THE CHAIRMAN

The second paper this afternoon is to be presented by MR R HAFNER and deals with some aspects of the Bristol Type 171 Helicopter MR HAFNER was educated at the Technical College in Vienna and has been solely engaged on rotary wing development for more than twenty years, having produced his first helicopter design in 1927 He came to England in the early 30's and continued with helicopter development and then turned his attention to the design and construction of a gyroplane which was known as the A R 3 and which aircraft proved to be very efficient and pleasant to fly

During the early part of the late war, he collaborated with DR BENNETT on various rotary wing projects under the M A P , and joined the Bristol Aeroplane Co , as Chief Designer of the Helicopter Division in 1944 in which capacity he is entirely responsible for the design of the Bristol 171 I can speak at first hand of the excellent qualities of this aircraft having been privileged to carry out the first few hours of the prototype flying I now ask Mr Hafner to present his paper

---

The Bristol 171 Helicopter

By RAOUL HAFNER

The technical problems confronting the helicopter designer are very clearly enumerated in Captain LIPTROT'S paper, and Wing Commander BRIE has drawn attention to a number of considerations of importance to the operator

With some of these problems I have dealt in an earlier paper[1], and in order to avoid repetition I propose to leave out from this discussion points which have already been raised

VIBRATION

Rotating-wing aircraft in forward flight are subject to vibrations for reasons which are fundamental and we must not therefore expect the elimination of these symptoms but only their reduction to generally tolerable proportions We need, therefore, more accurate methods of recording and analysing vibrations in helicopters in terms of component frequencies and amplitudes, and in addition we must have generally agreed standards for comfort, *i e* , limiting vibration levels as a yardstick both to makers and users of these aircraft

As to the causes of rotor vibrations there is in the first instance a change of velocity with blade azimuth the maximum being at the advancing and the minimum at the retreating side of the rotor orbit, and secondly a change of inflow angle with blade azimuth due to the coning of the rotor blade and the curvature of the airflow in the vicinity of the rotor, the maximum being aft and the minimum forward of the rotor centre Therefore, in order to maintain a constant rotor thrust (or constant blade lift) during rotation, which is one of the essentials for freedom from vibration, the blade incidence must be varied in a cyclic manner Because the factors governing blade feathering are complex, the feathering motion cannot be expressed mathematically in a simple form but only by an infinite Fourier series * Typical values for the Fourier coefficients A_0, A_1, B_1, A_2, and B_2 for a conventional rotor are given in Figure 1 as a function of the tip speed ratio μ. A_0 represents the constant part of the blade incidence which can be obtained by the collective pitch control. A_1 and B_1 relate to the fundamental harmonic of the feathering motion which can be produced by a simple tilt of the control orbit with respect to the rotor orbit. The conventional swash plate or spider control or the tilting of the hub in an articulated rotor will produce this effect. The coefficients A_2 and B_2 relate to the second harmonic of feathering motion. For small tip speed ratios these coefficients (and those of the higher harmonics) are negligible so that the conventional rotor control which can provide a simple sinusoidal incidence variation during rotation, does satisfy to a fair degree the theoretical requirements. However, at larger tip speed ratios, the higher harmonics, which cannot be produced by the conventional control mechanisms, become predominant components. Thus at higher translational speeds the blade lift cannot be held constant any longer during rotation and vibration arises.

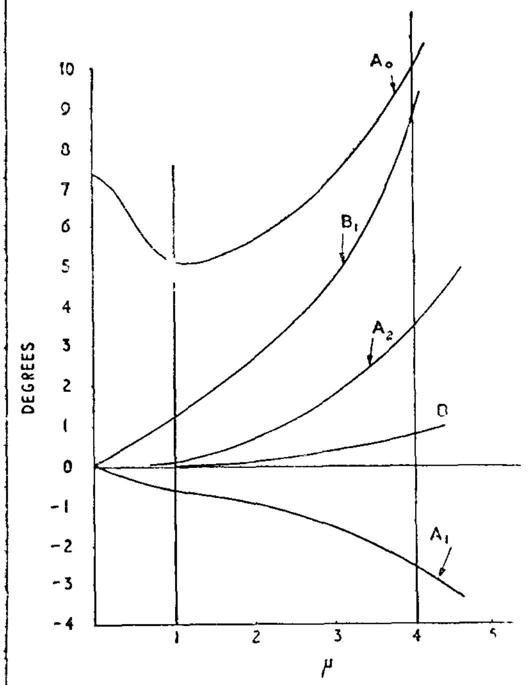


FIG. 1

Fourier coefficients of feathering motion for conventional helicopter rotor in level flight

$\theta = A_0 - A_1\cos\psi - B_1\sin\psi - A_2\cos 2\psi - B_2\sin 2\psi -$

One can, of course, think of mechanisms which are capable of producing cyclic movements of a more complex form including the higher harmonics referred to above. The mechanical elaboration involved in any such scheme is, however, considerable and in my opinion prohibitive and I regard therefore as a practical limit for conditions of rotating wing flight the tip speed ratio μ_{lim} where higher harmonics just become noticeable.

