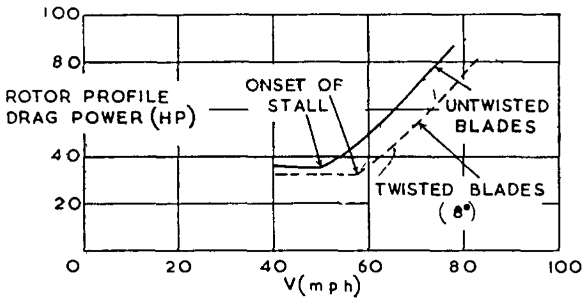

FIG 2 EFFECT OF BLADE TWIST ON TIP STALL

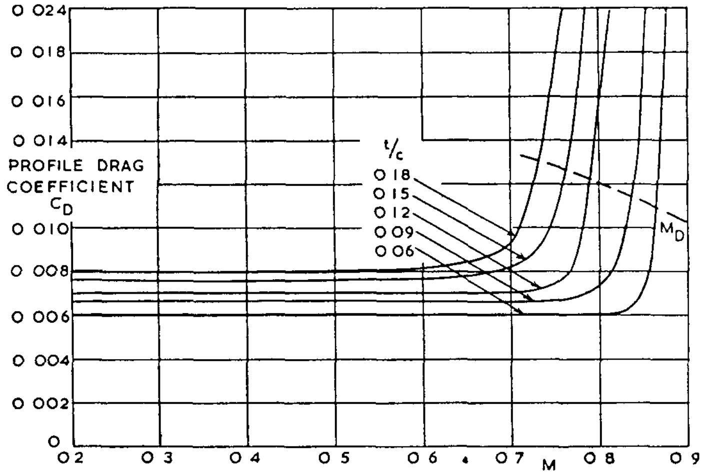

FIG 3 VARIATION OF DRAG COEFFICIENT WITH MACH Nº FOR SYMMETRIC AEROFOILS OF DIFFERENT THICKNESS RATIO AT $\alpha = 0$ MAXIMUM THICKNESS AT 30 % CHORD POSITION REYNOLDS NUMBER $4\ 5 \times 10^6$ AT M=0 8

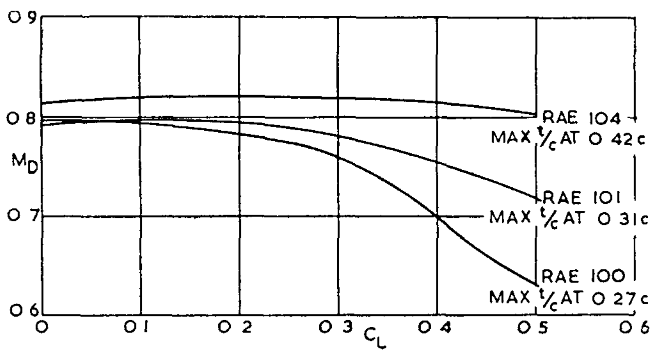

FIG 4 DRAG DIVERGENCE MACH NUMBER FOR 10 % THICK AEROFOILS

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may be regarded as the Mach number for drag divergence This is shown plotted against the angle of incidence of the section at the tip in Fig 5*

The rotor tower results are compared with wind tunnel data[7] on the blade of N A C A 23015 section and the agreement is quite close This method of plotting is very useful in that it shows up clearly that we are really concerned with compressibility effects over the whole range of incidence (There is often a gap on the high speed tunnel tests which generally stop at 6° or 8° incidence) The diagram also shows a difficulty in the analysis of rotor tests because the angle of incidence is calculated by a process which is necessarily rather arbitrary If possible this method should be replaced by a better one, though the difficulty of predicting or measuring the lift or incidence of the blade tip is formidable

