# SOME DESIGN ASPECTS OF TANDEM ROTOR HELICOPTERS

Part I
By J D SIBLEY, B SC, A F R AE S

Part II
By C H JONES, B SC (ENG), A F R AE S, A M I MECH E

*A paper presented to The Helicopter Association of Great Britain in the library of the Royal Aeronautical Society, 4 Hamilton Place, London, W 1, at 6 p m on Friday, 5th June, 1959*

Professor J A J BENNETT (*Chairman of the Lecture Committee*) *occupying the Chair*

C H JONES

J D SIBLEY

The CHAIRMAN, in opening the meeting, said that the first part of the Paper would be presented by Mr J D Sibley, who for the past 10 years had been Chief Aerodynamicist at the Helicopter Division, Bristol Aircraft Ltd, Weston-Super-Mare Mr Sibley would discuss the aerodynamic aspects The second part, relating to the dynamical aspects, would be presented by Mr C H Jones, who had been Dynamics Engineer at the Helicopter Division, Bristol Aircraft Ltd, since 1955 Afterwards there would be a short film

## PART I By J D Sibley

The paper discusses a number of aerodynamic aspects which have appeared during the development of the Bristol Helicopters I have deliberately avoided excessive comparisons of the advantages and disadvantages of tandem helicopters versus the single main rotor helicopters, as this subject has been covered by numerous authors

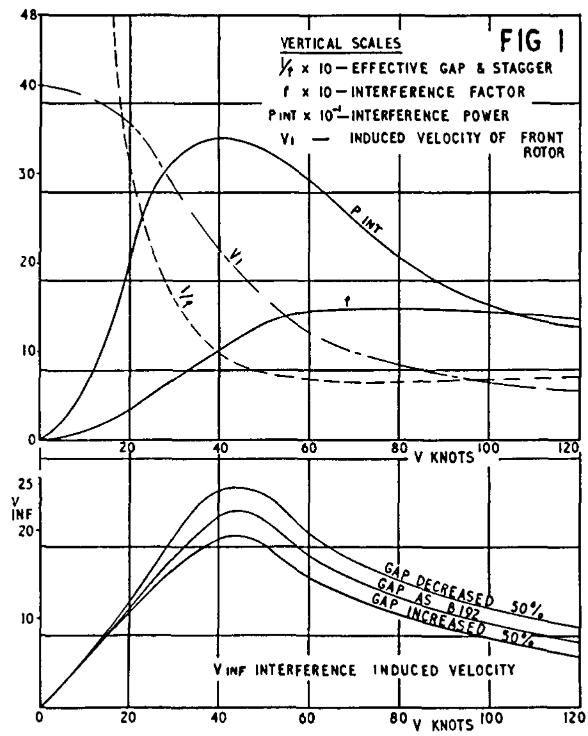
The subject matter covers information on performance, control and stability, which has been obtained in the course of development work on Bristol Tandem Helicopters Some of this information is directly applicable to other types of helicopter

The paper also describes a proposed development of the tandem helicopter using a wing to off load the rotor at high speeds , producing a transport helicopter capable of cruising at 200 knots

### Performance

The significant difference in performance estimates between the tandem helicopter and the "penny-farthing" is the mutual interference between rotors, causing an increase in momentum power This is compensated by the tandem needing no power for torque reaction

In forward flight the front rotor induces a downwash on the rear rotor so that, in order to produce a given amount of lift, the rear rotor has to do



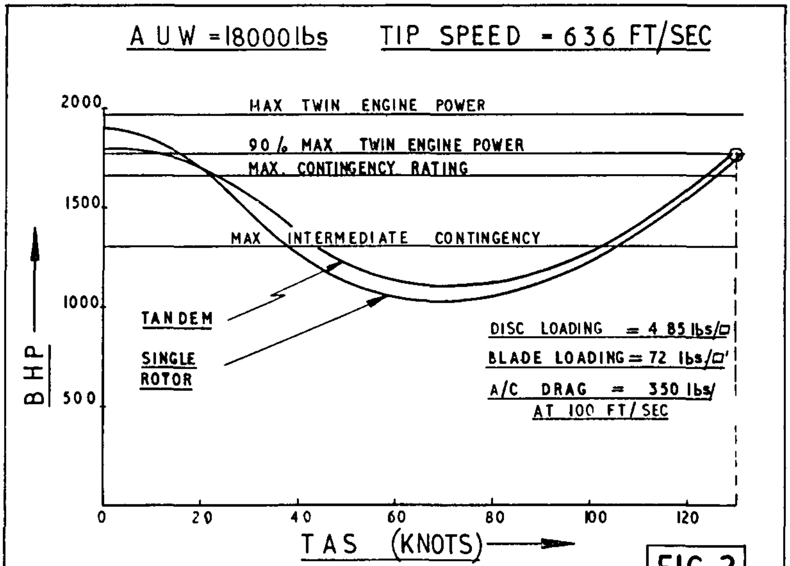



FIG 2

work on this interference velocity equal to the product of Thrust and Velocity The interference is a function of the front rotor induced velocity, and an inverse function of effective gap between the rotors

In hovering, assuming no overlap, the interference is zero, the gap being effectively infinite As speed increases the gap reduces and the interference rises to a maximum, then falls again as the induced velocity begins to fall rapidly The effect of changing the effective geometrical gap between rotors is shown in the Fig 1 There is a reduction in power at the front rotor, due to an upwash induced by the rear rotor, but this is small and at present neglected in calculations

We have found, for performance calculations on Bristol tandems, that an interference induced velocity at the rear rotor of 1 5 times the induced velocity of the front rotor gives good agreement with measured performance results, from minimum power speed upwards This quantity agrees with the theoretical predictions of Ref 1

The mutual interference between the tandem rotors gives rise to a different shaped power required against speed curve than for the " penny-farthing " For a given disc and blade loading, the hovering power is less for the tandem, due to the need of no power requirement for torque compensation, but the fall off of power with speed is less, due to the interference effect Fig 2 The minimum power is a little higher for the tandem helicopter due to the interference of the rotors, whilst the torque compensating power of the single main rotor type is at a minimum at minimum power

In forward flight, due to the interference being small, the two helicopters are similar In Fig 2 it is assumed that both helicopters have the same value of parasite drag, so that the tandem is shown to be at a slight dis-

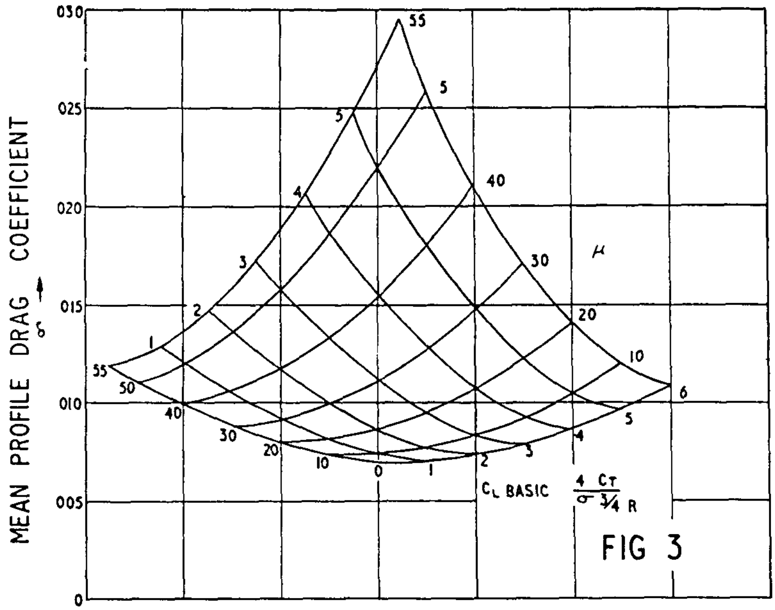


FIG 3

advantage In practice I think that the single main rotor type of helicopter would probably have a slightly higher drag value, due to the larger cross sectional area of the fuselage, necessary to obtain a given cubic capacity, unless this type of helicopter can be designed with C G ranges equal to a tandem helicopter

The basic shape of the power required versus speed curve shows it to be most suitable for a twin engine layout where positive engine out performance is required and turbine engines to the latest agreed helicopter ratings are used Max twin engine power (max continuous rating) gives vert R/C of 600 ft /min Cruising (say 90% max continuous rating) gives a useful cruising speed (130 knots) The max intermediate contingency rating, which is a single engine rating for enroute use up to periods of 1 hour, gives a rate of climb of 150 ft /min (2 5% gradient) and the max contingency (2½ min duration) gives a good single engine ground cushion hover at Max A U W (approximately 600 ft /min vertical rate of descent in free air) which should be ample power in the case of engine failure during T O or landing

The helicopter power required curve and required single engine performance fits the ratio of max contingency rating/intermediate contingency rating of 1 25 which was generally accepted by the engine manufacturers Before leaving the subject of performance I must discuss the subject of the profile drag coefficient of the rotor blades, particularly in forward flight

