SEVENTH EUROPEAN ROTORCRAFT AND POWERED LIFT AIRCRAFT FORUM

Paper No. 1

DEVELOPMENTS IN ROTARY WING AIRCRAFT AERODYNAMICS

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ABSTRACT

Helicopter aerodynamics have improved significantly in the past decade with the development of new blade sections and planforms coupled with improved understanding through computer modelling of the flow. In general mean performance is now understood but the prediction of oscillating loadings needs further work.

Aerodynamic developments of other parts of the helicopter have also been made. The need to reduce drag in forward flight is receiving greater attention as extended range operation is required and the need to reduce energy costs is tackled.

The paper will review developments in these areas and then briefly consider aerodynamic advances that may be expected in the next ten years.

1. INTRODUCTION

The helicopter is a sophisticated flight vehicle with lift, control force generation and propulsion incorporated in one device - the edgewise rotor. This integration of the various functions gives rise to some advantages and some disadvantages and, as always in life, it is the latter which stand out most clearly.

The operational advantages of the helicopter have led to it being regarded as the light truck or pick up van of the air rather than the thoroughbred streamlined executive aircraft. All too often one has heard the statement down the years - the helicopter is slow so the shape doesn't matter too much. This is a situation which is changing quite rapidly.

Various layouts of vertical take off rotor powered aircraft have been proposed and many flown. Is it likely that the day of the edgewise rotor is coming to an end, if so what will be its successor?

While writing this one realises that at anytime since the 1950s such a statement could be written. In fact Gessow and Myers classic textbook 'Aerodynamics of the Helicopter' is a summary of the problems and solutions of that period. When one looks at the improvements in aerodynamics which have been made since that time, the gains sometimes appear small in comparison with the effort expended. This paper will try to assess the situation in certain important areas.

2. ROTOR AERODYNAMICS

2.1 Blade Sections

The simple picture of a lifting rotor in hover as being equivalent to a pressure discontinuity is the basis on which low disc loading machines are specified when efficient extended hover is required. The model only gives a very simple and broad brush treatment of the general flow. In forward flight with the rotor at some incidence to the free stream it was traditional to depend on Glauert's hypothesis for the induced velocity, for once the induced velocity is determined it is possible to estimate the local forces and moments on the rotor blade and hence the performance and trim of the rotor. However, the induced velocity is the result of the production of lift and moments and these are influenced by many other factors - the most important of which are shown diagrammatically in Fig.1.

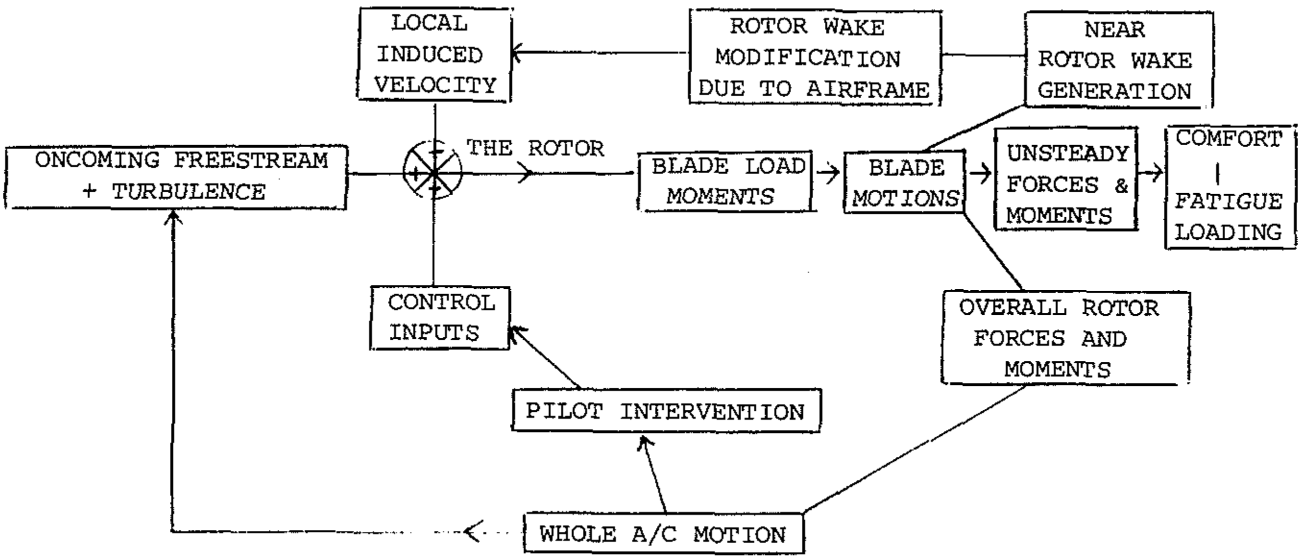


FIG.1 THE MAJOR FACTORS INTERACTING TO PRODUCE ROTOR FORCES AND MOMENTS

The blade loads and moments can be related to the performance of the isolated aerofoil which is used to form the blade. Until the early 1970's to talk of a helicopter blade section, at least in the Western World, was to mean the NACA 0012. This airfoil has lift, drag and above all pitching moment characteristics which enabled it to satisfy the drag rise criteria of the advancing blade tip region, the stall criteria of the retreating blade in forward flight and, above all, an acceptable pitching moment for the torsional motion of the blade and the implied control loads.

The use of NACA 0012 is, however, becoming a thing of the past due to the structural designer using new materials to give improved blade torsional stiffness which the aerodynamicist has exploited by experiment and theory in the design of new families of blade sections. The gains achieved are illustrated in Fig.2 which takes its examples from British, French and American research and contrasts the performance with NACA 0012.

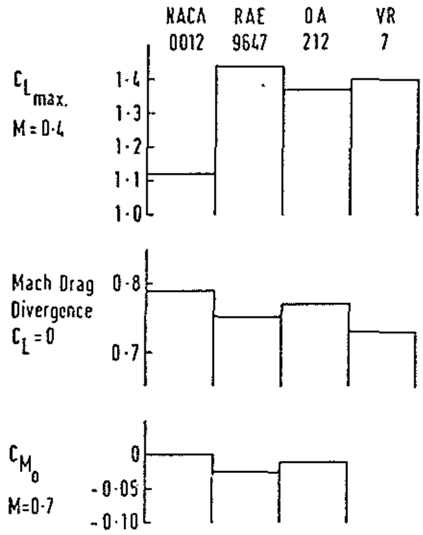


FIG. 2 SECTION PERFORMANCE OF SOME NEW ROTOR-BLADE PROFILES APPROXIMATELY 12% THICK CONTRASTED WITH NACA 0012

These results refer to sections which all have thickness chord ratios of about 12% and are chosen because for structural and aerodynamic reasons it seems that the inboard 60-70% of the rotor span is likely to remain at about this thickness. The gain in $C_{L_{max}}$ at low Mach number is clearly of importance in the retreating blade region. This increase in $C_{L_{max}}$ can be used to increase the thrust loading of the rotor (remembering that this will cost more power) and Wilby (1) has shown that for the same retreating blade stall margin, a 35% increase in $C_T/_\sigma$ is possible. Gains of this magnitude have been shown by Cook (2) for a cambered blade section tail rotor flown on a Westland Sea King.

The importance of low C_m values may be indicated by copying Fig.14 from reference (1) (Fig.3) which shows the root torsional moment for a Sea King size rotor using NACA 0012 blades and the calculated moment when the blade sections are replaced by RAE 9647. It will be noted that $C_T/_\sigma$ is 35% higher in the latter case. The implication of this change for the blade structural

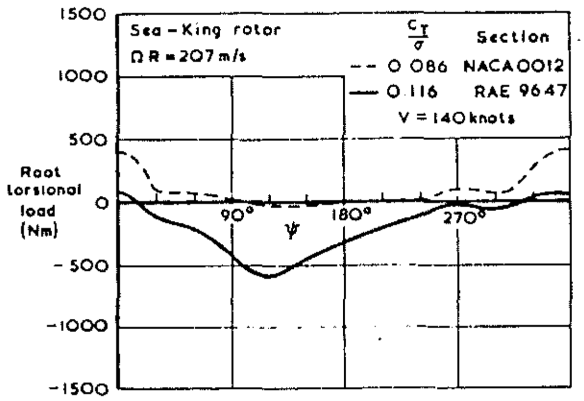


Fig. 3 Predicted variation of blade root torsional load with azimuth

designer is shown by assuming C_T and the blade radius remain the same so the blade chord reduces by 35%. For the same blade construction the torsional stiffness will reduce by 58% so the maximum value of $^{M}/_{GJ}$ will increase by 3.57 and change from nose up to nose down.

The optimisation of the blade section along its span is not new - Hafner used both changes in thickness and planform to optimise the aerodynamics on the Bristol Sycamore of 1947 vintage (Fig.4). It is worth noting that much of his hoped for gains over, for example, the Westland Dragonfly (anglicised S51) were nullified by the lack of structural stiffness and the poor section profile possible with the fabric covered blades.

Fig.4 Bristol Sycamore

However modern methods of composite construction, first practically used on the BO 105, have given designers greater freedom. Thibert and Gallot (3) have recorded what has been achieved using a 9% thickness chord blade fitted to the SA 365N Dauphin helicopter as shown in Fig.5 which is taken from Ref.3. The gain in blade performance may be used to increase thrust or speed or a combination of both.