The question arises thus: Can the Fourier coefficients be controlled by suitable design parameters and other means, with a view to suppressing

*Blade pitch $= \theta = A_0 - A_1 \cos\psi - A_2 \cos 2\psi$
$- B_1 \sin\psi - B_2 \sin 2\psi +$ higher terms in ψ

A list of those symbols not defined in the text will be found at the end of the paper.

the higher harmonics over as wide a range of μ as possible? As already mentioned non-linearity and indeed discontinuities in the functions governing blade feathering are the principle causes for the higher terms, which I propose to discuss briefly

The following assumptions are made —

1 The resultant of the blade lift is acting at approx $\frac{3}{4} R$

2 The blade must be substantially free from stall outboard of this point.
If the mean lift coefficient during hovering (Ref 2) is

$$C_{L\,basic} = \frac{\text{Blade Lift}}{\frac{1}{2}\,\rho\,\omega^2 \int_0^R cr^2\,dr}$$

then the lift coefficient in translational flight can be shown to be

$$C_L = C_{L\,basic} \times F$$

$$\text{where} \quad F = \left\{ 1 + (\mu/\tau)\ \sin\psi \right\}^{-2}$$

$$\text{and } \tau = r/R$$

F reaches a maximum when the blade is retreating, *i e*, (taking $\tau = \frac{3}{4}$)

$$F_{(\psi = 270)} = F_{max} = \left\{1 - \tfrac{4}{3}\mu\right\}^{-2}$$

and a limiting condition is therefore obtained when

$$C_L = C_{L\,basic} \left\{1 - \tfrac{4}{3}\mu\right\}^{-2} = C_{L\,max}$$

$$\text{or} \qquad \mu_{lim} = \tfrac{3}{4}\left(1 - \sqrt{\frac{C_{L\,basic}}{C_{L\,max}}}\right)$$

A high tip-speed ratio can therefore be obtained only where the blade is flying at a low basic lift coefficient with an aerofoil capable of producing a high $C_{L\,max}$

The above expression can be written in another form, *viz*

$$V_{lim\,(\mu)} = \tfrac{3}{4}\,(V_T - V_{T\,min})$$

where $V_{lim\,(\mu)}$ = limiting translational speed (μ_{lim})

V_T = $R\omega$ = blade tip speed

$V_{T\,min}$ = the minimum blade tip speed when $C_{L\,basic} = C_{L\,max}$.

$V_{lim(\mu)}$ for the rotor of the Bristol Helicopter 171 Mark 3 is given in Figure 2 for various altitudes In the same graph is similarly shown a high speed limitation $V_{lim\,(M)}$ which represents the critical Mach No for the airfoil near the blade tip An airfoil with a low t/c ratio (below 10%) is obviously desirable in this region, in order to delay shock stall

The area within the limits shown in Figure 2 can be utilized by the flight envelope of the rotor, which is, of course, subject to such additional limitations as may be determined by strength considerations or operating conditions of the power unit, etc The rotor flight envelope is defined in the cockpit by a very wide r p m range for landing purposes which applies to speeds below 35 m p h, a smaller normal r p m range applicable up to the speeds indicated on a special scale on the altimeter, and a narrow r p m applicable up to the maximum speed and for flight manoeuvres involving high normal accelerations