In most performance methods the rotor profile power for hovering is usually calculated from the drag curve for the section, and then the profile power in forward flight obtained by increasing this value by a function of $(1 + K\mu^2)$ where K is about 4 5 (Ref 2)

We have calculated several cases in forward flight using strip theory for a number of blade azimuth positions, using the best available wind tunnel data and find that the mean profile drag coefficient for the rotor in forward flight is higher than that estimated for hovering This has been confirmed by analysing a number of flight tests made with parallel chord 12% t/c ratio blades Fig 3 shows the mean drag coefficient $\delta$ v $C_L$ curve derived from tower and flight tests

The rotor tower curve ($\mu = 0$) was derived by subtracting an assumed induced power from the total power, using an induced velocity equal to 1 2 times the ideal value The flight test data was similarly derived by assuming an induced velocity factor of 1 045 times the ideal velocity, and a parasite drag of the aircraft derived from the mean of about thirty partial climbs, using the method of Ref 3 The factor K in the expression $(1 + K\mu^2)$ for the profile power term was 4 75

## Stability and Handling

*Longitudinal*

The predominant parameter in the longitudinal stability of the tandem rotor helicopter is the induced downwash of the front rotor on the rear rotor, as illustrated in the performance section, Fig 1 The longitudinal control is obtained by simultaneously tilting both rotors by applying cyclic pitch control, and at the same time applying differential collective pitch in order to produce a satisfactory ratio of linear acceleration to pitching acceleration for applied control We have found a satisfactory ratio of linear to pitching acceleration to be 3 0 Due to the large pitching M of I of the helicopter, insufficient pitching control is obtained by purely tilting the rotor, and pure differential collective pitch would produce a helicopter with sluggish response for accurate hovering As the helicopter moves forward from the hovering condition the effect of increasing the downwash on the rear rotor is to increase the inflow angle, reducing the angle of attack of the blades and consequently reducing the rotor thrust

This produces a tail down moment, and the cyclic stick has to be moved forward to correct this moment from the downwash The basic curve for the tandem helicopter of stick position to trim against speed, therefore, follows closely the interference velocity curve (Fig 1), giving rise to a positive slope for low speed and a negative slope for the cruise regime This deficiency in the cruise regime may be overcome either by the use of longitudinal dihedral between the rotors, or a negatively lifting tail plane

The longitudinal dihedral means that the rear rotor gets an increasing inflow with speed, so that the cyclic control must be moved forward with speed to obtain trim, thus giving a neutral or positive stick position to trim with speed A similar effect can be obtained with a tailplane at a negative angle of attack, so that there is an increasing download with speed, which again is balanced with a forward movement of the cyclic control This latter method is not so good as the former, due to the extra power that is required to produce the download on the tailplane, plus the rear rotor power needed for the extra thrust on the tail rotor

Dynamic longitudinal stability has not caused any predominant difficul-

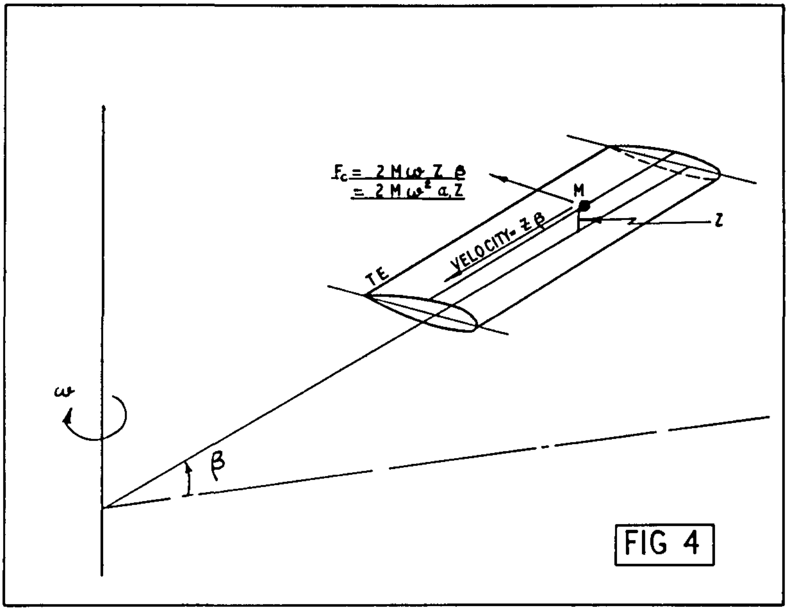


FIG 4

ties on Bristol Tandem Helicopters, but the parameters readily available to the designer to deal with such problems are as follows

(*a*) Longitudinal Dihedral between rotors, as described above, which gives speed stability derivatives

(*b*) Differential Delta three ($\delta_3$) flapping hinges between front and rear rotors, giving derivatives with angle of attack due to the resulting differential lift curve slopes on the two rotors

(*c*) Fixed lifting surfaces, which can give both speed and angle of attack derivatives in the cruising regime

*Longitudinal Control Force to Trim*

The Bristol Tandem Helicopters have manual controls, and the control loads are kept to a minimum by " tuning " the rotor The propeller moment and the tie-bar torque on the collective system are balanced by means of an anti-spring system, as also is the tie-bar on the cyclic system The remaining forces are due to inertia and aerodynamic moments

The rotor, as it flaps back relative to the control orbit to maintain its equilibrium of forces produces, due to its M of I about major chord axis, a control force giving a negative stick force gradient The principle is shown in Fig 4 A point mass on the blade due to flapping velocity has inward velocity $Z\beta$ giving aft coriolis acceleration $2\ Z\beta\omega = 2\ Za_1\omega^2$, and a T E down moment $= 2\ MZ^2a_1\omega^2$ This produces a negative stick force to trim proportional to $a_1$, which to a first order is proportional to the tip speed ratio $\mu$

$$\beta = \frac{d\beta}{dt}$$

$a_1$ = longitudinal flapping coefficient

$\omega$ = small omega (ang velocity)



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This negative force gradient is balanced by means of root tabs on the blades, which predominantly produce a change of pitching moment rather than a twist change on the blade, due to the relatively high stiffness at the root  A collective tab change on a rotor (*i e*, all tabs moved an equal amount) produces a change in cyclic stick force proportional to $\mu$ and a collective force change of the form $(1 + K\mu^2)$  Thus within reason, any cyclic force to trim curve may be produced, using a combination of equal tabbing on both rotors, which applies load through the pure cyclic system, or differential applied tabbing to the front and rear rotors, which applies a change of cyclic force through the collective coupling control

A typical curve of pilot's control to trim against speed for Type 192 is shown in Fig 5  This force is trimmed out by the pilot with electrically operated trimmers operating through very low rate springs  The use of this type of trimmer increases slightly the loads which the pilot has to apply, since he has to work against the spring as well as the loads from the rotors

To overcome this difficulty a new type of load trimmer has been devised, which utilises the spring in the system for balancing the load from the rotor tie-bars  By changing the datum of this spring it is possible to balance the rotor loads to trim, and has the advantage of not introducing extra loads in the controls when the pilot moves his control from the trimmed position

*Lateral Stability*

The original Type 173 with the dihedral tailplane exhibited lateral instability in the cruise regime  Tests using the step input method showed the instability to be predominantly a divergent oscillation in roll and sideslip

To quickly assess the order of the parameters necessary to cure this instability, a series of high drag drogues were made for towing behind the

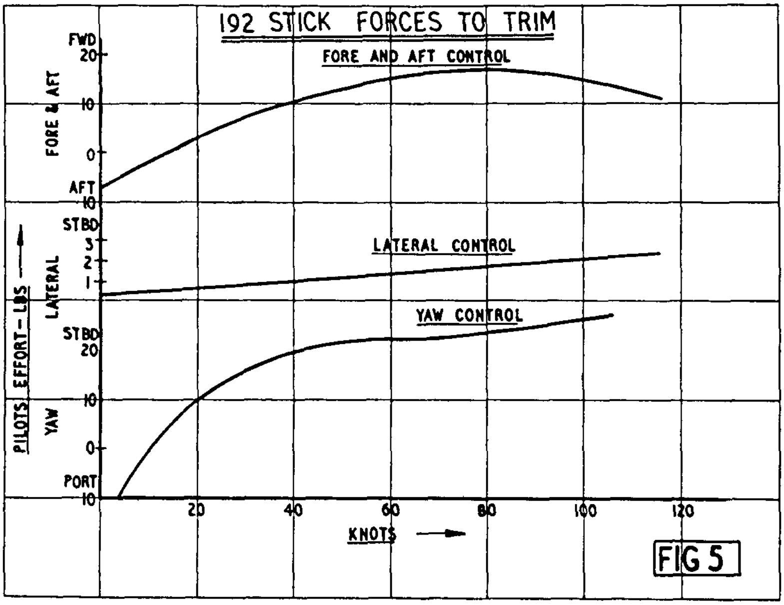


FIG 5

helicopter (Fig 6) Basically the drogue introduces side force and yawing moment with side slip $Y_V$ and $N_V$ respectively, and by vertical positioning of the attachment on the fuselage, variations in Rolling Moment with side slip $L_V$