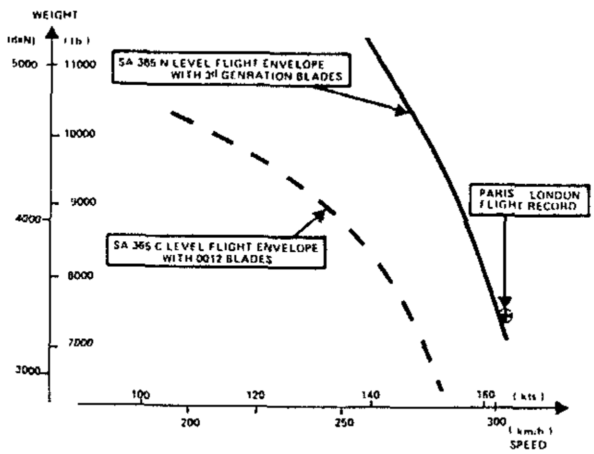


Fig. 5 Flight envelope extension with the 3rd generation blades

Before leaving this particular topic Boeing Vertol have used some of the blade aerodynamic gain to reduce tip speed on the model 234 in order to restrict rotor noise. The reduction of 7% in tip speed averaged for take off/approach/flyover 8 EPNdB which is more than twice as much as was expected. The improved blade loading both spanwise and chordwise clearly has contributed to this result.

2.2 Wake Development

As indicated in Fig.1, application of the new blade sections must be related to the understanding of the flow passing through the rotor. The determination of the flow character experimentally is difficult due to the large and often rapid changes in velocity. Techniques used, particularly in the hover, include pilot-static and hot wire traverses and laser anemometry. The results from the various methods are not always consistent and the interpretation is confused. Assuming that the rotor wake can be considered steady, the most successful and understandable models develop a wake based on the shed circulation from the blade.

To develop and use such a model it is necessary to make assumptions about the strength and distribution of the circulation which is shed and to determine the resulting flow field which must include the paths of the vortices themselves. Many models have been developed which range from the very simple models like that due to Willmer (4) for fast forward flight of semi-infinite or infinite trailing vortices which were tangential to the curved wakes just below the reference blade, through models which use a combined actuator disc plus horseshoe vortex pattern (5) similar to a fixed wing and which give some information throughout the whole speed range through the vortex ring model of Castles and de Leauw (6), to the full vortex models using either a prescribed (7,8), combined free-prescribed or entirely free wake model. The computing time rises by an order of magnitude for each of these developments. The problems of modelling the flow may be described under the headings (a) development of blade circulation model, (b) the mechanism of shedding the wake, (c) the ageing of the vortices and the conditions for stability of the vortex structure and lastly (d) the effect when a vortex comes close to, or intersects with, another vortex or a solid surface.

(a) The lift of the blade and hence the circulation description is usually obtained from isolated airfoil data in some cases modified to allow for dynamic effects as the blade incidence varies during each rotor rotation. Using the well known expression $\rho V\Gamma$ = lift/unit span the circulation Γ can be estimated and a simple lifting line model for the blade lift deduced. The changes in circulation over a given span may then be assumed to be shed at some point on that section and to stream away as the flow field of all vortex elements dictate. In addition as the blade advances through an azimuthal step $\Delta\psi$ the circulation may change in which case a starting vortex parallel to the blade span is shed. The lifting line representation of the blade used by Miller (9) and Piziali et al (10) and numerous authors since is simple and away from the tip is reasonably accurate. It, of course, cannot give any information on pitching moment or on three dimensional flow effects such as dominate at the tip of the blade. A lifting surface theory is then required. Several methods of obtaining such a solution have been developed in various centres of research. One method is to look at the solution of the potential equation allowing for small perturbations including transonic effects but assuming weak shock waves (entropy variations neglected). The method of solution is to solve the equations at a lattice of points, the more dense the lattice in general the more exact the solution. Such a method has been applied to the non lifting blade by various authors like Caradonna and Philippe (11), Grant (12) and Philippe and Chattot (13). Clearly such a method does not include provision for flow separation in the analysis. One conclusion which was demonstrated in Ref.11 was the importance of the unsteady terms in the potential equation.

On the retreating side the rotor blade is at high lift coefficient and may also be experiencing compressibility effects - clearly the above method is inappropriate. A paper by Send (14) suggests a method for incompressible flow based on the work of Prager and Martinson in which the viscous layer is represented by vorticity. A satisfactory solution to the representation of the blade is possible but the above paper only shows that the method can be applied to a Jaukowski 12% airfoil and a circular cylinder. Further developments will be watched with interest.

These methods are of particular value for shapes for which there is little previous information. In the case of airfoils and blade shapes which are similar to rotors already in use synthesised data which gives lift coefficients is available. This information will include the effect of incidence and Mach number as well as reversed flow and dynamic effects when near the stall (see Section 2.4).

It is now necessary to consider how the circulation on the blade becomes transmitted to the wake. Some excellent experimental surveys of the flow beneath a hovering rotor have been made. What was clear from hot wire records was that there was considerable fluctuation in the instantaneous vertical velocity in both an azimuthal and a spanwise direction (Fig.6).

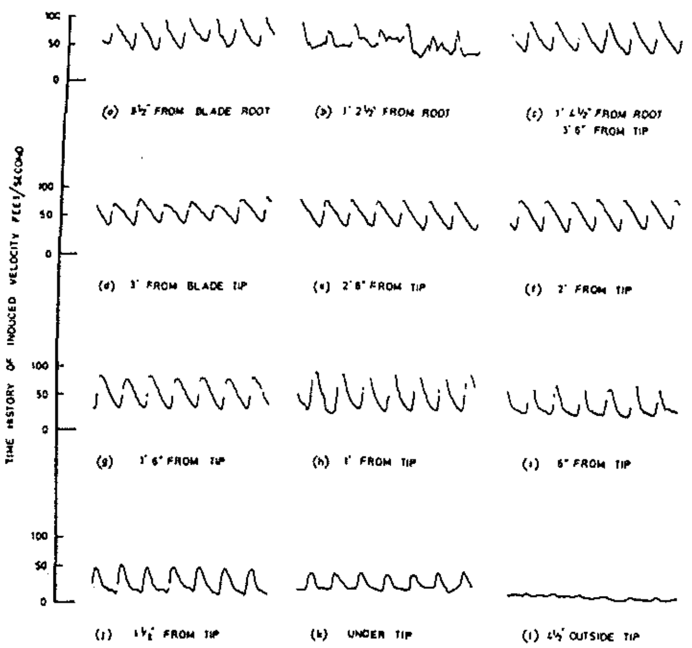
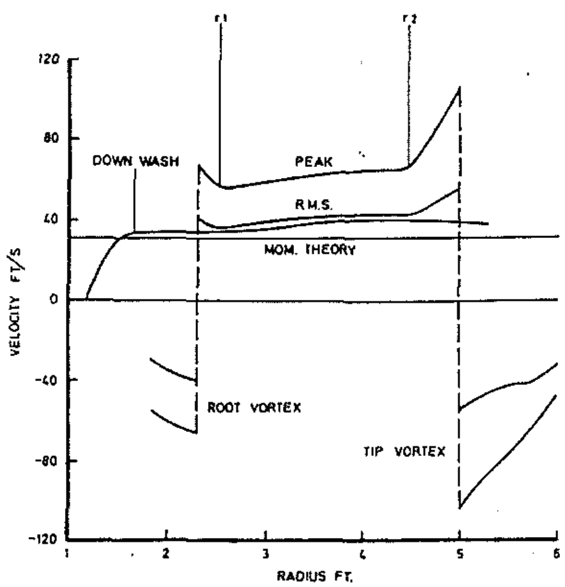


Fig.6 Induced velocity recordings at 580 lb lift and 390 rpm.



Induced velocities at rotor.

Furthermore velocity components plotted against time may show a form which is typical of a cross section through a vortex with solid body rotation core (Fig.7 taken from Ref.15). Vortices of this form can be found at an aximuthal distance behind the blade of 45°. The main problem is the interpretation of the signal which is the sum of the velocity due to the vortex in question and from all the other vortices in the wake. It is therefore necessary to eliminate the other vortices contribution in order to determine the circulation. Caradonna and Tung in Ref.15 give a good description of the various techniques which have been used and they assess them against experimental data taken from their hovering two-bladed rotor which had pressure tapped rotor blades and hot wire sensors located just below the plane of the rotor. Their results suggest that the full circulation

F. X. CARADONNA and C. TUNG

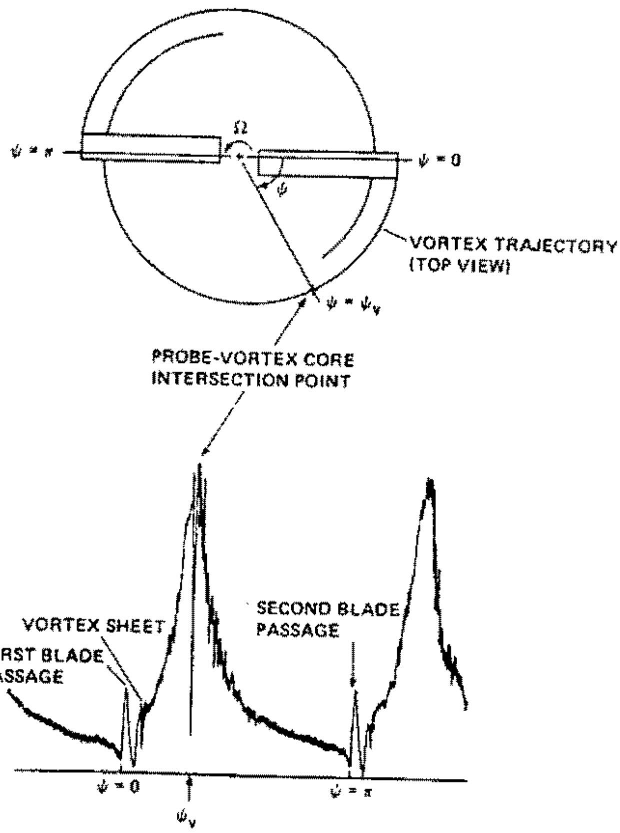


Fig.7 Typical wake probe data.