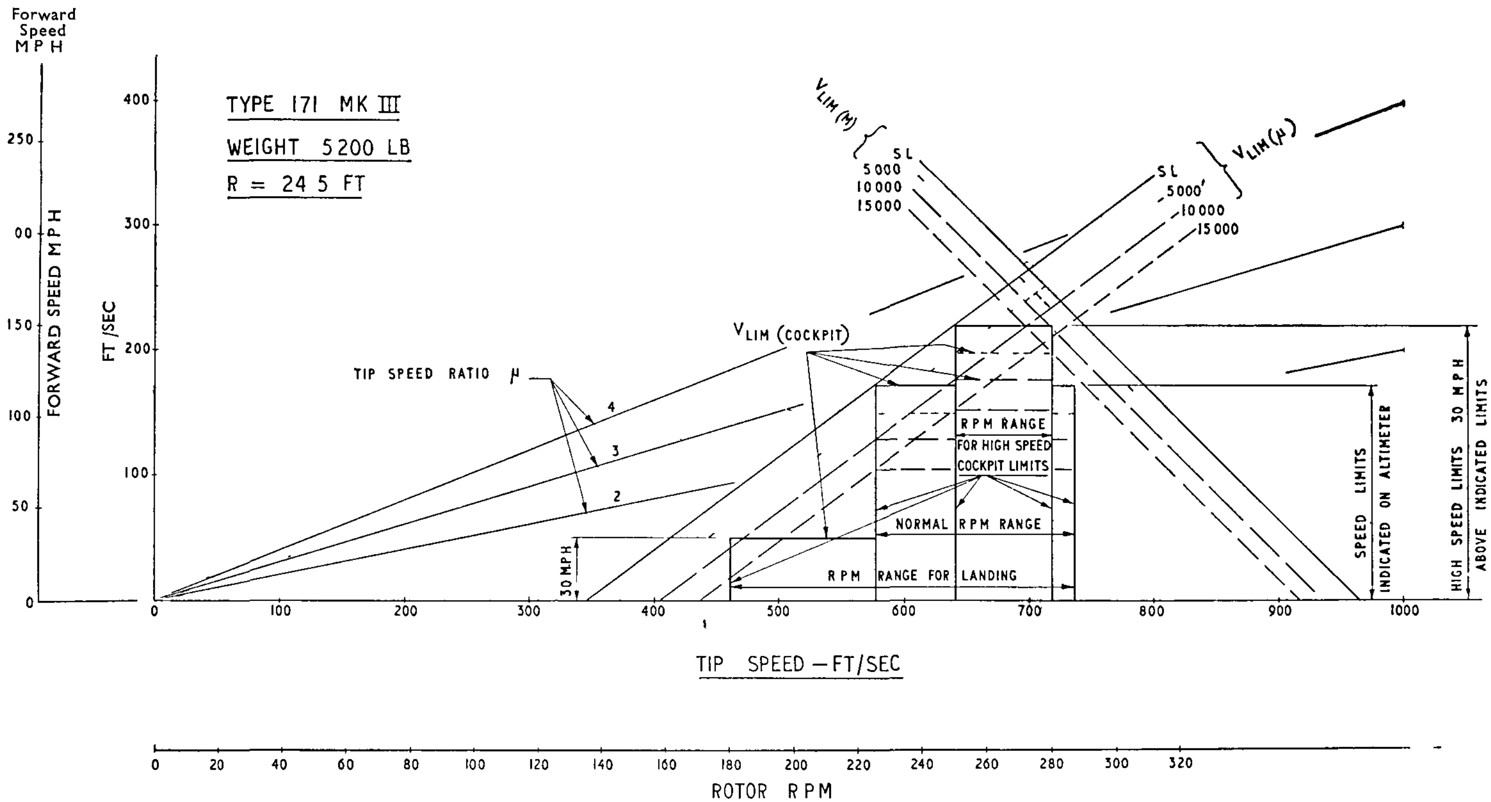

TYPE 171 MK III
WEIGHT 5200 LB
R = 24 5 FT
Forward Speed M P H
FORWARD SPEED M P H
FT /SEC
TIP SPEED RATIO μ
V_{LIM} (COCKPIT)
$V_{LIM}(M)$
$V_{LIM}(\mu)$
S L
5000
10000
15000
R P M RANGE FOR HIGH SPEED COCKPIT LIMITS
NORMAL R P M RANGE
R P M RANGE FOR LANDING
30 M P H
SPEED LIMITS INDICATED ON ALTIMETER
HIGH SPEED LIMITS 30 M P H ABOVE INDICATED LIMITS
TIP SPEED — FT/SEC
ROTOR R P M

FIG 2

The above considerations relate to blade lift There is in addition the problem of maintaining constant the force acting on the blade in the plane of rotation This force is made up by the profile drag of the blade and a small component of the lift vector due to the airflow through the rotor