For the tests the drogue was attached below the principal axis of inertia of the fuselage

Dynamic stability tests showed the helicopter to be more stable, and static stability tests showed that this was achieved by increasing the yawing moment with sideslip ($N_V$) (Rudder position to trim) and by reducing rolling moment with sideslip ($L_V$) (lateral control to trim) A horizontal tailplane

FIG 6

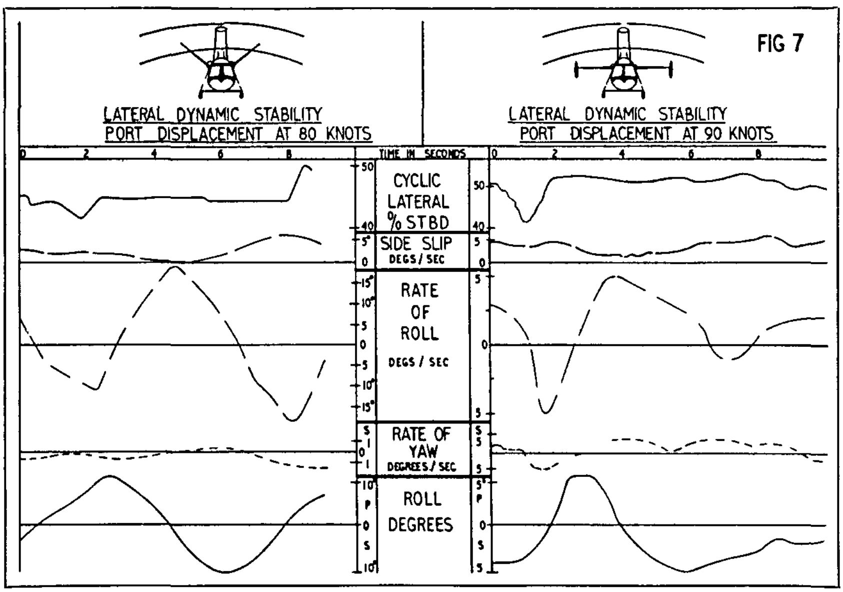


FIG 7



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with tip fins was designed to give increased yawing moment with sideslip, and, at the same time to reduce the rolling moment with sideslip

The latter derivative was achieved by positioning the tip fins as low as possible to counteract the rolling moment from the main pylon fin Dynamic stability tests showed that in cruising flight the helicopter was stable Fig 7 gives a comparison of the aircraft motion resulting from a lateral control step input for the two types of tailplane Fig 8 shows the order of the calculated divergent oscillation with the two tailplanes , note that theory is not quite as good as the actual helicopter The effect of introducing anhedral to the " H " type tailplane is also shown

Wind tunnel tests have shown that an important parameter is the positioning of the tailplane longitudinally on the fuselage If the tailplane is positioned several chord lengths forward of the tail end of the fuselage, there is an effective cross flow round the fuselage with sideslip, giving the tailplane an effective anhedral and hence reducing the rolling moment derivative This effect is not present when the empennage is positioned at the extreme tail of the fuselage Fig 9

## Future Development

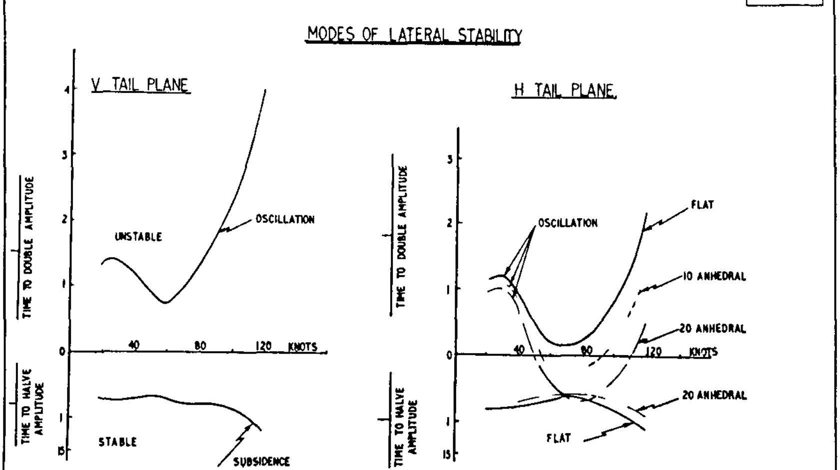
With the advent of various proposals for STOL and VTOL aircraft it is necessary to take stock of the present position of the helicopter and its future

Current helicopters and their derivatives, with cruising speeds of 100 knots plus, should be able to hold their own for stage lengths up to 100 miles, but to compete in the future for the longer hauls, say 100—400 miles, the present cruising speeds must be substantially raised

The four principal parameters to consider in a high speed helicopter, are

1 Compressibility effects on the rotor, giving rise to both high rotor profile power and large changes of blade pitching moment
2 Retreating blade stalling

FIG 8



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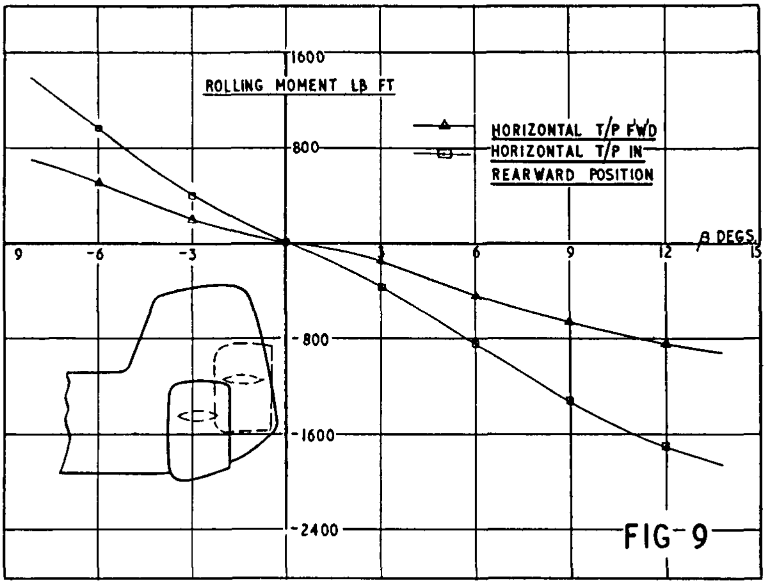

FIG 9

3 Reduction in specific power required to cruise, and in particular reduction of fuselage parasitic power
4 Reduction in Vibration

*Compressibility*

Recent reports from America (Ref 45 and 46) give the results of full scale tower tests on 9, 12 and 15% thickness/chord ratio blades The reports show that the 12% t/c ratio blade at low $C_L$'s has a critical Mach No for drag divergence (based on initial drag rise) of the order 8, which is about 1 higher than that given by two dimensional wind tunnel tests Further, the critical Mach No only rises about 008 per 1% decrease in t/c ratio, which is of the order of a half of the rate generally given from wind tunnel data

Other data indicate that a 12% t/c ratio blade can achieve critical Mach Nos for pitching moment divergence up to 85 for only a small rise in drag coefficient Thus, at the expense of small blade profile power rise, it should be possible to design a high speed helicopter using conventional blade design to operate without large cyclic changes of pitching moments producing high control forces

Small gains can be obtained by using thinner blade sections at the rotor tip, although it would be necessary in the design to take into account the effect of the associated reduction in $C_{L\,Max}$ on the retreating blade

To date, Bristol rotor blades with parallel chord and 12% t/c ratio have achieved a tip Mach No in forward flight of 77, on a helicopter with a manual control system, without compressibility phenomena becoming apparent It is intended in the near future to make tests to extend our knowledge on this subject

*Retreating Blade Stall*

For some time it has been generally accepted that the criterion for a helicopter's limiting forward speed is a max angle of attack at the tip of the retreating blade (Ref 7) This method shows that an increase in inflow through the disc, due to an increase in fuselage drag, reduces the allowable forward speed for a given limiting angle of attack at the blade tip A recent full scale wind tunnel test (Ref 8) examined the problem of rotor stalling for a range of tip speed ratio $\mu = 3$ to $4$ for 12% t/c ratio blades

Fig 10 shows the result plotted as limiting $C_{L\ (basic)}$ for retreating blade stall against tip speed ratio It is of interest to note that the tests were made over a fairly wide range of inflow angle and showed blade stalling to be independent of this variable Also shown in the graph is max tip speed ratio which has been flown on the Bristol 171 with 12% t/c ratio blades In this case the rotor was not stalled, but mild feed-back into the controls was felt in manoeuvres The points for the other helicopters were taken from published data (Ref 9), with the exception of the Rotodyne which I have "guestimated" based on her record run with an assumed range of wing lift