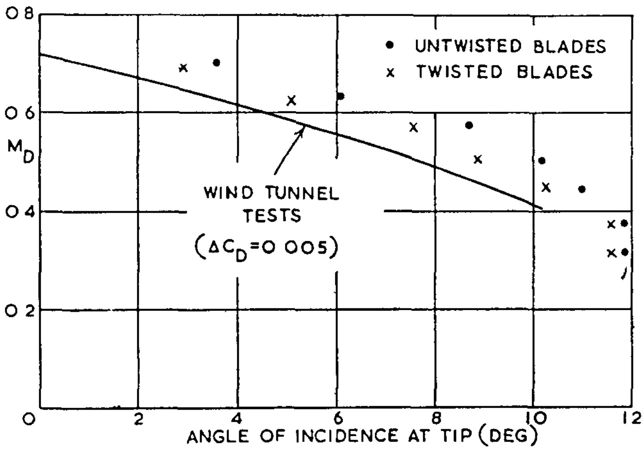

FIG 5 DRAG DIVERGENCE MACH NUMBER FROM ROTOR TESTS WITH BLADES OF NACA 23015 SECTION

(5) Effect of Compressibility on Lift and Pitching Moment

I do not propose to discuss the effect of compressibility on lift and pitching moment because the important changes generally occur later than the effects on drag, but it may be stated that, if the tip of the advancing blade is to operate at Mach numbers up to 0 8, then the blade section should be not more than 12% thick and should be symmetrical, if serious effects on lift and pitching moment are to be avoided

(6) Variation of $C_{L\max}$ with Mach Numbfr

The stall at the tip of the retreating blade mav be affected by the influence of compressibility on the maximum lift coefficient This is complicated by Reynolds number effects on $C_{L\max}$ which are also important The results of tests[8, 9] on the aerofoil N A C A 0012 are shown in Fig 6 At low values of M ($<$0 25) we get a considerable increase in $C_{L\max}$ with increase of Reynolds number from 0 5 $\times$ 10^6 to 5 $\times$ 10^6 but this gain has disappeared for values of M $>$0 4 The tests of the tapered N A C A 230 series wing[10] which are also shown in Fig 6 indicate that camber is of considerable advantage in increasing the maximum lift coefficient in the range 0 25$<$M$<$0 5 Since the tip of the retreating blade of a rotor in the top speed condition will be operating in the range M = 0 25 — 0 4 it is clear that the effect of compressibility on the stall of the retreating blade is important

(7) Special Aerofoils Sections

The development of special wing sections with the object of maintaining extensive laminar flow on aeroplanes naturally suggests that the same idea can be applied to

* It might seem from this diagram that the untwisted blades are slightly superior to the twisted blades, this is not the case because at a given thrust and tip speed the angle of incidence at the tip is less for the twisted than for the untwisted blades