The profile drag of the airfoils of the type used in rotor blades can be expressed by a single curve giving $E_p \, C_{L\,max} / C_{D\,min}$ as a function of $C_L / C_{L\,max}$ where E_p = (profile drag)/(lift) and which is shown in Figure 3 Thus as C_L varies with ψ as the expression

$$C_L = C_{L\,basic} \left\{1 + \tfrac{4}{3}\mu \sin\psi\right\}^{-2}$$

there will be accordingly a variation of profile drag during revolution In the upper part of Figure 3 are given as a function of μ the maxima and minima of F From this can be obtained with the aid of the straight lines $C_{L\,basic} / C_{L\,max}$ the range $C_L / C_{L\,max}$ covered during revolution which in turn indicates the variation of E_p

This figure clearly shows the following significant features

1 F_{max} increases very rapidly with μ whilst F_{min} decreases only moderately

2 A very low " basic " lift coefficient will produce excessive drag at the advancing blade (F_{min}) and a high $C_{L\,basic}$ excessive drag at the retreating blade (F_{max}) In view of the statement under (1) the latter is more likely to arise and therefore it is generally safer to fly at a low $C_{L\,basic}$

3 A low $C_{D\,min} / C_{L\,max}$ is beneficial all round

The component of lift in the plane of rotation D_L is dependent (Ref 1) on the inflow angle v_n / v_{eff}, where v_n is the component of velocity normal to the plane instantaneously containing the rotor blade, and v_{eff} is the component, in the plane instantaneously containing the blade, and perpendicular to the longitudinal axis of the blade

$$\begin{aligned} D_L &= (\text{Blade lift}) \quad v_n / v_{eff} \\ &= L \left\{\frac{\tfrac{4}{3}(\lambda + \iota) + (\tfrac{4}{3}\mu\beta_0 + \zeta\iota)\cos\psi}{1 + \tfrac{4}{3}\mu\sin\psi}\right\} \end{aligned}$$

where μ, λ and ι are the ratios between blade tip speed and forward speed, axial speed and induced speed at the rotor respectively

ζ is a factor related to the curvature of the induced flow at the rotor

β_0 is the coning angle of the rotor

L is the blade lift

and τ has been taken to be $\tfrac{3}{4}$

The above expression can be written as a series as follows

$$\begin{aligned} D_L = L \Big\{ & \tfrac{4}{3}(\lambda + \iota)(1 + 8\mu^2/9) + (4\mu\beta_0/3 + \zeta\iota)(1 + 8\mu^2/9)\cos\psi \\ & - \tfrac{16}{9}(\lambda + \iota)\mu\sin\psi + \text{higher terms in } \psi \Big\} \end{aligned}$$

In this only the first term is constant, the others change with ψ and thus represent varying forces It is therefore desirable that the coefficients preceeding cos ψ and sin ψ be kept as small as possible μ, λ and ι can be reduced by increasing tip speed λ moreover can be reduced by reducing the parasitic drag of the whole aircraft, Apart from such minor variations however λ in the pure helicopter is essentially an expression of flight condition of the aircraft, being necessarily larger in a climb than in a level flight In an auto-rotating rotor λ is always negative ζ cannot be controlled materially by design

In order to illustrate the above arguments the drag of a rotor blade of the Bristol Type 171 is now analysed on the basis of the above formulae

Conditions of flight

Steady level flight at sea level

Forward speed = 136 m p h = 200 ft per second

Rotor speed = 260 r p m = 670 ft per second

$\mu = 0\ 30$ $\lambda = 0\ 20$ $\iota = 0\ 0043$ $\beta_0 = 0\ 065$ $\zeta = 0\ 8$

$C_{L\,basic} / C_{L\,max} = 0\ 37/1\ 30 = 0\ 285$

$C_{L\,max} / C_{D\,min} = 180$

In Figure 4 the various components of blade drag are plotted as a function of ψ and it is shown how by a suitable choice of design parameters the drag variation due to inflow conditions ($\lambda + \iota$) can be counteracted by profile drag (E_p) Where the latter is a maximum the former reaches a minimum and vice versa This has been achieved by the use of a low "basic" lift coefficient giving an E_p – curve variation in the form of a positive