The Bristol 171 case corresponds to a tip angle of attack of 11 6° Using this as a datum, a general curve of $C_{L\ basic}$ v $\mu$ has been calculated and falls fairly close to the test data

It has been suggested (Ref 9) that blade stall at high tip speed ratios is not important, and this is probably true since at $\mu = 5$ the velocity at the retreating blade tip is only half the tip speed and only half the blade is producing lift, so the effect of this region on the overall rotor lift must be small To maintain equilibrium the lift on the advancing blade must be equally small, so that the work can be done only by the fore and aft sections of the rotor disc

RETREATING BLADE STALLING CRITERIA

PARALLEL CHORD BLADES 12% T/C

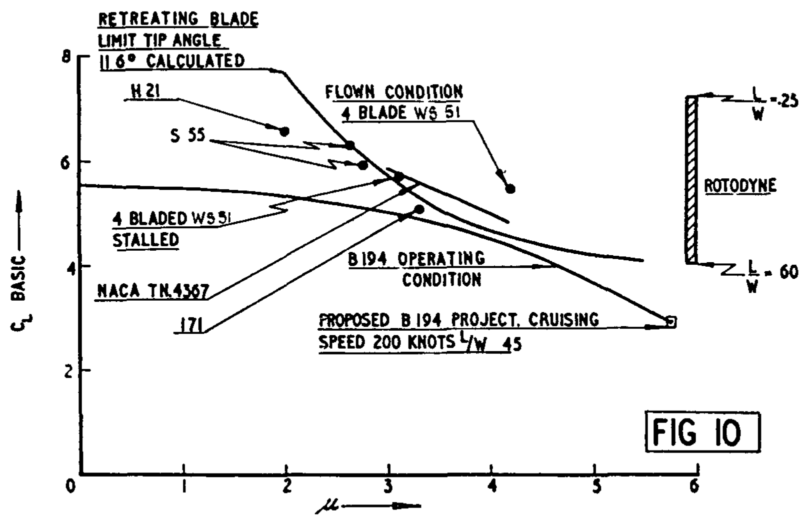


FIG 10

However, there are several factors to consider

1 If in the cruising regime the rotor is not stalled, then the vibration level will be less even if by a small amount
2 The helicopter has to fly through a high vibration region produced by blade stalling at the beginning and end of each flight
3 The multi-engine helicopter will be required to stand off at minimum power speed, which for the large high speed helicopters with high disc loadings will probably occur at, or slightly above, the condition for blade stall, which condition limits the speed of current helicopters, due to high vibration levels
4 The thrust at high tip speed ratios is mainly carried on the fore and aft sectors of the rotor disc, which means that these sectors are operating at nearly double the basic hovering $C_L$ Thus, these sectors are liable to stall under gust conditions unless the rotor is designed with a very low basic hovering $C_L$, which would introduce a high blade profile power penalty

These four points can be overcome by the compound helicopter, using a wing to reduce the rotor blade loading with increase of forward speed The effect of a wing carries 45% lift at a cruising speed of about 200 knots is shown in Fig 10, noting that the curve comes nearest, but not touching, the stalling criterion around a $\mu$ corresponding to the minimum power speed

In the case of the compound tandem helicopter, a single wing would be situated near to the mid point between rotors, to minimise download from rotor downwash in hovering

*Power*

*Reduction in Specific Cruise Power* In order to keep the transmission weight to a minimum it is desirable to limit the engines under multi-engine conditions to that power necessary for a satisfactory vertical take-off in the atmospheric design conditions The power required to cruise should be limited to say 90% of this condition, so that to achieve cruising speeds of the order of 200 knots considerable improvements over current helicopters must be made

*Induced Power* The total induced power of the rotors and wings of a compound helicopter, including the mutual interference between wings and rotors, is approximately the same as if all the lift were carried on the rotors, so that no appreciable saving can be made on this item

*Rotor Profile Power* The use of the wing to reduce the basic $C_L$ of the rotor in cruise gives a large reduction in rotor profile drag coefficient, as can be seen from Fig 3 At 200 knots, a reduction of specific power required of 035 HP/lb A U W (25%) from the power required for a pure tandem can be achieved by carrying 45% of the A U W on a wing

*Parasite Power* The high speed helicopter needs the same standard of aerodynamic cleanness as the modern fixed wing airliner, since the helicopter operating at low altitudes has the same order of E A S The helicopter must be designed with a good aerodynamic shape, plus flush rivetting and fully retracted landing gear The hub drag can be reduced by keeping the projected area small (more compact mechanical design) and by use of suitable fairing

*Vibration*

Vibrations translated to the fuselage are at frequencies of the product of number of blades per rotor times the rotor speed, and simple multiples of this frequency Rotor harmonic flapping coefficients decrease approx 10 times for each increase in order, thus, by using 6 blades per rotor, the

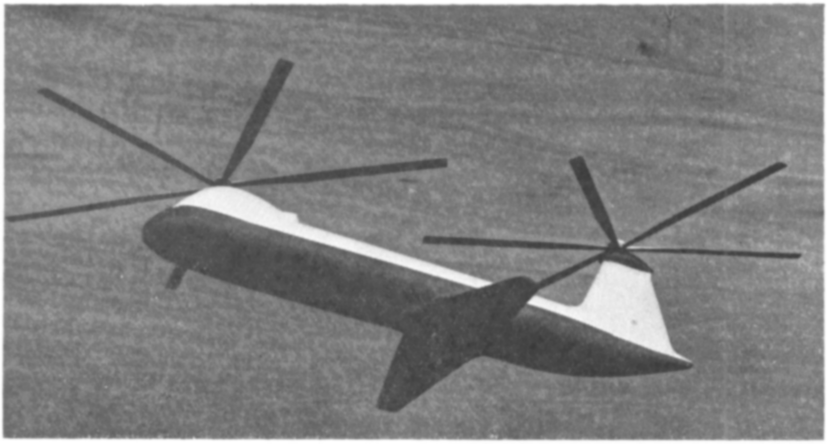

FIG 11

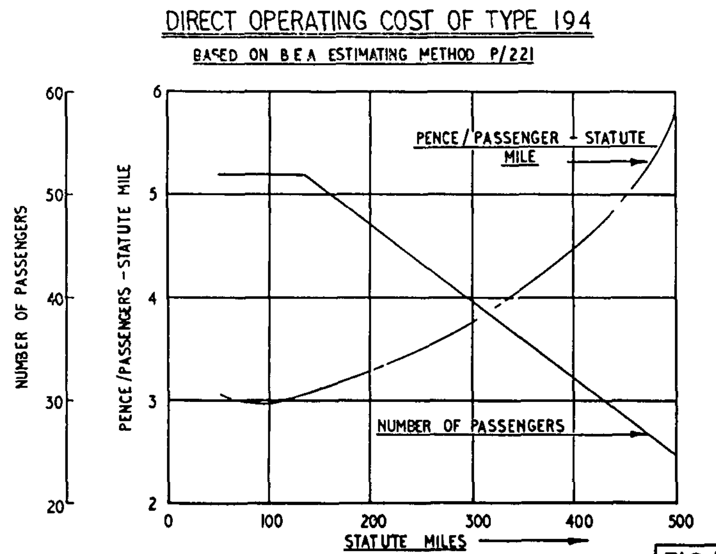


FIG 12



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vibration translated to the fuselage will be smaller than for say a 4-bladed rotor carrying the same thrust by virtue of

(*a*) Smaller lift component per blade
(*b*) Smaller flapping harmonic
(*c*) Less fuselage response to higher frequency
(*d*) In the case of the compound helicopter proposed, the rotors are not operating in a stalled state

In the case of the tandem with non-intermeshing rotors, these can be phased to give minimum response Fig 11 shows a model of the resulting helicopter which is designated Bristol B 194 , note the anhedral on the wing to reduce the rolling moment with sideslip from the main fin Fig 12 shows the order of direct operating costs

In conclusion I should like to thank Bristol Aircraft Ltd for permission to publish the material, and to point out that any opinions expressed are my own and not necessarily those of the Company

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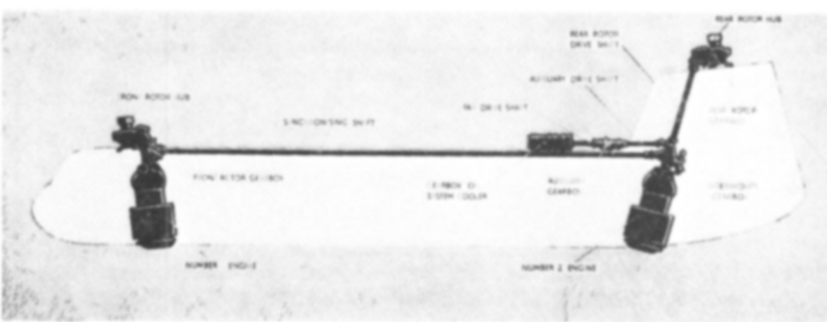