implied by the blade lift was found in a vortex which was only 50° old in azimuth and they contrast these with Cook's (Ref.16) results obtained from a full scale rotor run at full scale speed on a test tower which gave vortices that did not show the Rankine profile nor did it appear to contain 50% of the blade lift implied circulation. Caradonna and Tung suggest that the Rankine form of vortex is a less exact fit to their model data as tip speed increases, as illustrated by Fig.8 (taken from their paper). Andrew (Ref.17) has made an analysis of the limited information (Table 1 from Ref.17) on vortex wakes behind rotating and fixed wings. He defined the vortices by a core radius r at which the peak swirl velocity V_s is obtained and he suggested the following empirical relations

Source	Comments	A_r	M	Vortex Position	Experimental V_s/V_T	Approximating V_s/V_T $(\frac{t}{c}\bar{C}_L)^{\frac{1}{2}}$	Approximating V_s/V_T $(1 + \frac{6.6}{A_r})(\frac{t}{c}\bar{C}_L)^{\frac{1}{2}}$	Experimental r/c	Approximating r/c $(\frac{1-M}{\sqrt{M}})\alpha_g \frac{t}{c}$
Cook (27)	Rotor blade hot wire probe	41	0.53	75° azimuth	0.3	0.29	0.34	0.016	0.013
Simons et al (28)	Rotor blade hot wire probe	54	0.125	300° azimuth	0.23	0.25	0.28	0.05 to 0.075	0.042
Present Author	Rotor blade hot wire probe	28	0.102	120° azimuth	0.3 ± 0.03	0.25	0.31	0.074	0.068
Rorke et al (29)	Fixed wing wind tunnel test hot wire probe	4.2	0.2	2c downstream	0.48	0.22	0.57	0.02 to 0.03	0.018
Zalay (30)	Fixed wing wind tunnel test hot wire probe vorticity meter	5.6	0.133	6.5c downstream	0.6	0.26	0.57	0.03 to 0.05	0.036
Panton et al (31)	Fixed wing free flight hot wire probe	9.2	0.123	39.6c downstream	0.72 +.12 -.25	0.36	0.62	0.046	0.045
Iversen et al (32)	Fixed wing wind tunnel test hot wire probe elliptic tip *based on 92% chord	11.4*	0.135	3.25c downstream	0.42	0.35	0.55	0.050	0.049
Chiger et al (23)	Fixed wing wind tunnel test hot wire probe	5.33	.089	trailing edge	0.37	0.33	0.74	.079	.095

TABLE 1 TRAILING TIP VORTEX DATA

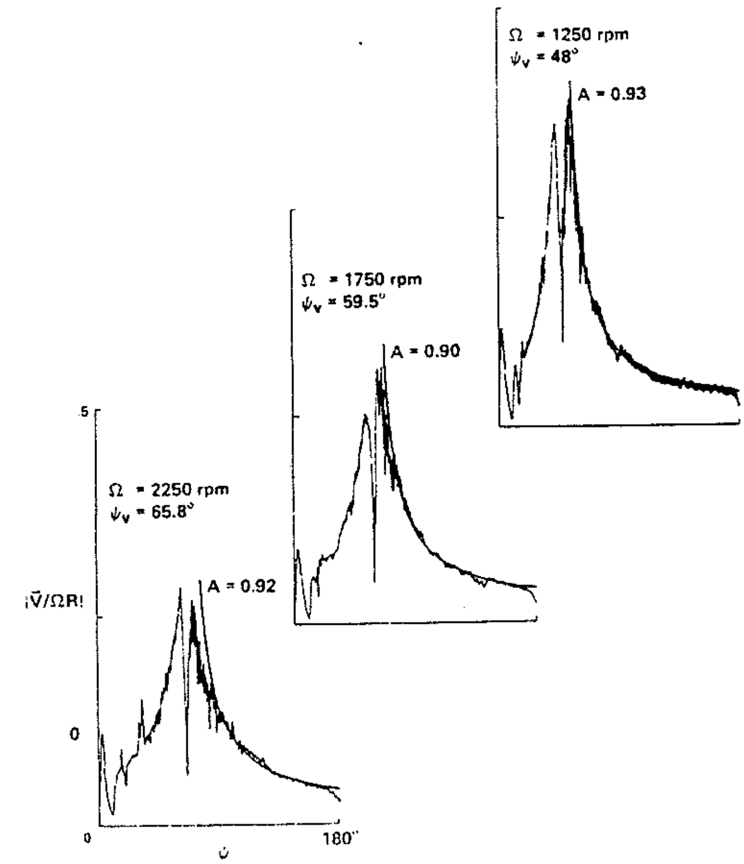


Fig.8 Typical velocity-time trace and I.R curve fit for various rotor speeds. Collective pitch. $\theta_c = 8°$. Vortex age, $\psi_v \cong 50$–$65°$.

$$\frac{V_s}{V_T} = \left(1 + \frac{6.6}{AR}\right)\left(\frac{t}{c}\,\overline{C_L}\right)^{\frac{1}{2}} \quad \text{and} \quad \frac{r}{c} = \left(\frac{1 - M}{\sqrt{M}}\right)\alpha_g \frac{t}{c}$$

where AR = aspect ratio = 2R/c for rotors

$\overline{C_L} = \frac{6C_T}{\sigma}$ for rotors $\left(C_T = \frac{\text{Thrust}}{\rho V_T^{\,2} \pi R^2}\right)$

α_g = the geometrical angle of attack

σ = solidity

V_T = tip speed

M = the tip Mach number

t/c = blade thickness chord ratio

$V_s r$ is a measure of the circulation shed.

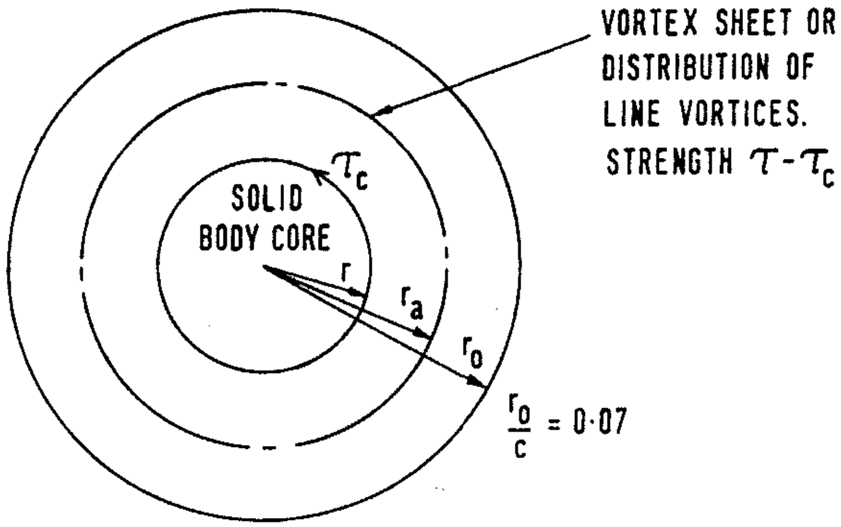
It is clear that V_S/V_T as a function of V_T is the same when the rotor is operated at the same thrust loading, but that the core radius expressed as a fraction of the blade chord decreases. It is therefore clear that the implied circulation $V_S r$ will decrease more rapidly than the tip speed increases. This statement appears to be inconsistent with Caradonna and Tung who suggest on a 'far field' agreement model that the blade circulation is in the wake. Examining Fig.7 and assessing the 'core' radius in Cook's way it is clear that r/c is decreasing with increasing tip speed as Andrew has suggested and that the circulation defined as peak swirl velocity times radius is decreasing. Incidently $r/c \approx 0.03$ which fits in between the model results of Simons and Andrew and the full scale values of Cook suggesting that tip speed is not the only important variable. Could it be shear layer thickness on the blade? The size of the solid body core is important as is the implied velocity just outside the core because as has been shown by many authors the first vortex passage beneath the succeeding blade is very close. In the early models with an inviscid vortex model velocity approaching infinity was induced on the blade with consequent meaningless values of local lift which in turn led to a converged wake solution that was in error. The actual mechanism by which the core of the vortex is formed and how it varies with age is a subject which requires much more investigation. Understanding of the process may also lead to methods of controlling the vortex roll up and hence the induced velocity on the following blades. Accepting that circulation must be conserved it would appear that for full scale rotors the vortex structure near to the blade may be considered to consist of a solid body which may be given by Andrew's formula, surrounded by a region in which there are a large number of vortex lines, the total circulation of which plus the effective vortex at the limit of the solid body equal the total circulation and then finally a normal line vortex fall off region corresponding to the total circulation as a line vortex when the distance from the centre of the core is some 5 or so core diameters.