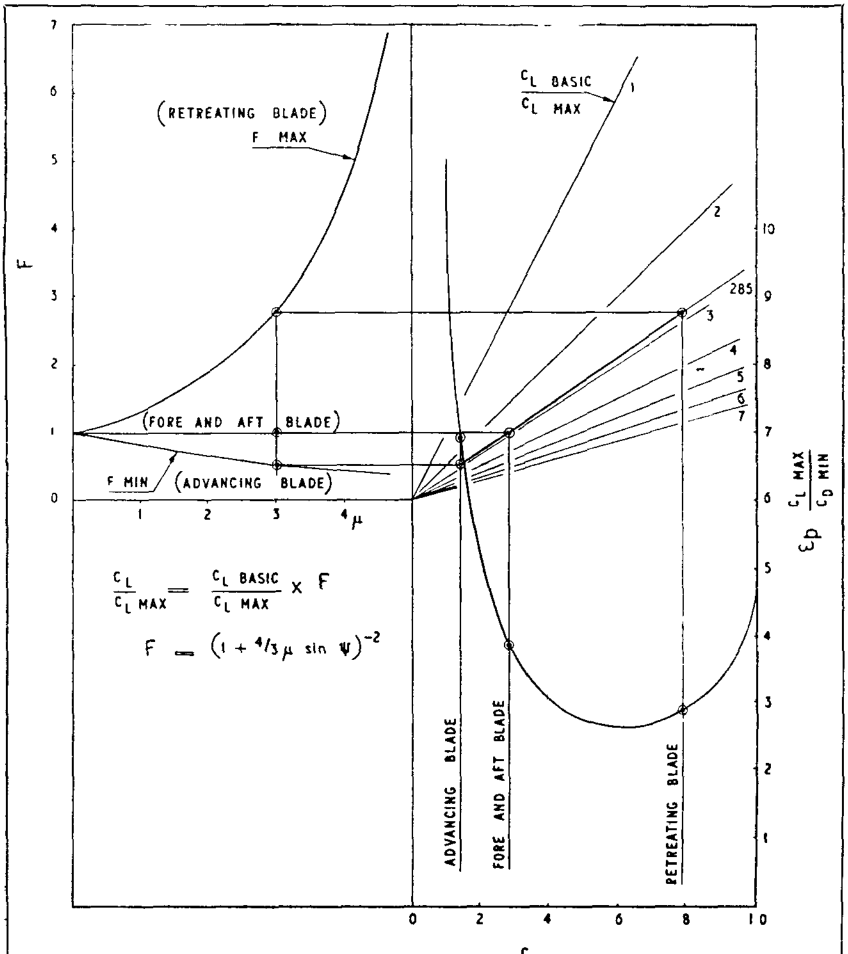


FIG 3 $\frac{C_L}{C_{L\,MAX}}$

https://doi.org/10.1017/S2753447200000512 Published online by Cambridge University Press