*Fig 13 Rotor transmission*

## PART II By C H Jones

In attempting to set down some of the dynamic problems which arise in relation to the tandem helicopter, I cannot help feeling that, after all, these problems are so like those of any other helicopter that the distinction in the title is barely warranted Nevertheless, that this is so may not be so obvious to everyone

Of the many aspects of design which could be selected, I propose to discuss only three topics, those of transmission and engine control, undercarriage and ground resonance and the effect of the flying control circuit on blade flutter and forced vibration

### TRANSMISSION

The elements of the transmission system of a typical twin engine tandem helicopter, the Bristol Type 192 are shown in Fig 13 In this case, two Napier Gazelle N Ga 2 free turbine engines drive into the front and intermediate gearboxes through freewheels The two rotor gearboxes are interconnected through the synchronising and rear rotor shafts The interconnection requires particular attention because it is obvious that no failure can be allowed and the standard of reliability must equal that of simple structural elements (The ways of achieving and proving this are of great interest but are not peculiar to the tandem helicopter and have no part in this paper )

The drive to the auxiliaries is also regarded as Class 1 in its entirety to safeguard electrical and gearbox cooling services This is in contrast to the engine reduction gear and free turbine, which are isolated by the freewheels in the event of mechanical failure

#### *Dynamic Properties*

Fig 14 shows the three lowest frequency normal modes of the Type 192 Transmission when two engines are driving and with the rear engine driving only Similar modes to the latter are obtained with the front engine only driving, whilst in autorotation only the fundamental mode is of interest , this mode is almost unchanged and the frequency remains at 89 c p s

The fundamental mode is of rather low frequency and can be excited by oscillating the pilot's control column longitudinally However, the mode has 12% critical damping by virtue of the lag hinge dampers giving 1 1 secs to half amplitude , it is this damping which prevents the pilot from producing excessive response in the mode

It also transpires that, since the synchronising shaft has very high stiffness compared with the lag hinge C F stiffness, the engines respond very little in the fundamental mode This mode can be excited only by differential rotor torques and so responds when differential collective pitch is applied through forward or aft stick movements The second mode is rarely distinguishable , it has 14% critical damping, taking 4 sec to half amplitude, and can be excited only by simultaneous operation of the engine throttles The third mode in which the two engines swing against each other is not damped deliberately , the natural damping is 7 6% critical,

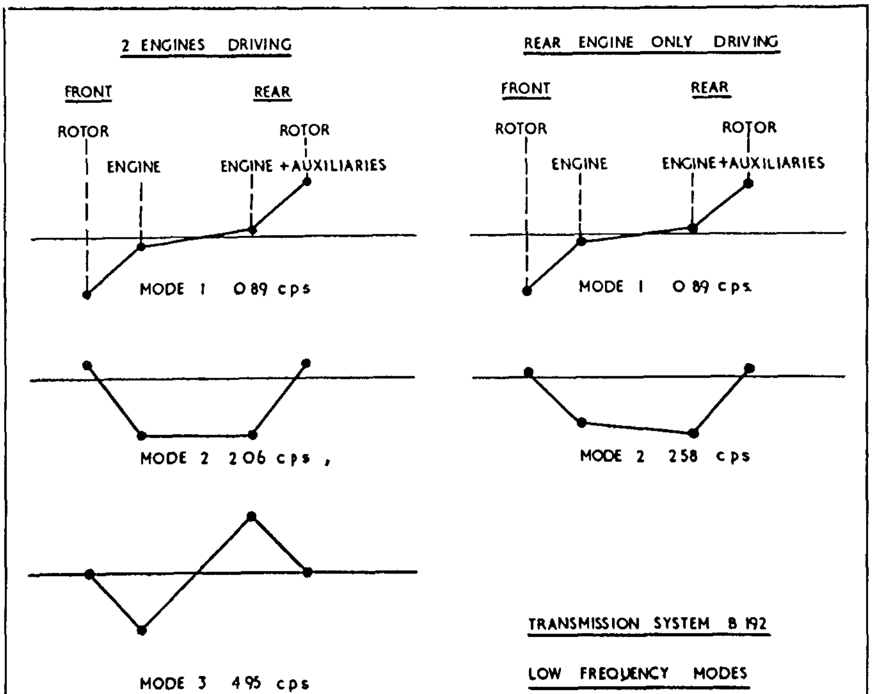


FIG 14

requiring 0 3 seconds to halve amplitude This mode is not excited by rotor order vibrations partly because the natural frequency is 1 4 times that of the rotor rotation and partly because the rotor blades are nearly nodal in this mode

*Design Cases for Maximum Torque and Maximum " G "*

The requirements of A P 970 and of B C A R Section G define a normal load factor of " g " to which the helicopter must be designed

This information is of limited help, since a normal acceleration cannot be applied to a helicopter very easily without also having a pitching acceleration Then too, there must be some torque reaction at each of the rotor gearboxes as well as rotor moments and in-plane forces The problem of specifying design cases is therefore how should the helicopter be manoeuvred in order to develop maximum rotor torques and maximum rotor thrust? This is, of course, basic information for the design Not only do these cases determine the strength requirements for the fuselage and the rotor mountings, but also for the transmission system and for the angular clearances required at the lag hinges

Maximum " g " cases occur in pull outs from descents, when the collective pitch is initially low and can be increased over the whole available range  If this is done quickly, so that the rate of sink is not greatly reduced, then high angles of attack and high thrust are easily obtained

An envelope of permissible speeds and rates of descent can be constructed, by considering initially that the collective pitch is raised at its maximum rate, while the cyclic control is moved in a variety of ways to give the helicopter a range of pitching accelerations  It is necessary then to stop or reverse the control movements so that the rotor remains inside its overspeed and underspeed limitations and in control  Also the helicopter must not be permitted to acquire displacements or velocities in pitch from which a recovery could not be made

Maximum torque design cases occur from manoeuvres at maximum power and may be calculated for the maximum rate of cyclic control movement for a helicopter having no artificial restriction on control rate, a movement from the trimmed position of 60% travel at a rate of 100% travel/sec is reasonable  Such a movement would perforce require smart remedial action since only 40% control range at the most would be available to reverse the pitching velocity

This analysis for Type 192 resulted in torques at the rotor mountings and in the synchronising shaft of between 1 7 and 1 8 times their maximum steady torques

*The Engine Controls*

The separation of the engines by a long synchronising shaft as typified in the Bristol family of tandem helicopter designs to date has its influence on the effects of transient engine torques  These effects are very mild as it happens, and this I want to show

The twin-engined helicopter, pays a penalty in transmission and structure weight if designed to take the full 2½ min "emergency" rating of the two engines  Accordingly, the engines are "gated" in the twin engine regime and the helicopter is designed for steady powers less than the sum of the two 2½ min ratings  This raises very important considerations affecting safety, because one cannot allow a failure of the engine controls which would give too much power  At the same time the means of obtaining emergency power must be automatic so as to avoid overpitching and to minimise the loss of height in the event of an engine failure

The system developed for the Type 192 Helicopter in collaboration with Messrs D Napier & Son Ltd, is very successful and deserves description in some detail  For the Type 192, the twin engined power is limited to a torque equivalent to 1,020 s h p per engine at 260 rotor r p m  This gives adequate performance below the critical height for landing and above the critical height for climb away  The engines can give 1,300 s h p at 1 hour rating at sea level I C A N and 1,650 s h p for 2½ mins

The control is open loop and the engine throttle levers are connected mechanically to the hand throttles, the twist grip and to the collective pitch lever, so that engine throttle position is controlled by the sum of the 3 pilot demands

Now in the Gazelle engine, the free turbine is protected from overspeeding in the event of drive failure by a device which senses loss of torque meter pressure and operates the high pressure fuel shut off cock  Closure