A model which uses Andrew's formula for V_S/V_T and r/c to define the solid body core size and central vortex circulation Γ_C plus a series of individual vortex lines (the number depending on the sophistication of the programme) or a vortex sheet with total circulation equal to $\Gamma - \Gamma_C$ where Γ is the total circulation shed from the blade, these lines being placed at a radius r_a given by

$$\frac{r_a}{c} = (0.07 - \frac{r}{c})/2$$

which appear to satisfy the data to date (Fig.9).

In forward flight the form of the vortex and its path is of even greater importance. In flight at very low advance ratio the way in which vortices deviate from the hover path to give a large increase in downwash over the rear of the rotor disc and the lift variation and control moment which result was highlighted by the accident to the prototype Sikorsky XH59A ABC rotorcraft. Fig.10 taken from Ref.19 shows how the real induced velocities can differ from those of a simple analytical model. Although the coaxial behaviour was extreme the problem will occur with a single rotor machine and emphasises the need for a free wake solution. It is interesting that the simple Glauert hypothesis which suggests that induced power falls steadily with increasing advance ratio is not always true and both experiment and theory now suggest that at very low advance ratio the rotor power required may be slightly higher than in true hover.

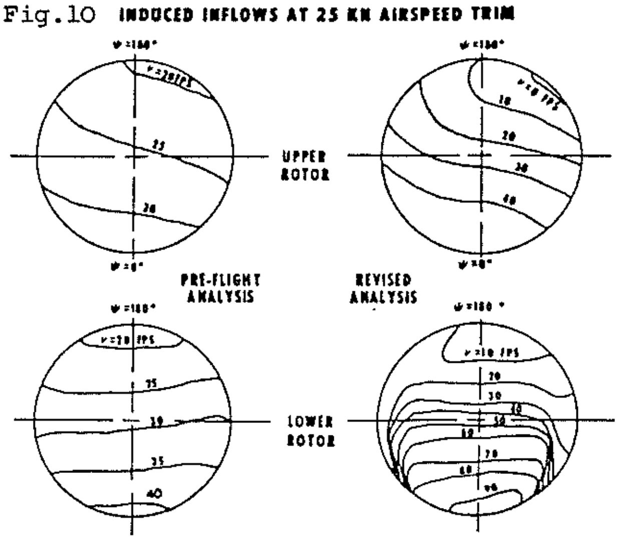


TOTAL CIRCULATION SHED FROM BLADE τ

FIG. 9 SUGGESTED MODEL OF SHED BLADE VORTEX

For a high advance ratio the vortices tend to cross near to, or even intersect, the blade in the forward quadrant and to pile up in the rear quadrant close to the plane of the blades. The various ways in which these patterns occur was illustrated by a simple model of Luccasson (Ref.20). Some of the vortices which are close to the rotor blades in the fourth quadrant are quite old. Probing of such vortices has indicated a growth of core size which is greater than suggested by eddy viscosity normally required to explain the growth of the core of say a trailing vortex from a fixed wing. This premature ageing of the rotor appears to be related to its curvature and to the fact that it is in a significant three dimensional flow which may induce axial velocities in the core which are known to give rise to rapid core growth and possibly to disintegration.

Fig.10 INDUCED INFLOWS AT 25 KN AIRSPEED TRIM



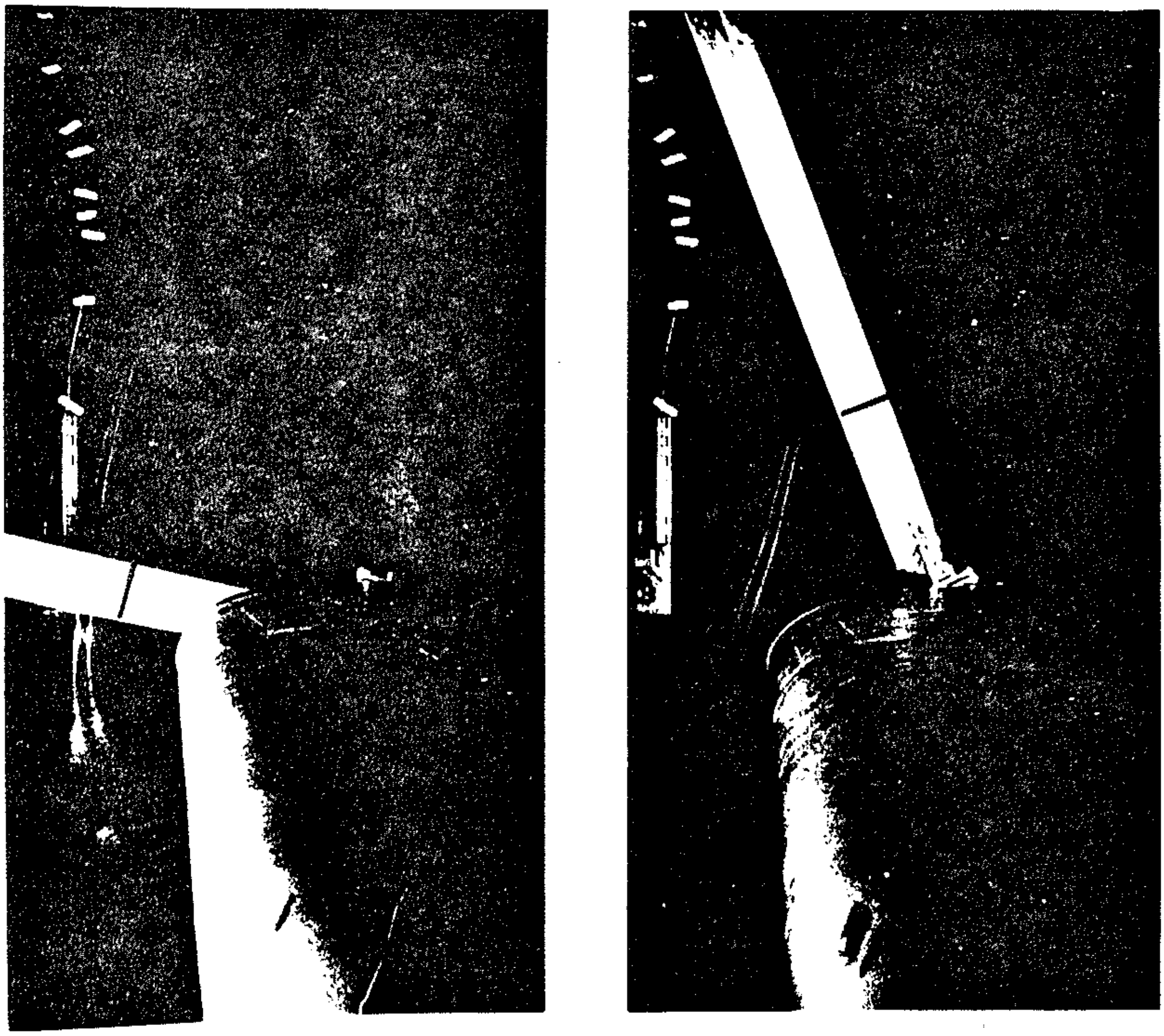


Fig.11

In wake models which allow for vortex/blade or vortex-fuselage interaction there is the question of the form of the vortex after impact. Flow visualisation pictures by Jones and Pacifico (Fig.11) have indicated that in some cases the vortex appears to stretch around the obstruction but to retain its form and therefore its effect can be modelled by assuming the vortex line is continuous and just above the surface which is modelled by image vortices in the standard way. Other flow visualisation photographs have shown that a vortex after blade intersection has become diffuse and ill defined and gives the impression that the regular structure has been totally destroyed. The need to conserve total circulation must be observed but the induced effect on the impacting blade produces a new circulation distribution which appears to result in a very diffuse circulatory flow after the blade. Langrebe and his co-authors (Ref.21) have shown velocity profiles before and after impact (Fig.12) as well as photographic evidence of such diffusion. The prediction of the subsequent form of the diffuse vortex is important for the accurate prediction of induced effects on succeeding blades.

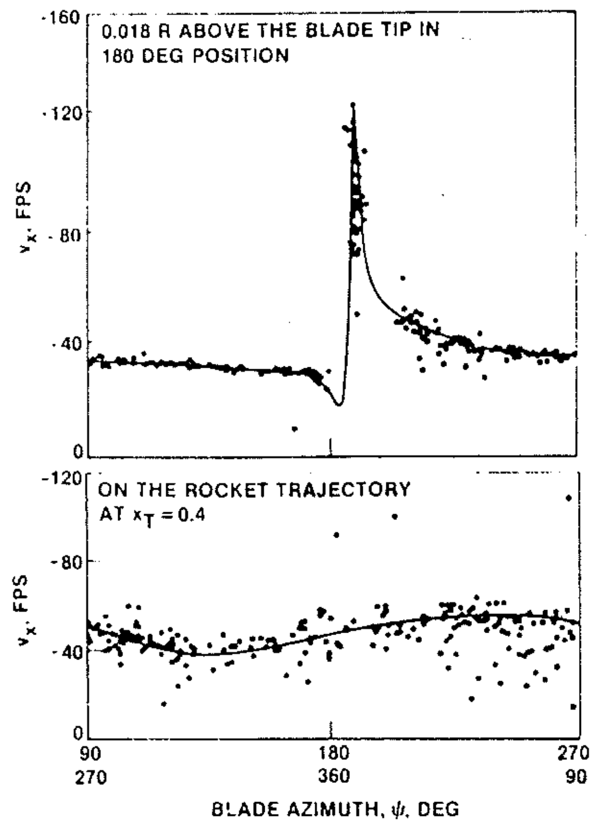


Fig.12 Comparison of LV data for points near the rotor and in the wake near the wake boundary for a low speed condition.