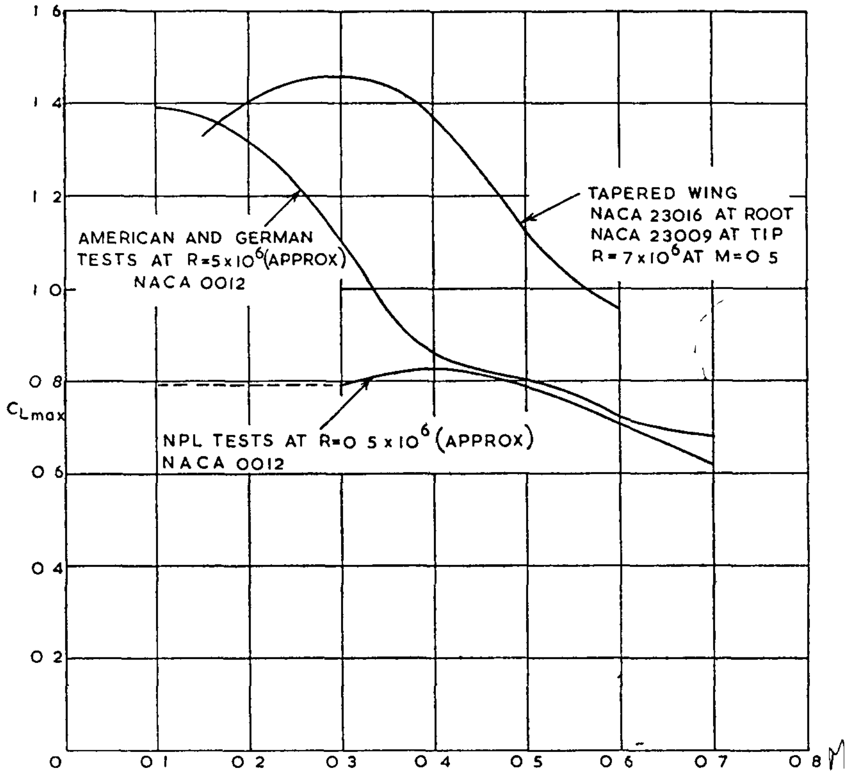

FIG 6 EFFECT OF COMPRESSIBILITY ON MAXIMUM LIFT COEFFICIENT

rotor blades The latter problem is more difficult because of the need to keep the movement of the centre of pressure small (small value of C_{M_0}) In addition operation over a large range of lift coefficient in forward flight is required The results of tests[11] of some sections for rotors designed by the N A C A and tested at Langley Field are shown in Fig 7 As far as I know they have not yet been tested on any helicopter The reflex at the trailing edge is a disadvantage as it would have to be maintained undistorted in flight Also there are indications that compressibility effects would give rise to appreciable nose-down pitching moments on reflex sections of this type

(8) Effect of Trailing Edge Distortions on Aerofoil Section Characteristics

If the trailing edge of the aerofoil is bent up or down it will introduce an extra pitching moment about the quarter chord point This effect may be used to adjust the blade characteristics by bending part of the trailing edge up or down to correct for errors in the blade shape Alternatively any small errors in blade shape near the trailing edge are liable to have important effects The theoretical and mean experimental results[12] for the effect of deflection of a plain flap near the trailing edge are shown in Fig 8

For example, suppose that the rear 10% of the blade chord is deflected up by 1° Then we find

$$\delta c_m = 0\ 007$$

and for a lift coefficient of 0 2 this corresponds to a centre of pressure at 0 215 of the chord from the leading edge

It is interesting to speculate whether it would be possible to use this variation of pitching moment with trailing edge deflection to design a rotor which did not flap

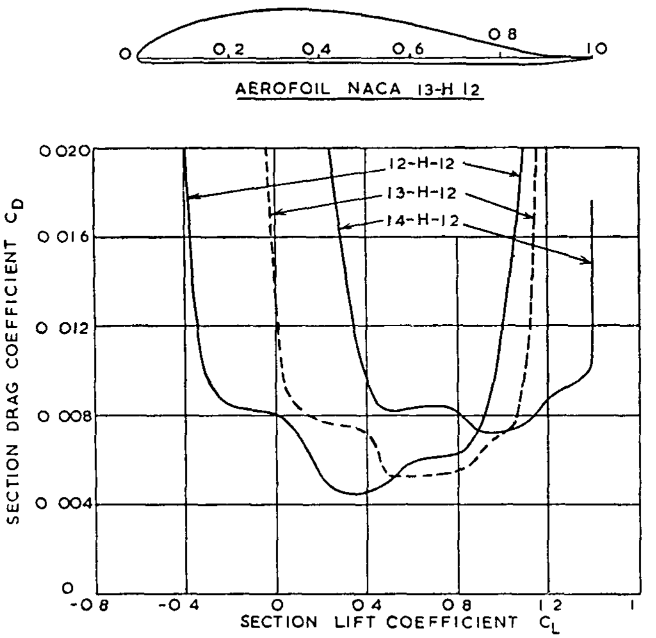

FIG 7 AEROFOIL SECTIONS DESIGNED FOR ROTOR BLADE
$R = 2\ 6 \times 10^6$

The flapping hinge would still be required for coning but it would be one step in the direction of the rotor without hinges at the root

(9) Effect of Shape of Aerofoil near Trailing Edge

The variation in position of the aerodynamic centre of an aerofoil with thickness[12] is shown in Fig 9 for the N A C A 63 series wings which have cusped trailing edges and the N A C A 230 series which have nearly wedge shaped trailing edges of angle about 15° for t/c = 0 12 It will be seen that the aerodynamic centre is further back for the wings with cusped trailing edges It seems to me that the best trailing edge shape is wedge-shaped of small angle , this avoids the difficulty of constructing accurately very thin trailing edges and also largely avoids the slight disadvantages of the blunter trailing edges Certainly no blunter or more convex shapes at the trailing edge than the N A C A 230 series should be used