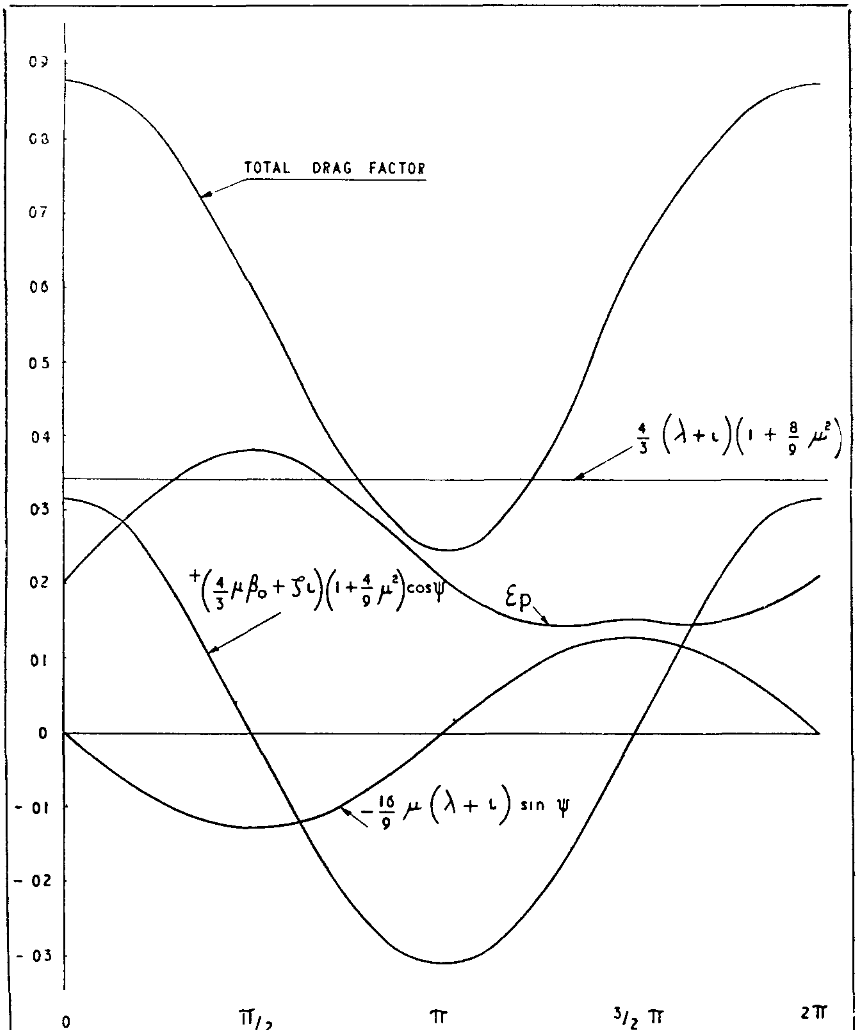


FIG 4

sine curve On the other hand had the " basic " lift coefficient been high then the E_p – curve would have taken the shape of a negative sine curve thereby amplifying the variation caused by $(\lambda + \iota)$ A low " basic " lift coefficient moreover represents a safety margin against stalling in accelerated flight conditions

It will be seen therefore that variation of blade drag is mainly caused by blade coning This variation even with the exceptionally small coning angle of the Bristol 171 rotor represents more than 50% of the total mean drag which illustrates the importance of this design parameter Under cruising conditions this drag variation is much more serious than the variation in blade lift referred to earlier

The drag of all blades add up vectorially to the total rotor force in the plane of rotation Thus the more blades in a rotor the steadier will be this force so that the 3-bladed rotor from the vibration point of view is very much better than the 2-bladed rotor which, as already mentioned earlier (Ref 1) is only justified in the very small sizes I have indicated too my preference for individual blade articulations over a flexibly mounted rotor (except for small sizes) as well as certain features in blade design such as careful distribution of blade mass radially as well as chordwise and a high torsional stiffness The ideal blade tapers from root to tip in thickness and plan form, and is of metal monocoque construction

Controls

There is clearly a need for simplification of controls More simplification in engineering generally means more efficiency and more safety This argument, however, does not apply to the *number* of flying controls in the cockpit which is determined only by the degree of freedom of movement of the aircraft (Ref 1) I am of the opinion that the conventional arrangement consisting of yawing control, rolling and pitching (or azimuth) control and vertical (or collective pitch) control together with the rotor speed control represents an absolute minimum There is, however, ample opportunity for simplification in the mechanism of the control circuits and this has been made a fundamental feature in the design of the Bristol Helicopter It is noteworthy that the rotor control mechanism of this aircraft (commencing at the 3 blade levers and terminating at the dual azimuth and collective pitch control levers) consists of only 27 moving parts The use of the tie rod in this helicopter is now well known and needs no further comment

The design of cockpit controls is at present handicapped by the lack of agreement on control layout for helicopters Standardization of cockpit controls is an urgent need

Stability

I propose to make to-day only a few general observations on this subject The stability of present day helicopters is far from satisfactory and " the " solution of the problem which must be simple as well as adequate does not appear to be around the corner yet

There are two distinct types of stability, distinct from the theoretical as well as the pilot's point of view

Stability in forward flight

In this respect the helicopter does not differ materially from the fixed wing aircraft Most of the flying time is accumulated in forward flight and therefore positive stability in this condition will prevent pilot's fatigue and therefore increase safety

Hovering Stability

The hovering stability is based on different principles and is more difficult to obtain than stability in translational flight As hovering flight is carried out in circumstances when concentration on the part of the pilot is needed in any case because of the proximity of the ground or the need to remain in a given attitude with relation to a fixed point, and the time spent in this condition is comparatively short, there would not seem to be the same need for positive hovering stability as for positive forward flight stability If the latter is fully achieved whilst in hovering the aircraft is not unduly unstable, in my opinion a good advance is made on the way towards the ideal

The Bristol Type 171 shows in forward flight a fair measure of positive stability in pitch, yaw and roll and has been flown in gusty air for extended periods without touching of any controls Minor displacements are damped out and only severe gusts require corrective action on the part of the pilot This has been achieved mainly by a fuselage carrying a small tailplane outside the rotor disc and a rotor with a large moment of inertia providing damping in pitching motion