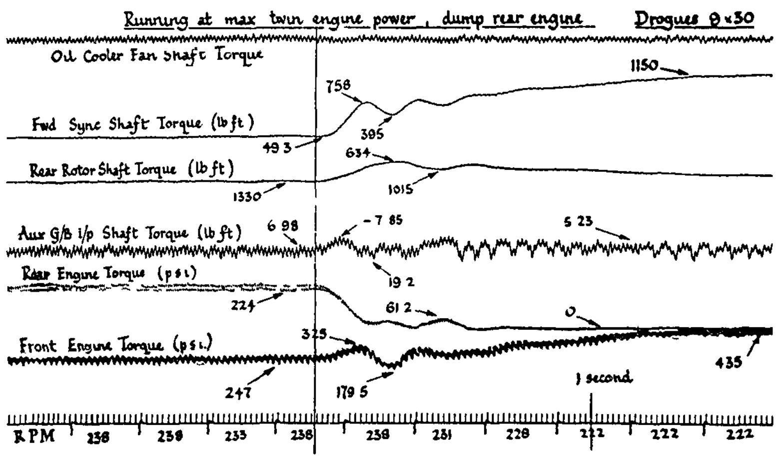


FIG 15 SIMULATED FAILURE OF THE REAR ENGINE AT MAXIMUM TWIN ENGINED POWER

( 2 1/2 MINUTE RATING GIVES 445 PSI TORQUE METER PRESSURE )

of this cock shuts off the fuel and also makes two series micro switches which energise a solenoid on the other engine, causing its throttle gearing to be doubled  Thus the second engine will double its power output  Provision is also made that when a switch on the cyclic control column, known as the " relight " switch is " on " the emergency throttle solenoid can be energised by moving the throttle of the other engine below the flight idle position

This facility permits single engined flight with the " emergency " throttle selected on one engine and with the other at ground idling  It also permits relighting without loss of emergency power, till the time comes to open up the relit engine  Furthermore, the feature gives the pilot a second way of obtaining emergency power

Fig 15 shows the history of a simulated engine failure at maximum twin engined power  The time from operating the high pressure shut off cock to the achievement of steady conditions again, is less than 2 seconds  In this time, the rotor r p m drops by 16 r p m  It will be seen that this happens without any overswing at all  At lower powers, when the total power required does not exceed the $2\frac{1}{2}$ min power, the rotor r p m drops less and then returns to the initial r p m  I believe that this feature will not only prove popular with pilots, but will set a new standard of twin-engined safety  Literally if a failure of one engine occurs, the other will take over with little if any change of r p m before the crew have realised what has happened  Moreover, there is no possibility whatever of the wrong engine being opened up

*Future Developments*

Future helicopters will have governed engines so as to relieve the pilot from the responsibility of maintaining rotor r p m within the allowable limits

In a multi-engined installation with the engines located at the front and



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rear of the aircraft either singly or in pairs, duplicated governors for each engine will be located at a common point in the transmission The obvious point is the centre of the synchronising shaft, since in this location the governors would not respond to either the 1st or the 3rd torsional mode This reduces the stability problem to that of a one engine, one rotor system, but it does require electrical signalling from the governors

### The Undercarriage and Ground Resonance

Following early troubles with the Type 173 Helicopter, the undercarriage was modified, giving low frequencies in the important rigid body modes of the helicopter when standing on its wheels The docility of this helicopter has been quite striking and was maintained throughout the range of wheel loading until the aircraft was fully airborne Furthermore, instability could not occur due to bouncing from wheel to wheel As a natural consequence, the same feature was sought for the later designs, Types 191 and 192

In specifying the requirements for the Type 192 undercarriage it was considered essential that the helicopter should be free from " ground resonance " for all possible load distributions and rotor thrusts and with any or all the wheels on the ground The helicopter should also be stable with any one tyre burst In addition, good static stability was demanded when resting on the ground, with the ability to stand up perpendicular to sloping ground

These requirements were met by designing the undercarriage such that

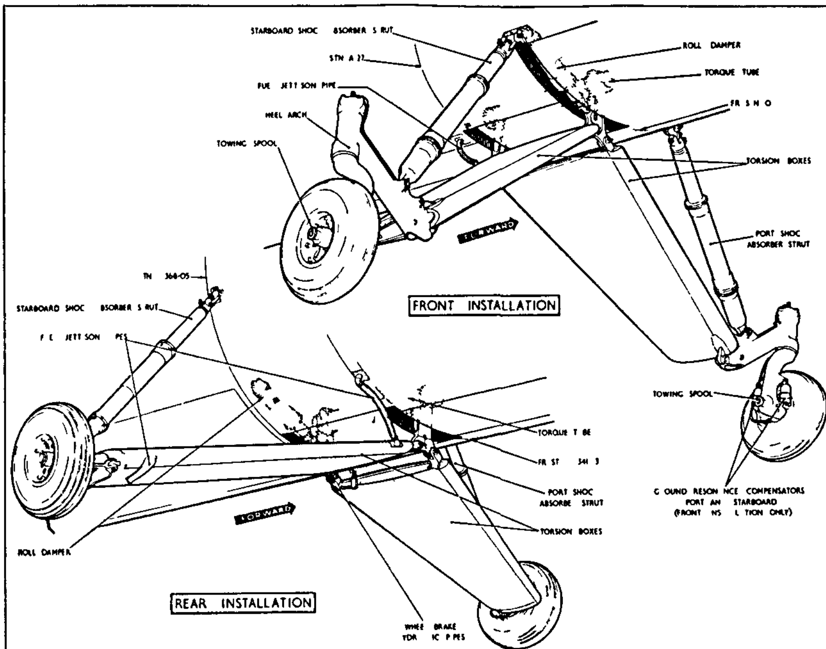



Fig 16 Alighting gear

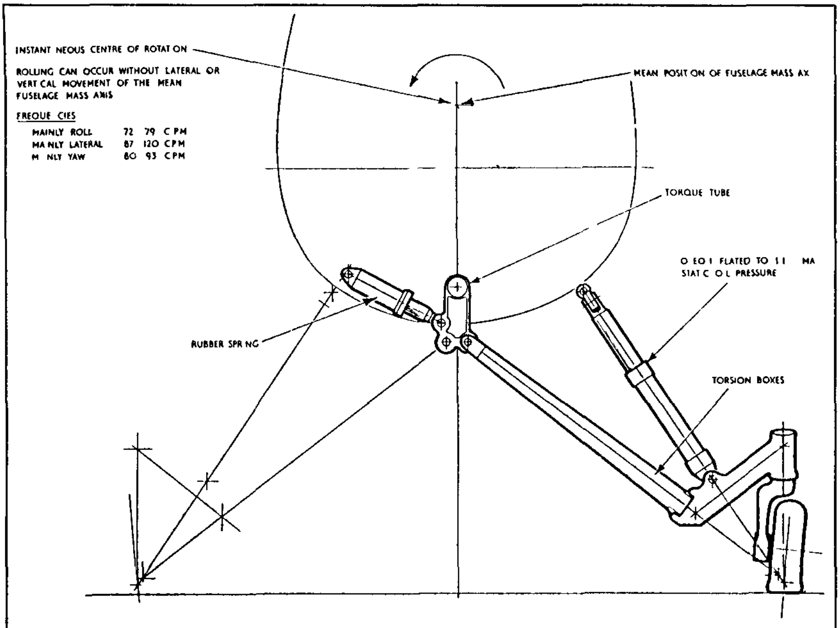
rolling of the fuselage relative to the wheels involved very little or vertical motions of the fuselage mass axis

In engineering this arrangement (Fig 16) the oleo legs were inflated to 1 1 times the maximum static pressure required and their axes made to intersect near the fuselage mass axis The torsion boxes which carried the wheel arch forgings at the front (and the wheel axle forgings at the rear) extended under the fuselage and were pinned to levers carried by two longitudinal torque tubes in the fuselage

This arrangement, shown in Fig 17 for the front undercarriage permits the fuselage to roll relative to the wheels without significant vertical or lateral displacement Thus, there is nearly zero gravitational stiffness and also very small inertia coupling between lateral and rolling motions of the fuselage Rolling stiffness is obtained entirely from two rubber springs which centralise the torque tubes

This undercarriage cannot be stressed in the conventional way for side and vertical loads An analysis was therefore made of the motion in drifted landings This showed that if the coefficient of friction between the tyres and the ground was 8 then a landing at 4 ft /sec downwards and 4 ft /sec drift would be made without exceeding the normal landing reaction factor

Fig 18 shows the modes of vibration concerned in the light weight and fully loaded cases These are compared with the mode shapes obtained in an undercarriage locked case It will be seen that the articulation has