2.3 Unsteady Wake Effects

The discussion to date has assumed that there is no perturbation of the wake by extraneous forces nor is there any fluctuation in the lift on the blades due to elastic effects or to lost motion in the control circuits to name two effects. Any experimenter who has attempted to locate and measure the strength of rotor wake vortices will have noted how these move about a mean position. How important are these variations? An indication of the variation of the harmonics of the blade forces induced is shown in Fig.13 from Ref.40 in which a stiff bladed non articulated rotor has been operated in a low speed wind tunnel with good quality flow, the cyclic control adjusted to give trimmed flight and the resulting forces normal to the blade deduced from strain gauges mounted along the blade spar. Harmonic analysis of these signals for successive revolutions of the rotor have been

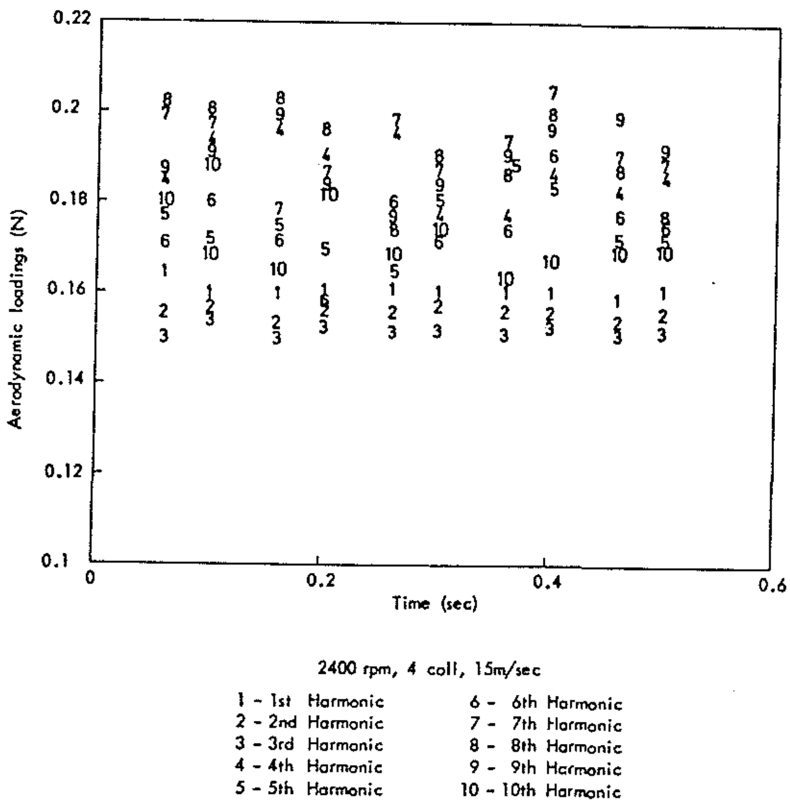


Fig.13 VARIATION OF HARMONIC LOADING Vs. TIME

made and the variation in amplitude of the harmonics (up to the tenth) determined. It is seen that up to the fifth harmonic the repeatability is good, but that the fluctuations become much larger as the harmonic number increases. This suggests that the use of a steady wake is adequate when there is no risk of blade stalling and that there is no near resonance between blade deflection modes and the higher order excitations. Alternatively the introduction of higher harmonic blade control may well increase the excitation of these higher order modes while reducing some of the lower frequencies and therefore a model which explains these variations is likely to be needed in the coming years. The analysis mentioned so far determines the mean trajectory of the vortices and corresponds to the formulation of the wake which has been described in hover in Refs.7 and 8. When control applications are made the change in rotor thrust and moment is required. An alternative to a transient wake analysis is to use an unsteady actuator disc theory. A family of solutions which satisfy Laplace's equation and give the required pressure discontinuity across the disc was solved by Kinner and these solutions may be compounded together with suitable amplitude and phase relationships to represent the rotor transition performance - a discussion which was opened by Pitt and Peters in Ref.22. Such a discussion leads naturally to the more extreme aspect of the problem when dynamic stall occurs.

2.4 Dynamic Blade Stall

The dynamic stall of the retreating blade has occupied research workers for as long as modern rotary wing aerodynamics have been studied. It is a well known fact that the flow does not break down and the blade stall is delayed beyond the static angle of attack if that angle is increased rapidly. The typical lift angle of attack hysteresis loop is generated which is a function of oscillating frequency and is shown in Fig.14 from Ref.23. "Point a illustrates the delay in divergence pitching moment. After a further delay (point b) the lift which up to the time has been calculated on the basis of attached flow, decays according to a simple exponential function of time toward the fully separated value. As the angle of attack reduces below α_1 (point c) a process of reattachment is initiated, using the separated value as an initial condition the lift change vs time is implemented using the same expression as for attached flow." It is, of course, easy to produce a two dimensional test in which an airfoil is oscillated in pitch, the oncoming flow oscillated in direction and the airfoil heaved in a steady airflow or moved transversely in the flow. For a rotor blade these effects are all happening simultaneously plus a changing sweep angle of the main flow and of the spanwise loading. It is clear that to perform a meaningful wind tunnel experiment is difficult. One interesting series of experiments are due to Maresca, Favier and Rebout (Ref.24) when they produced fore and aft, heaving and combined (oblique) motions. They

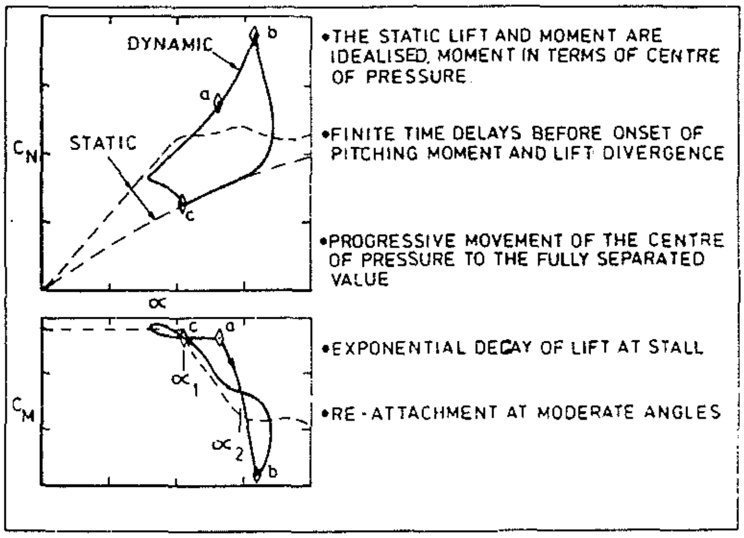


Fig.14 FEATURES OF THE DYNAMIC STALL MODEL.

showed the usual hysteresis loops in the first two cases when the incidence went beyond the static stall. In the case of the oblique motion they also considered the phase between heave and fore and aft motion. The effect of the phase had some effect but the mode of breakdown of the flow with the generation of the usual vortex along the leading edge which then developed over the upper surface remained the same as in the heave motion. This vortex was convected downstream with a velocity = 0.45 free stream and produced the overshoot in lift and drag. This result lends support to the idea of using an empirical model of this class of flow breakdown.

For use in aerodynamic performance and control prediction work it is necessary to synthesize data which may be used in computer programmes. Two examples among many may be cited - Beddoes (Ref.23) and Bielawa (Ref.25). Ref.23 examined 300 test cases and used 150 which demonstrated lift and/or moment divergence. Defining a time delay Δt for the lift and moment breaks he was able to produce mean values and confidence limits for the range of conditions shown in Fig.15 as well as the motion of the centre of pressure (Fig.16). Beddoes concluded, to his surprise, that within the range of

NON DIMENSIONAL TIME = $\frac{\Delta t\ V}{C}$

TIME DELAY:

	MEAN	STANDARD DEVIATION	SAMPLES
MOMENT BREAK	2·44	·49	142
LIFT BREAK	5·41	·61	123

RANGE OF TEST PARAMETERS:

REDUCED FREQUENCY	·04 — 0·67
MEAN ANGLE ~ DEGREES	2·5 — 20·0
AMPLITUDE ~ DEGREES	2·5 — 10·5
$\frac{\dot{\alpha} C}{2V}$ AT DIVERGENCE	-0·1 — +0·05
MACH NUMBER	0·2 — 0·6
NUMBER OF AIRFOILS	4
MODES OF MOTION	3

Fig.15 **STATISTICAL ANALYSIS OF TIME DELAYS.**

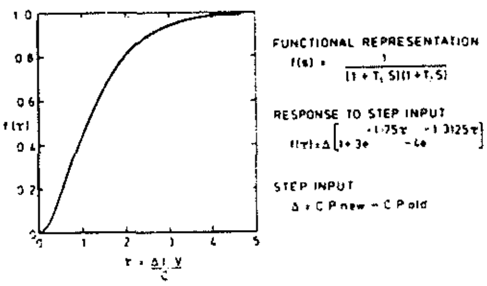


Fig.16 **CENTRE OF PRESSURE TRAVEL.**

conditions tested there is no significant dependance on frequency, mean amplitude, Mach number, mode of motion (c.p. conclusion from Ref.24) or even airfoil profile. It did depend on whether the angle of attack was increasing or decreasing and Beddoes accounted for the speed up with decreasing angles of attack by doubling the effective time interval. Refinement of this and similar methods continues.