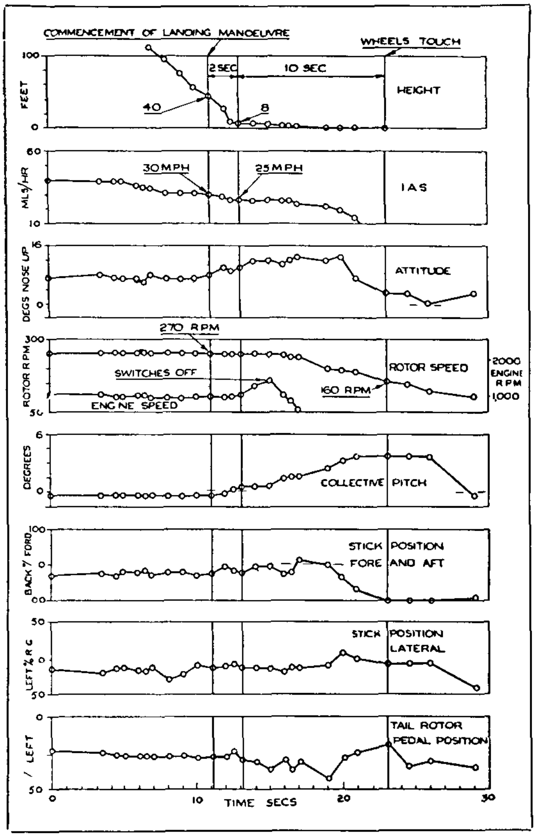
Safety

Apart from structural and mechanical safety an important safety criterion is the emergency landing in the event of power failure Whilst the rotative wing is reputed to offer a high measure of safety in auto-rotative descents, recent reports have drawn attention to so called " danger zones " from which an auto-rotative landing could not be made without damage to the helicopter The vertical velocity of a conventional helicopter during auto-rotative descent, especially at low forward speeds, is too great to be absorbed by the undercarriage and must therefore be checked prior to landing by momentary increase of rotor lift which involves a certain amount of energy In the absence of engine power such energy is available in the form of kinetic energy from the horizontal velocity component of the aircraft as well as from the rotational velocity (useful r p m range) of the rotor When these two sources of kinetic energy are insufficient to provide the required check then damage will result on landing In order to eliminate this hazard, the rotor of the Bristol Helicopter has deliberately been designed to give a high rotational moment of inertia in conjunction with an unusually wide r p m range The kinetic energy which is stored in this rotor at maximum permissible rotor speed is 680,000 lb ft which represents an amount of work equivalent to about four seconds hovering in the ground cushion at full load Many landings have already been made with this aircraft with its engine switched off, which have indicated an ample supply of kinetic energy during the landing manoeuvre

Auto-observer records of such a landing are reproduced in Fig 5 The steady conditions during the auto-rotative descent prior to the landing manoeuvre appear to be as follows —

Rate of descent	1,600 ft /min	Engine Speed	Idling
I A S	35 m p h	Collective Pitch	0 degrees
Rotor Speed	270 r p m	All-up Weight	4,740 lb

At about forty feet from the ground the collective pitch is increased slightly and the engine switched off This results after about two seconds in a checking of the descent at a height of approximately eight feet, and a reduction of forward speed by some five miles per hour During the following ten seconds the aircraft sinks slowly until, at first the tail skid, and then the wheels touch down During this period the collective pitch is steadily increased to 11°, the rotor speed drops to 160 r p m and the forward speed down to walking pace Apart from an aft movement of the control column



at touch down, all controls remain substantially stationary This serves to show that with sufficient kinetic energy and good control, an engine-off landing can be made in safety and comfort, even with a heavy helicopter

FIG 5

Auto-observer records of typical auto-rotating landing of Bristol Helicopter Type 117

CAPITAL AND MAINTENANCE COSTS

One of the main arguments against the helicopter is that it is expensive to buy and an excessive amount of time and money is spent on its maintenance

The helicopter of to-day is expensive to buy not for reasons peculiar to this type of aircraft but simply because the market to-day does not justify production on a large scale, without which a material reduction of price cannot be achieved

Maintenance costs can be reduced if the following design principles are adhered to —

1 Mechanical simplicity and economy in the number of moving parts
2 Fatigue resistance and high life factors in parts subject to cyclic forces and to wear
3 Accessibility
4 Interchangeability