Bristol 192 - Diagram of front u/c geometry

Fig 17

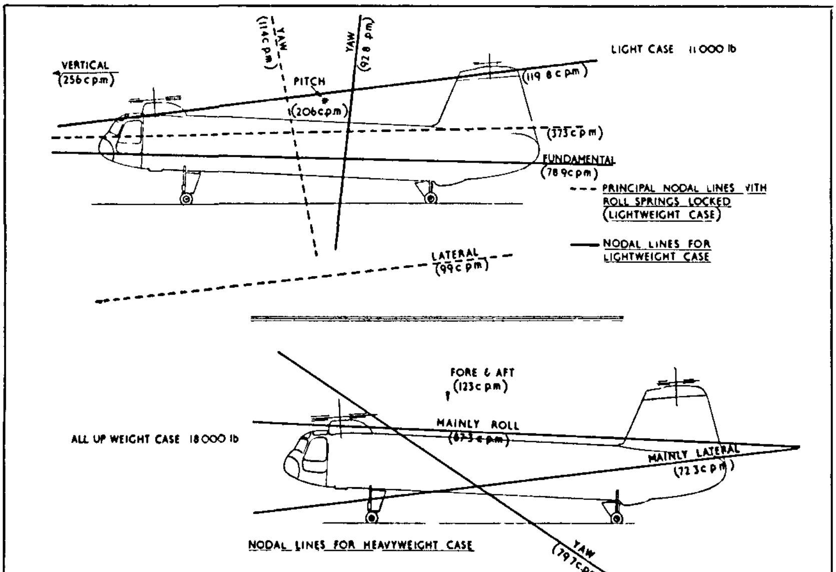


FIG 18. NODAL LINES FOR FUSELAGE MODES

reduced the high frequency mainly rolling mode from 373 c p m to 78 9 c p m

A further point of interest is that coupling between the rotor and the other lateral cum roll mode, is very small indeed In fact, the only significant modes are the yawing and the lower frequency rolling modes

*Theoretical Treatment*

In analysing the ground resonance problem for the Type 192 the correct choice of undercarriage and fuselage damping was found to be of prime importance Much of the analysis was performed by use of an analogue computor having 6 degrees of freedom and built to represent the flutter equations

$$[A]q + [\beta V_o + D]q + [CV_o^2 + E]q = 0 \qquad (1)$$

The velocity scalars $V_o$ and $V_o^2$ could be altered by a single multiposition switch

The ground resonance equation, however, can be written in the form

$$[A]q + [\beta\Omega + D]q + [C\Omega^2 + F\Omega + E]q = 0 \qquad (2)$$

by a transformation of the rotor blade co-ordinates into real co-ordinates representing the longitudinal and lateral displacements of the C G 's of the rotors Thus each rotor requires 2 degrees of freedom, leaving, in our case two for representing the fuselage

Normal modes of the fuselage were therefore used for these co-ordinates



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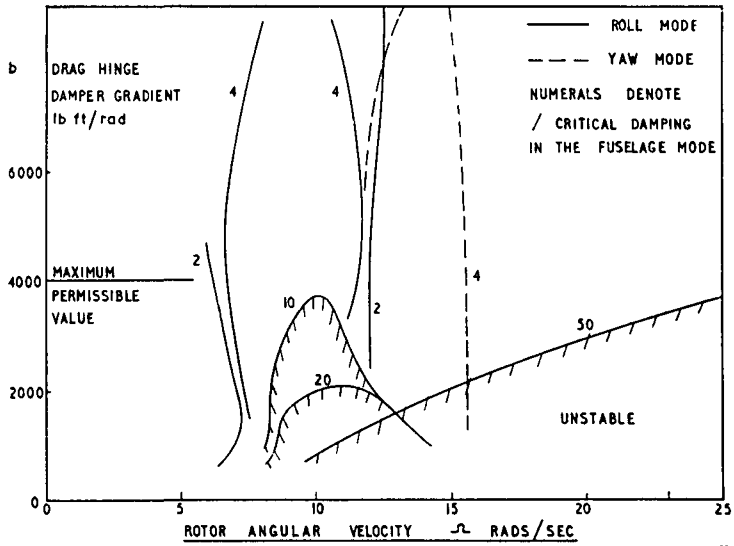
It was soon found that the coupling between these fuselage normal co-ordinates was so slight, for the damping co-efficients of interest, that each fuselage mode could be examined on its own

Now, Bristol Tandem Helicopters make use of a multiplate friction damper at the lag hinge, in which the friction torque increases in steps with lag angle The assumption is made that the damper can be represented by an " equivalent " viscous damper giving the same energy dissipation per cycle and by a spring of rate equal to the gradient of the damper torque displacement characteristic The damper gradient, b, cannot be included directly in the equation of motions (2) without increasing the order of the equation , solutions are therefore found for a range of values of b




TYPE 192 GROUND RESONANCE STABILITY BOUNDARIES FIG 19

The validity of the approximation was discussed by Dr Jones[2] for a single degree of freedom and this corresponds directly with the problem of torsional motion of the transmission In the ground resonance motions, the blades oscillate with a phase displacement , this may in fact reduce the errors involved , it will certainly cause coupling with torsional motions

*Simulator Solutions*

Fig 19 shows the stability boundary for the rolling and yawing modes in which rotor angular velocity is plotted against damper gradient, b, for various values of fuselage damping in the modes With the very low damping in roll, 4% critical, say, the system was unstable between $\Omega = 7$ and 13 5 radians/sec and little was gained by increasing rotor damping In fact b could have a value of 4,000 lb ft /rad , a limit imposed by consideration of the strength of the rotor blade and hub in the drag plane and by the lag angle over which the damper law must hold Thus to obtain stability, damping in roll must exceed 10% critical

However, if the damping was increased to 50% critical, for instance,



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then instability was found at $\Omega = 26$ radians/sec, within the range of flight rotor speed (23 to 27 rads/sec) moreover there was no upper end to the unstable range

A yawing instability was predicted from 11 8 to 15 6, this was very mild on the simulator and well below the operating range These results showed that it was very important to control fuselage roll damping within the safe limits What was more, the instability encountered with low fuselage damping was very mild and could be safely run through on starting and stopping, but the instability encountered with 50% critical damping was explosive

In order to ensure that the helicopter was stable at the limits of flight r p m latent roots of the equations of motion were obtained by digital computation for a 12 degree of freedom system comprising

4 rotor degrees of freedom

6 rigid body motions of the fuselage

2 degrees of freedom corresponding to the tilt of each undercarriage

*Testing*

To confirm that the helicopter was stable the rotors were run up without any fuselage restraints and attempts were then made, oscillating the flying controls alternatively in roll and yaw, to excite the corresponding modes In this test it was possible to maintain an oscillation in roll and yaw This showed that undercarriage friction was initially too great, in fact, there had been some pick up By changing bearing materials and increasing clearances, the friction was halved so effecting a complete cure

Exciting the aircraft by hand, the actual damping was found to be 4% in yaw and 14% in roll However, the two modes differed in that yaw required no movement of mechanical joints as happened in roll Hence the former was free of friction whilst the latter had a large frictional component depending on how much load the wheels carried This difference manifests itself when ground running the helicopter, in that a gentle yawing oscillation persists from 100 to 220 r p m This is stable, probably because of the " steps " of the friction damper Certainly, the rotor r p m can be maintained in the range indefinitely

The motion in roll is quite different Here the undercarriage and rotor both possess static friction, so for normal ground running, the motion is governed by the undercarriage locked case, and is stable

Now let us consider what happens on landing, that is when a large disturbance occurs Both the undercarriage and the rotor hinges are " unstuck " and the motion is stable, decaying rapidly As amplitude decreases, both the effective rotor damping and the fuselage damping increase due to the effect of Couloumb friction The motion remains stable, as the ratio of fuselage to rotor damping does not increase and so will die out completely This is achieved in Type 192 by making the first plate of the damper operate however small the lag hinge displacement may be

And so it appears that if static friction is present in either the undercarriage or the rotor it is necessary in both, otherwise the helicopter would always oscillate when running on the ground

## Integration of Flying Controls with the Rotor

It has been a primary aim in the Bristol range of helicopters to achieve a standard of manual control (that is a control without any powered assistance at all) which is acceptable at least as a stand-by control This aim has been pursued in the belief that whatever servo assistance is given, the ability to do without it if need be is a very valuable safety factor

That this can be achieved is demonstrated by the fact that over 100 hours have been obtained in Type 192 at 18,000 A U W with completely manual control This is possible only by the reduction of friction, for example by the use of tie rods in place of rotor blade thrust bearings, and by balancing out the spring terms in the controls arising from blade thickness, tie rods twist, propeller moment, and thrust moments and by good trimming facilities

It is also required that the shake of the pilot's controls shall be minimised, this can be done by the use of tuned " inertia dampers " giving very high impedance at the frequency of the control shake (rotor rotational frequency times the number of blades n $\times$ R)

If, on the other hand, powered controls are to be fitted, there are two possible locations They may be located at the rotor heads, when six actuators are required, or in the cockpit in front of the coupling mechanism, when four actuators suffice, one for each control channel (The coupling mechanism is a mechanical system which translates the pilot's commands into individual commands for each rotor )

*Control Modes of a Rotor*

Fig 20 shows the elements of the rotor control reduced to the simplest form, that is with collective pitch impedance represented by a spring $K_1$ and cyclic pitch impedance taken to be symmetrical and represented by the