3. FUSELAGE AERODYNAMICS

Bramwell and Clarke made an analysis of the drag of the non lifting components of a helicopter, the result of which is shown in Fig.17 which is taken from Ref.26. The comparison with a clean fixed wing aircraft makes a striking and depressing comparison.

A typical breakdown of helicopter drag was produced from wind tunnel tests and analysis by NASA for the LOH designs and Table 2 is taken from that summary. Two aspects of that problem will be considered here, these are the drag of the body and of the hub.

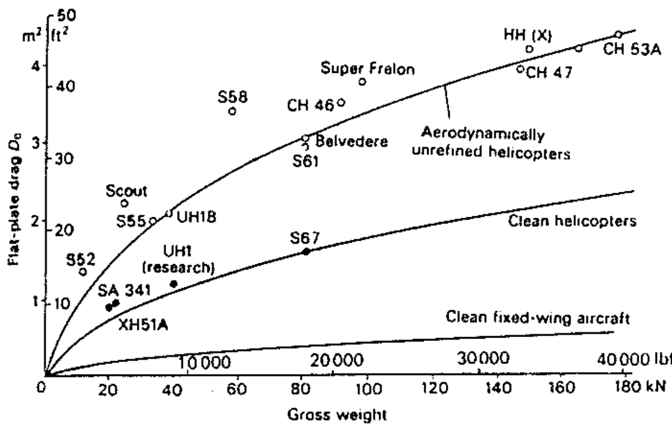


Fig.17 Parasite drag of helicopters

Parasite Drag	% of total
Fuselage	13
Pylon	3
Hub	20
Skids	8
Antennae, protuberances, etc.	13
Induced drag of above (3^{o} nose down)	2.5
Sideslip of above (3^{o})	5
Main rotor controls	3
Tail rotor (hub, controls etc.)	12.5
Leakage, engines etc.	12.5
Transmission cooling	5

Table 2

The helicopter is a beast of burden rather than a thoroughbred stallion. The fuselage must therefore be able to accommodate bulky and awkward shape loads, allow ingress and egress from a variety of points as well as providing a platform for the propulsion devices which must have easy access. In this respect the helicopter has some affinity with the large transport aircraft like the Belfast, Starlifter and Galaxy - all of which have had drag problems. An excellent summary of the rotorcraft drag problem is given in Ref.27.

One point which is clear to all is that the helicopter with its large protrusions has a rapidly changing cross section - a typical example for a single rotor aircraft is shown in Fig.18. The external flow in accelerating around such a shape experiences high levels of super velocity which increases the skin friction drag. The rapid contraction of the aft end of the fuselage is conducive to after body flow separation. This flow separation may lead to flows with strong incipient vorticity or to distributed vorticity. These two types of flow not only lead to different base pressure but the concentrated vortex flows may induce undesirable

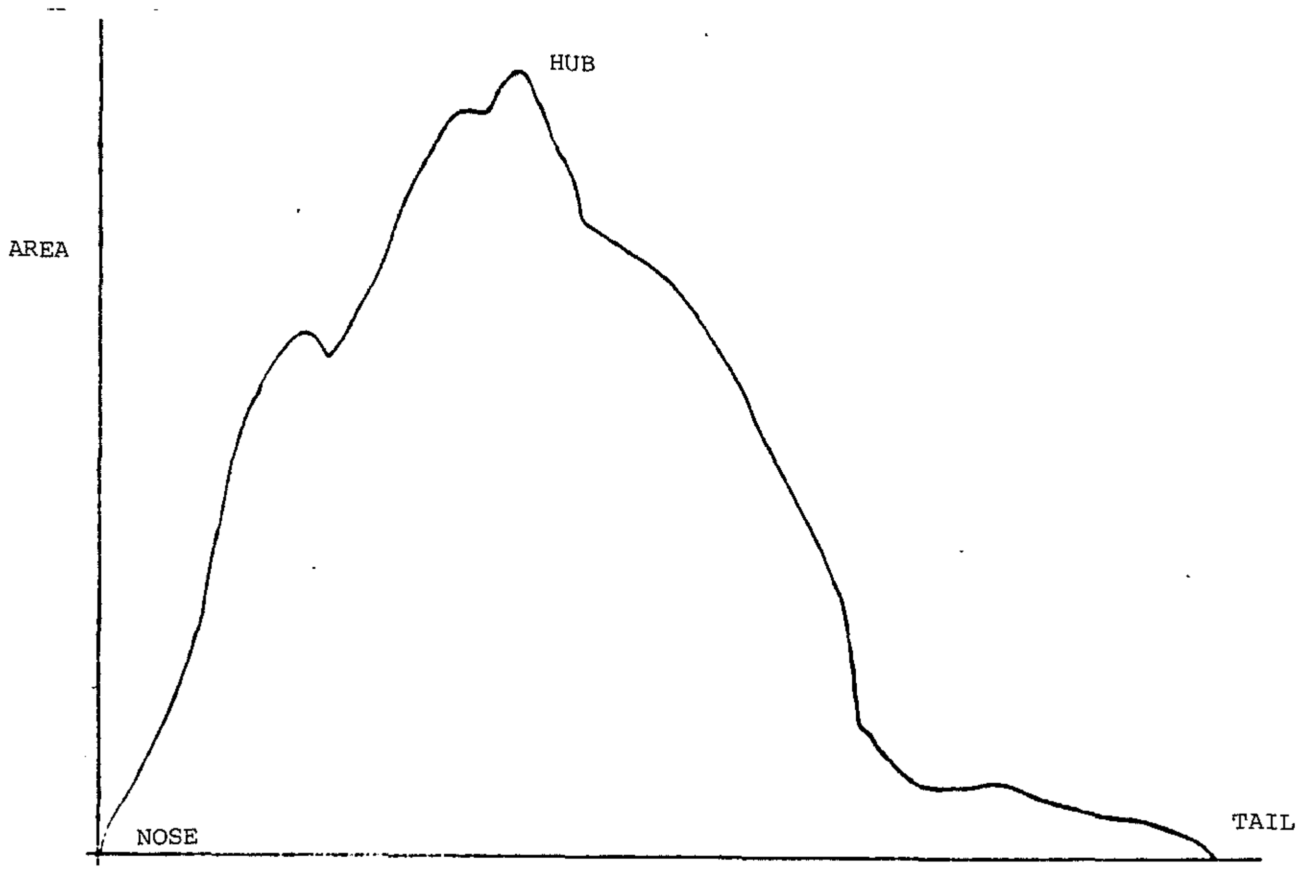


FIG.18 CROSS SECTIONAL AREA DISTRIBUTION OF A HELICOPTER

velocity components at the tail rotor with possible adverse control effects. Work done on car aerodynamics may be in advance of thinking in the helicopter industry. For example, the work of Morel (Ref.28) has shown two flow patterns as illustrated by Fig.19. For a circular cylinder with sloped rear end the pressure coefficients at three tappings are shown in Fig.20 (N.B. $\alpha = 0$ corresponds to a right circular cylinder). Morel showed that the flow patterns in regime I and II were each stable and that the changeover occurred with $47.25^\circ < \alpha^\circ < 48^\circ$. If the 47.25 model was inclined tail downwards by 5° so making $\alpha \approx 52^\circ$ the flow changed from regime I to II and returning the model back to horizontal did not bring the flow back to regime I - to do this it was necessary to incline the model tail up by 1° thus giving a hysteresis effect over about 6°. Attitude changes greater than 6° are possible in helicopters in manoeuvring flight at speed and if Morel's effects were to occur this may lead to handling problems.

It is clear that an area rule concept needs to be applied to fuselage design and that a method of calculating the velocity at all points on the fuselage including the separated flow is ncessary. Perhaps the best known work on this subject is in Refs.29 and 30 in which the fuselage is represented by a series of panels to enable the potential flow around the basic shape to be determined. This information is then used to calculate the boundary layer development, including separation, which then defines a new equivalent body shape for the process to be repeated iteratively. The method at present does not allow the separated flow to deform or the shed vorticity to roll up. It also does not allow for the rotor flow which may be very important in low and medium speed conditions.