Figure 6 shows the entire mechanical assembly of the Bristol 171 helicopter which comprises the power unit with clutch, cooling fan, and cowlings, the main rotor gearbox and hub carrying the main rotor blades, the tail rotor gearbox and hub carrying the tail rotor blades and transmission shafts between these units and in addition the rotor controls The extreme simplicity of this layout is evident

https://doi.org/10.1017/S2753447200000512 Published online by Cambridge University Press

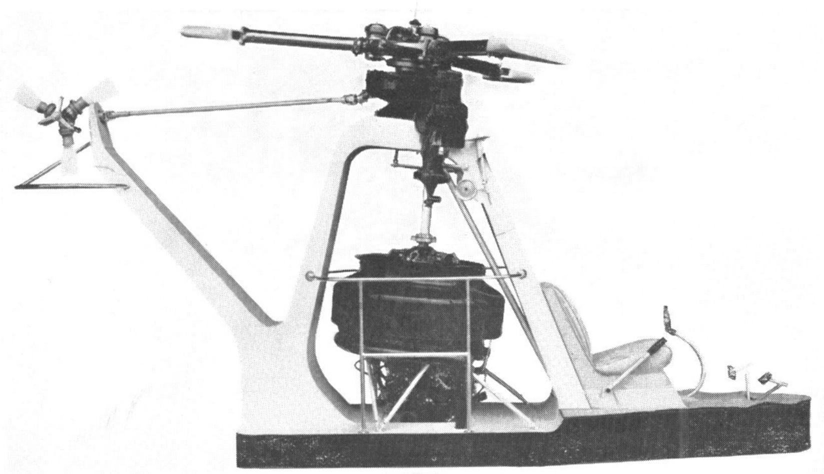


FIG 6

It has been possible to design all wearing parts for a life of 7,500 hours without great sacrifice in weight This feature apart from beneficially affecting the safety of the aircraft will it is hoped eventually permit the periods between overhauls to be increased to 1,000 hours or more, (excepting the power unit) which not only simplifies maintenance but ensures that the adjustment of wearing parts is left undisturbed over long periods which allows these parts to run-in under ideal conditions

Fatigue is one of the topics in aircraft engineering There is no royal road to endurance and a long and generally arduous test and development programme with the proverbial " exploring of every avenue and leaving no stone unturned " which involves a great amount of test equipment and cost is the only real safeguard against fatigue failures Special testing equipment which has been developed in conjunction with the Type 171 has been discussed in an earlier paper (Ref 1)

The principle of interchangeability is, of course, applied generally We hoped originally to build individually interchangeable rotor blades which

FIG 7 The Bristol 171 Helicopter in flight

were to be balanced dynamically as well as aerodynamically against a master blade, but experience to date has indicated that blades can be assembled satisfactorily only in rotor sets This does not, however, in the event of damage to one blade in a rotor, imply the scrapping of the remaining good blades which can be " paired " with others to make up new rotors

---

My acknowledgements are due to the Ministry of Supply and The Bristol Aeroplane Co , for permitting the publication of information relating to the Bristol Helicopters Types 171

References

1 Hafner, R Helicopter Rotor Systems and Control Problems *Proc Anglo-American Aeronautical Conference*, 1947 R Ae S

2 Hafner, R The Hafner Gyroplane *J R Ae S*, XLII, 109, 1938

Symbols

V	=	Forward velocity of helicopter
ω	=	Rotor angular velocity
r	=	Radial distance of blade section, measured from hub
R	=	Rotor tip radius
μ	=	Tip speed ratio $= V/\omega R$
ψ	=	Blade azimuth angle measured in direction of rotation from downwind position of blade
β_0	=	Rotor coning angle
ρ	=	Air density
$C_{L\ max}$	=	Maximum lift coefficient of blade before stalling
$C_{D\ min}$	=	Minimum profile drag coefficient of blade

THE CHAIRMAN

The last paper this afternoon is to be read by Mr J S Shapiro, and is a description of the Cierva helicopter Type W 11, popularly known as the " Air-Horse " Mr Shapiro is a graduate in Mechanical Engineering of the Swiss Polytechnic and an A F R Ae S and had varied experience in the design of ancillary aircraft equipment in France before coming to this country in 1940 He was then for a time with Power Jets Ltd , after which he had further experience on design of aircraft instruments and power assisted aircraft controls In 1943 he joined the Cierva Autogiro Company and is now Senior Technical Officer to that firm

Unlike the other two aircraft you will have heard described to-day, the Cierva W 11 has not yet flown, but has done a good deal of preliminary ground running and it is hoped that it will take the air in the very near future I will now ask Mr Shapiro to read his paper