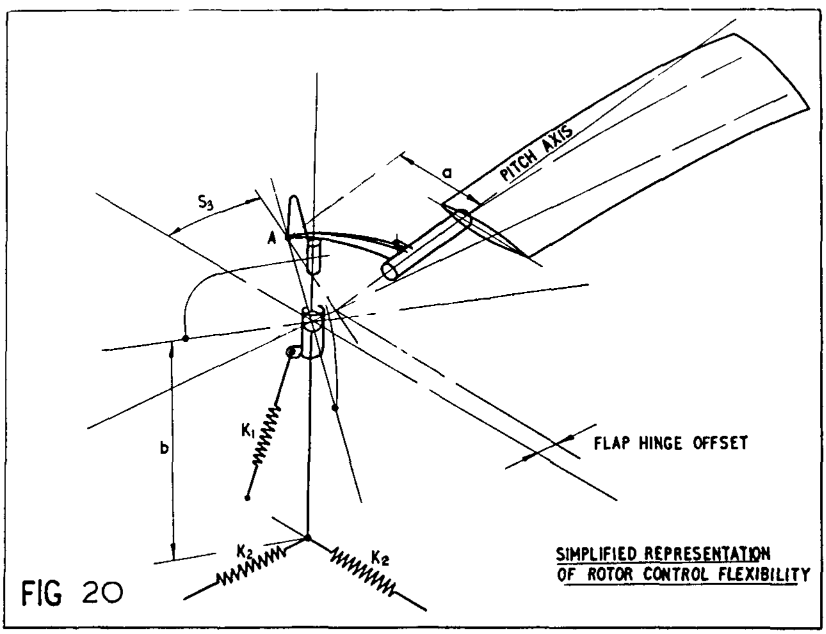



FIG 20

SIMPLIFIED REPRESENTATION OF ROTOR CONTROL FLEXIBILITY

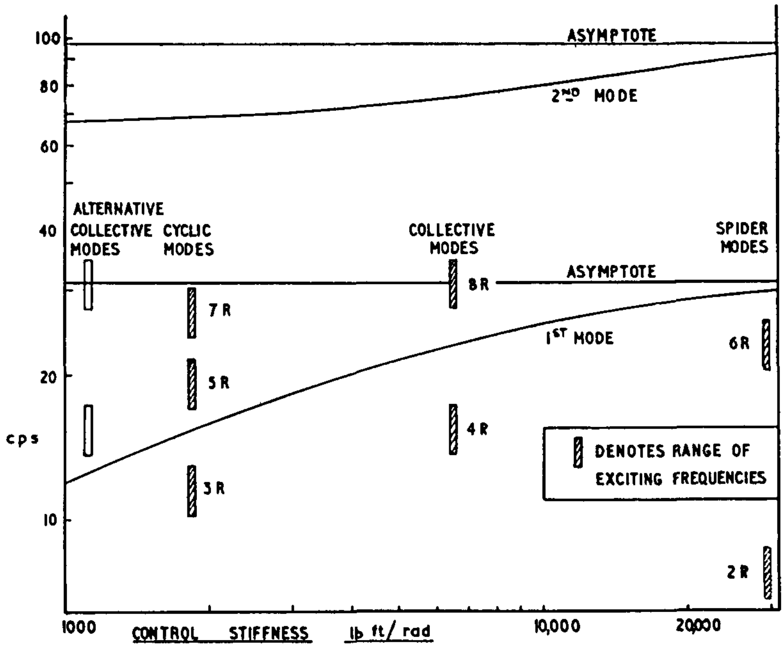



FIG. 21 CHOICE OF CONTROL STIFFNESS TO AVOID RESONANCE FOR 4 METAL BLADES

springs $K_2$ For our purpose, the spider mass will be ignored and we will consider that the effects of the flexibility of the spider arm and blade lever are included in the rotor blade torsional system

There are no possible ways in which the rotations of the roots of the blades may be related in phase , that is, each blade may lead the one behind it by $2\pi\frac{k}{n}$ where K is an integer and n is the number of the blades This gives rise to four " control modes " which have to be considered The control stiffness per blade referred to the pitch axis will be

$K_1\frac{a^2}{n}$ when $k = n$, the " collective control mode,"

$2K\frac{b^2}{n}$ when $k = 2$ or $n - 1$, the 2 " cyclic control modes,"

infinity when $n - 1 > k > 2$, the " spider modes "

In all such modes, there is a balance of vertical forces and of moments acting on the spider so no coupling with the control circuit can occur

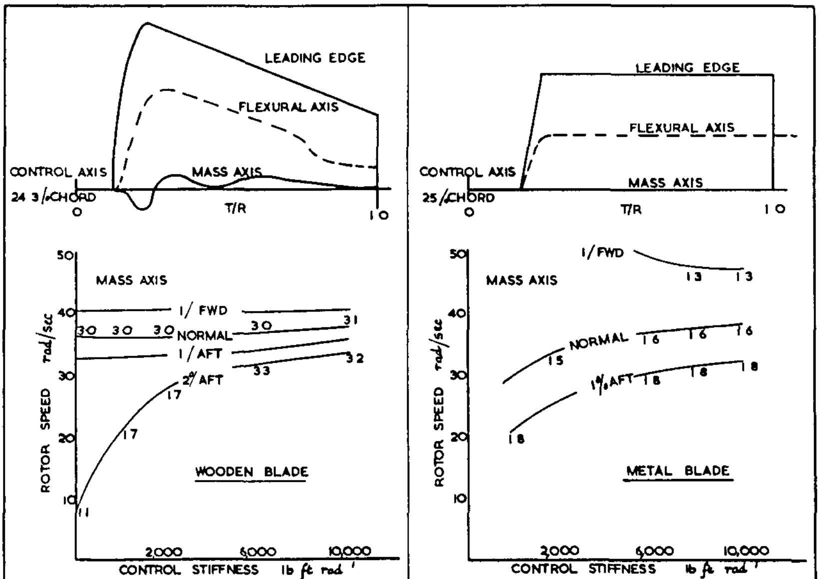
*Choice of Control Stiffness*

The frequencies of the 1st and 2nd torsional modes of a metal blade are shown in Fig 21 for a range of control stiffness referred to the pitch axis

Also shown are the ranges of exciting frequencies for a four bladed rotor This figure shows the big improvements in frequency separation which can be made by the choice of suitable stiffness for the three types of mode In particular, if the control stiffness was equal in each mode, resonance could not be avoided without restricting the rotor r p m range

*The effect of Control Flexibility on Blade Flutter*

In general, flutter will take place in such a mode that the effective control




THE EFFECTS OF MASS AXIS LOCATION AND CONTROL STIFFNESS ON FLUTTER SPEED FIG 22

stiffness is least, although the reverse is a possible alternative Accordingly, the flutter of a blade must be examined for a control stiffness appropriate to each " control mode "

An investigation has been made of the effects of mass axis position and control stiffness on the flutter of two rotor blades, each having a rotor diameter of 49 feet

The analysis used was similar to that described by W E Hooper[4] The flutter equation was solved by an analogue computor with 6 degrees of freedom Freedoms used were flap, 1st and 2nd flap bending, control circuit displacement and 1st and 2nd free-free torsional modes Two dimensional aerodynamic coefficients were used, calculated for frequency ratios of 3 for the wooden blade and 2 for the metal blade

The results obtained are compared in Fig 22 The first point to observe is that both blades, although normally mass balanced have finite, and quite low flutter speed This is due to the forward position of the flexural axis—a property which only has significance in relation to blade bending and, incidentally, one which is not at all tractable

For the wooden blade, if the mass axis was moved aft, then for a 1%

movement, the flutter speed was reduced equally whatever the control stiffness and the ratio of the flutter frequency to the rotor angular velocity was about 3 0 However, if a further 1% movement was made, then for a low control stiffness, a change in the flutter mode occurred, giving a drop in frequency ratio to 1 7 and a very severe fall in flutter speed The metal blade, however, gave a frequency ratio of 1 6 for the normal case and flutter speed was very sensitive to mass axis location In this case, it was necessary to bring the mass axis forward to prevent flutter

*Flight Evidence*

The relevance of this work was demonstrated to us during early flying with Type 192 when a lateral cyclic control twitch was felt from time to time This was cured by changing the front rotor for one which had its mass axis 1% further forward After the change, the aircraft has flown for 70 hours without hint of a further occurrence

It therefore appears that the incidents were due to flutter and indicates that adequate warning of the onset of flutter is available, at least with a manual control

## Conclusions

In this paper, only a brief look at some of the dynamic problems of interest in the design of a tandem rotor helicopter has been possible , much has been left out

Nevertheless, it has been the intention of this paper to give an idea of the way in which some of these problems are approached

Looking ahead, to faster transport helicopters the prediction of blade motion at high values of $\mu$ requires intensive study, for there are considerable gains in forced vibration levels to be obtained by the integration of the rotor, the control system and the airframe I believe also that stand-by manual control will be possible for larger helicopters than Type 192, although such machines will require duplicated power and ultimately, a full all-weather capability

I should like to state in conclusion that the views expressed in this paper are my own and not necessarily those of the Company

## Acknowledgements

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