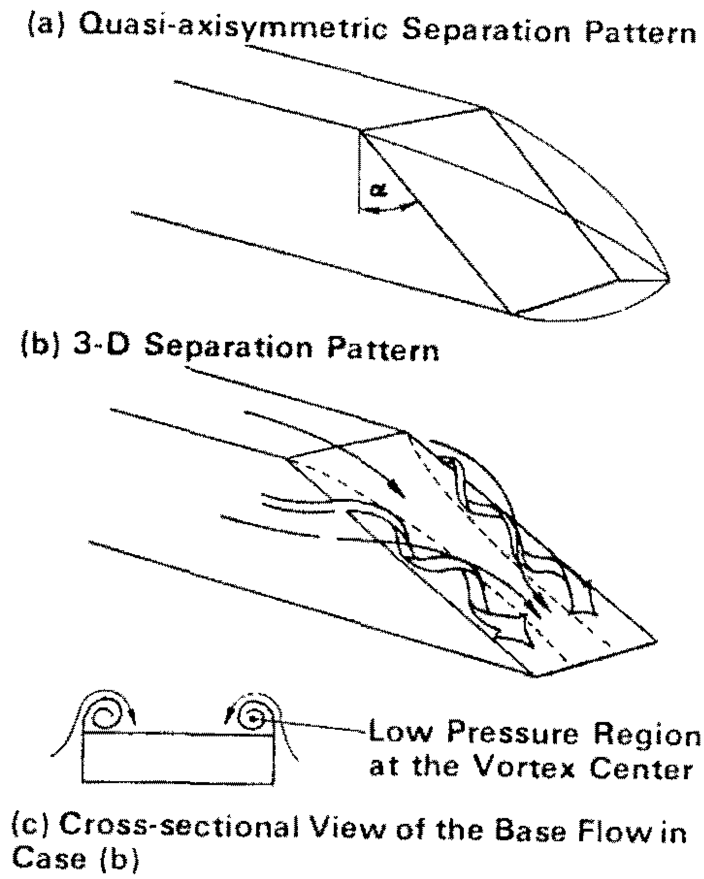


Fig.19 Two types of separated-flow pattern on a slanted base.

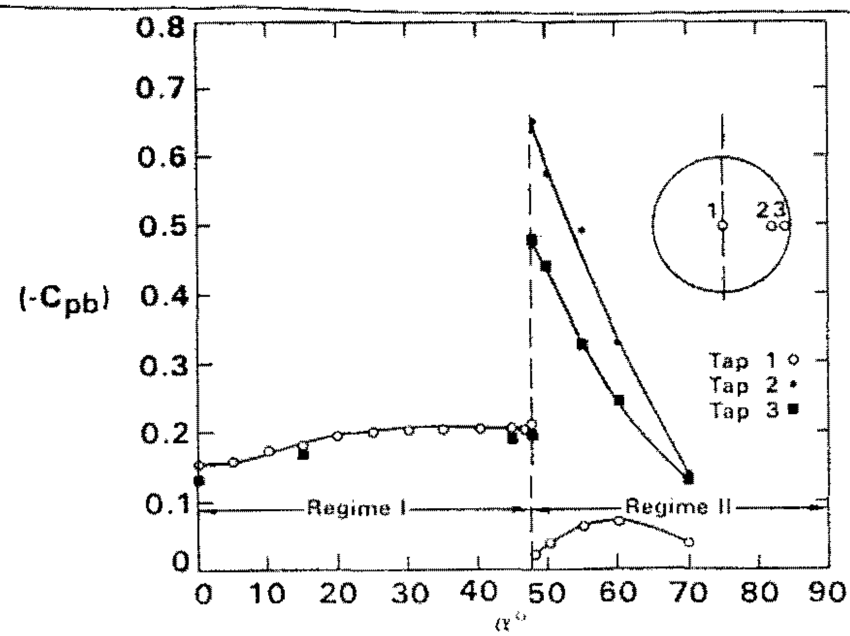


Fig.20 Base-pressure variation with slant angle.

One demonstration of the effect of rotor flow on fuselage forces was experienced in the Sikorsky XH 59A and reported in Ref.31. This aircraft had a very smooth cylindrical fuselage and when hovering (doors closed) the separation of the downwash from the fuselage was not fixed. As the separation point wandered lateral forces were generated (Fig.21) which were undesirable. The separation point was fixed with strakes which was satisfactory for a research vehicle but undesirable for high speed production aircraft. As Jenney points out - perhaps perfectly cylindrical fuselages should be avoided on rotorcraft.

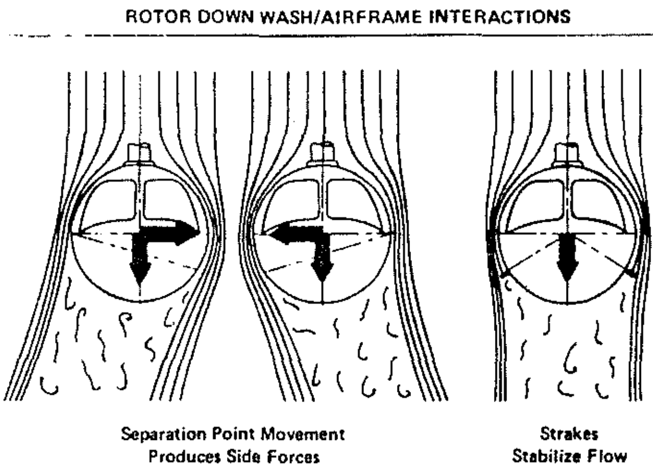


Fig.21

Table 2 brought out the contribution of hub drag to the total drag of the rotorcraft. The requirement to trim the rotorcraft means that the hub has to occur over the position of maximum depth of the fuselage and hence the chance of it operating in the highest supervelocity is great. The presence of the hub of course contributes to the area of the fuselage unless it is placed on a long rotor shaft like the Bell 47 or some other light helicopter - a solution which has other problems.

Seddon (Ref.32) has made a careful basic study of the drag of a hub by breaking down the drag into the various components. The paper concluded that the hub drag D was given by

$$D/q = \alpha\, C_D(A_p - A_z + A_s)$$

where $q = \frac{1}{2}\rho V^2$, α is the azimuth factor to allow for the change of geometry as the hub rotates (0.92 suggested), C_D the drag coefficient of a circular cylinder at the appropriate Reynolds number, A_p projected frontal area of head, A_z area allowance for fuselage boundary layer and A_s area for flow spoiling on canopy. The relation of these factors to details of the hub and canopy are described in the paper. The hubstested in Seddon's paper were clean semi-rigid configurations. When the method is applied to more complex hubs the result is usually an overestimate of the drag because the shielding effect in one part or another is not correctly estimated. The Seddon method does not allow for rotor head fairings which have been considered by Sheehy (Ref.33). The development of the new hub technologies which dramatically reduce the need for inspection and routine maintenance make the use of fairings acceptable although operational requirements like blade folding will restrict the size and shape of the fairing.

4. AERODYNAMIC DEVELOPMENTS

The future of rotor aerodynamics and the future of the edgewise rotor cannot be separated. Consideration is therefore given to the potential for the edgewise rotor as used in the helicopter.

The development of $C_{L_{max}}$ has led to a significant increase in thrust loading (≈ 35%) on the main rotor but because the edgewise rotor is not an efficient method of producing horizontal force the gain in forward speed is only about 6%. Thus for thrusting devices, like the tail rotor, full advantage of the thrust gain is taken (at an increase in power required) but for the main rotor this gain has to be seen in terms of a more compact (higher disc loading) rotorcraft with the possibility of a lighter basic weight machine and therefore better payload fraction. For conventional airfoils $C_{L_{max}}$ could become 2.0 provided that the pitching moment can be accepted. In fixed wing aircraft where higher lift coefficients are required the result is achieved by variable geometry - flaps, slats or other high lift devices. The requirement of high speed aircraft for take off and landing maximum lift and efficient lift/drag ratio at cruise is similar to the edgewise rotor advancing and retreating blade situation. Why then does the rotorcraft designer not take advantage of the fixed wing designers experience and knowledge. Such an idea is not new - an experimental rotor was run with flaps in the early 1960's and the system was shown to work. However the mechanical difficulty of such a scheme at that time made it unattractive but new control systems and, above all, materials and manufacturing technology make it reasonable to reopen such an investigation.

The use of controlled sweepback in non-rotating wing aircraft has now progressed so that the swing wing aircraft is now conventional rather than exceptional. The edgewise rotor has continuously varying sweep angle which varies along the span. Swept rotor tips have been introduced with clear advantages on the advancing side but with some disadvantages on the retreating blade. A variable sweep blade tip would be of value - and that might be one description of the lead-lag rotor tested in Germany.

The variation of spanwise loading is something which many would wish to achieve as the forward speed increases. Various experiments - like the Hill Isoclinic wing - which was intended to twist as airspeed became transonic and so minimise the adverse lift effects with changing pitching moment have been tried. Certain rotor blades in the past have produced change in lift with speed and weather conditions but these have been unintended. From the brief summary presented in this paper it is clear that the tools are now becoming available to describe the forces and moments which will arise in various flight conditions and with differing blade shapes. The structural engineer and the dynamicist appear similarly equipped with tools and understanding to devise a structure which can deform to a desired pattern. But what may this achieve? It is conceivable that thrust loading could increase by a total of 70% and the forward speed by perhaps 20% - both gains worth realising but hardly marking a step forward in rotary wing flight comparable to the gains that the swept wing gave relative to the straight wing.