(10) Boundary Layer Control and High Lift Devices

Boundary layer control is more difficult to apply to rotor blades than to wings because of the variation in conditions round the disc and I am not hopeful of successful application of this technique, at any rate without a lengthy programme of research The use of high lift devices such as flaps and slots seems more promising but I do not know of any specific proposals for their application

(11) Surface Conditions

Effect of isolated excrescences on transition

Tests to study the effect of isolated excrescences on transition have been made by the N A C A and by the N P L The results of tests at the N P L [13] on conical pimples at a Reynolds number of $2\ 9 \times 10^6$ give the permissible pimple height on a blade section of chord 1 ft , to be between 0 002 and 0 004 inches for pimple locations near the leading edge

Fffect of roughness on drag with turbulent boundary layer

Over the rear part of the blade there will be a turbulent boundary layer and there will be increases in surface drag if the roughness is too great From the work of Prandtl and Schlichting as given in the Royal Aeronautical Society's data sheets we can derive for the limiting permissible roughness height

$$\frac{Uk}{\nu} = 100 \quad \text{for} \quad R < 10^7$$

where k is the mean height of excrescences For a blade moving at 500 ft /sec this gives 0 4 $\times$ 10^{-3} inches as the permissible roughness The effect of exceeding this limit somewhat is not very large, but there should be no difficulty in attaining a surface finish which is within the above limit

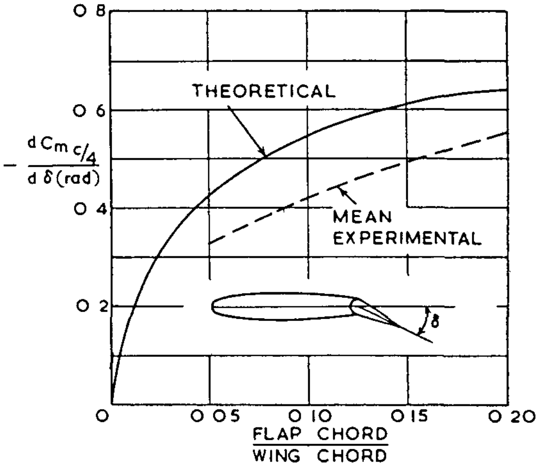

FIG 8 EFFECT OF FLAP DEFLECTION ON PITCHING MOMENT

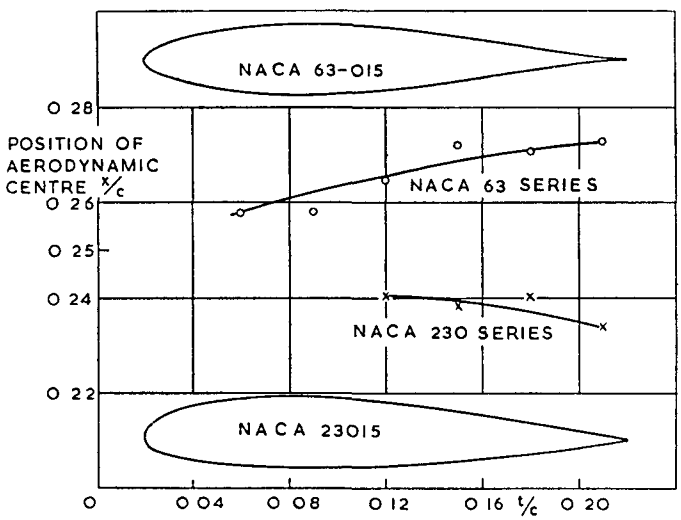

FIG 9 EFFECT OF AEROFOIL SHAPE ON AERODYNAMIC CENTRE

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Other surface imperfections

It is difficult to give any useful data on the effects of surface waviness, poor joints, fabric sag, and deformation in flight on drag. An attempt should be made to keep the surface within 0.01 inches of a smooth curve. Tests of several rotors made some years ago by the N.C.A.A.[14] showed that bad surface caused a considerable increase in the power required for hovering.

(12) Concluding Remarks

The above survey of some of the problems of blade design has impressed upon me the importance of the development of the technique of measurement so that the actual conditions at a blade section can be determined. This could best be done by pressure plotting the blade and I hope that this will be attempted sometime in the future.

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