Assuming that the edgewise rotor is retained what are the strategies which can make a quantum jump in performance. One philosophy is to pretend that the retreating blade does not exist and work entirely on the advancing blade which is the Sikorsky ABC concept. The coaxial rotor has some attractions and it is clear that this concept has not been fully exploited aerodynamically but reference to hub drag shows one of the disadvantages as can easily be seen from a Seddon type analysis. Assuming that the solution is sought in one lifting rotor rather than two then either one will work at low advance ratio (≈0.5) which means supersonic tip speed (Ref.35) or the advance ratio must increase which implies that the retreating blade and, in particular, the reversed flow region must work efficiently. The requirement for very high lift coefficients has led to consideration of blown augmentation systems such as the jet flap being considered by the Dorand Company in France and which resulted in a series of valuable tests in the USA (Ref.36) or the circulation controlled rotor currently being actively pursued by the David W. Taylor Naval Ship Research and Development Center. One major problem with blown rotor systems is the power required to compress the air which is used to fix the blade circulation. The clear advantage of circulation control over the jet flap is demonstrated by the fact that for C_L values of 4 or more, the lift/slot thrust augmentation ratio C_L/C_J is typically 4 → 5 for the jet flap and 10 or more for the circulation control for similar thickness airfoils.

In achieving high advance ratios the recent work reported in Ref.37 is of interest where a model circulation controlled rotor has been tested at $\mu = 1.4$. Fig.22 taken from that report shows the variation in power required through the transition - the peak at $\mu = 0.7$ standing out most clearly but of greater significance is the acceptable value of C_p/σ above $\mu = 1.0$. For interest some predicted values for an early idea of such a rotor operating at a $C_T/\sigma = .103$ are taken from Ref.38 and shown by the + symbols. The advances made in understanding the induced flow above $\mu = 0.7$ and the slightly better C_L/C_J performance of the blown airfoil at high lift coefficients account for the difference.

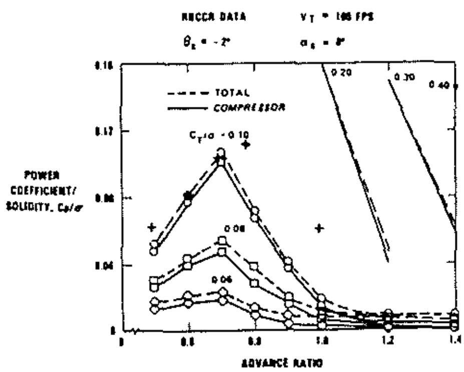
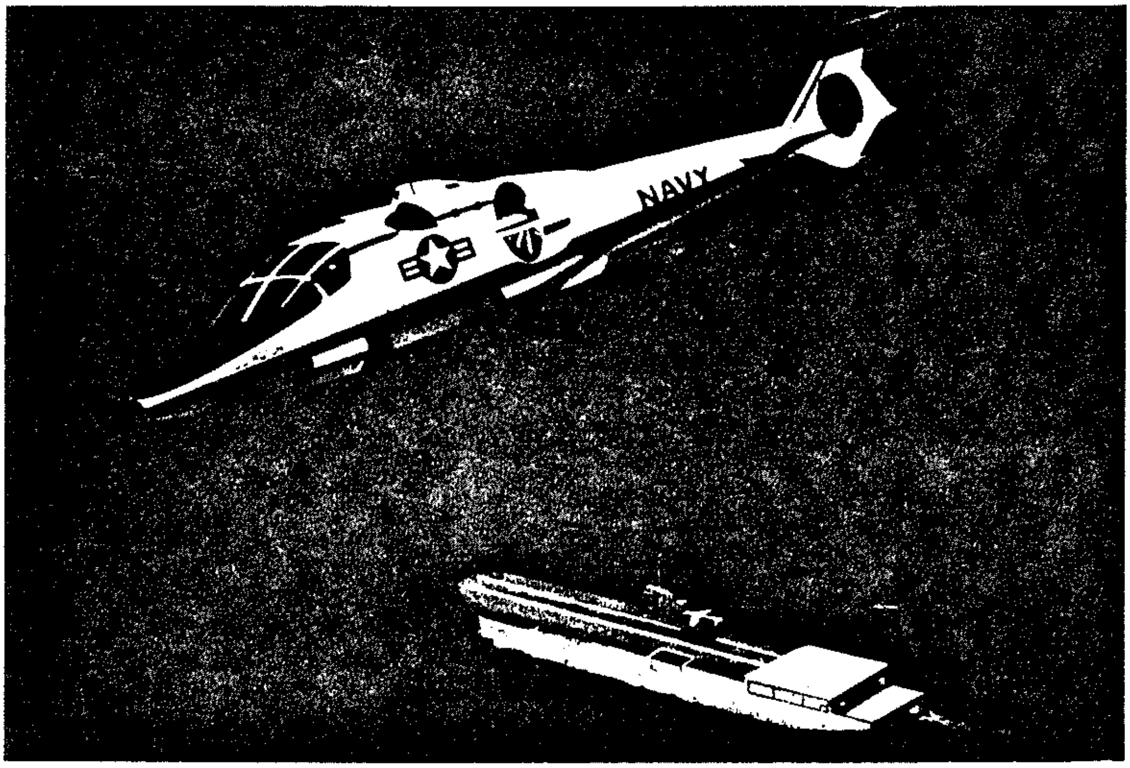


Fig.22 Power Trends at High Advance Ratio (Experimental Results)

It must be noted that it is assumed that ancilliary propulsion will be used for advanced rotorcraft. Ref.39 considers the limits to the speed at which a lifting rotor could produce sufficient horizontal thrust and suggests with modern technology included forward speeds will not exceed 225 knots. Apart from disc inclination the only area of the rotor which can produce a propulsive force is the drag of the outboard section of the retreating blade - so the development of high induced drag in this region produces some advantages although it still increases the rotational power.



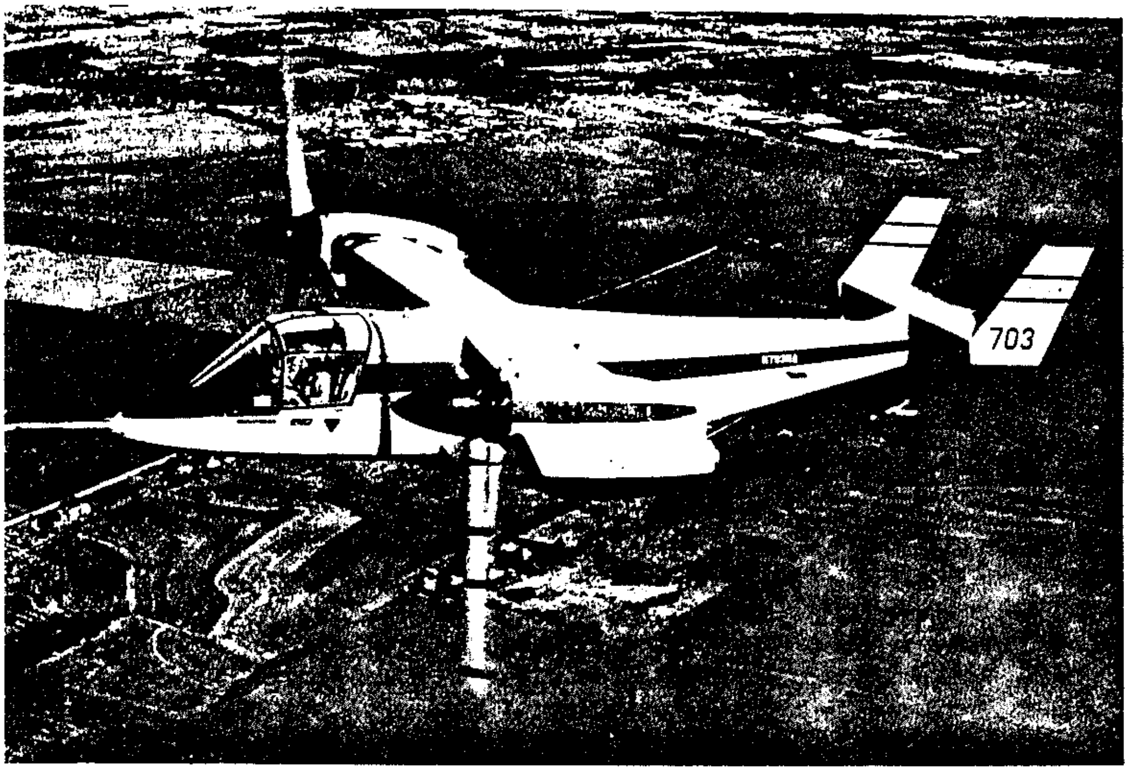


Fig.23 Alternative Forms of Future Rotorcraft - CCR or XV15

A personal summary of aerodynamics for the future is:

Aerodynamic research for the conventional helicopter edgewise rotor as far as performance is concerned has reached a point of diminishing return. Higher thrust coefficients are possible with more cambered blade sections if the pitching moments and the increased power requirements are acceptable.

Developments of advanced tips and greater understanding of the development of the induced flow to minimise induced power in the hover and low speed flight regime have increased potential. Active control of an advanced nature may also be of increasing importance in achieving better figures of merit.

The development of cleaner fuselages and hubs must have a high priority. In the case of hubs the drag of the coaxial rotor appears unattractive at high speed but for low speed applications the layout has many advantages which could repay further study. If higher speed performance is required the only possible objective seems to be rotors operating at advance ratio well in excess of unity (stopped rotors $\mu = \infty$). The only possible way to date of achieving this is by circulation control. The disadvantage of such a technique is the need to use compressed air to generate the lift in regions where a normal airfoil can operate very successfully. There is clearly scope for both innovation and the application of engineering science in parallel to find a solution.

The alternative seems to be to say that the edgewise rotor is rapidly approaching the limit of its development and that the future lies with other designs like the interesting Bell XV15 (Fig.23).

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